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ORBITAL LAUNCH OPERATIONS

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VOLUME I

SUMMARY

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MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

PREPARED BY: SPERRY RAND SYSTEMS GROUP STAFF

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TABLE OF CONTENTS

Section		Page
I	INTRODUCTION	1-1
II	SUMMARY	2-1
	2-1. Operational Model	2-1
	a. OLO Vehicles	2-1
	b. Operational Schedule	2-3
	c. Operational Concepts	2-3
	2-2. Conclusions	2-6
	a. Communications and Telemetry	2-6
	b. Navigation, Guidance and Rendezvous	2-9
	c. Attitude Control	2-11
	d. Computers	2-12
	e. Checkout Equipment	2-13
	f. Secondary Power	2-15
III	DESCRIPTION OF RECOMMENDED SYSTEM	3-1
	3-1. Communications	3-1
	a. Operational Model	3-1
	b. Recommended System	3-2
	3-2. Navigation, Guidance and Rendezvous	3-3
	a. Operational Model	3-4
	b. Recommended Guidance Systems	3-9
	3-3. Attitude Control	3-17
	a. Operational Model	3-17
	b. Recommended System	3-17

TABLE OF CONTENTS (Cont)

Section		Page
	3-4. Computers	3-23
	a. Operational Model	3-23
	b. Space-borne Computers	3-24
	c. Earth-based Operations	3-27
	3-5. Checkout Equipment	3-29
	a. Operational Model	3-29
	b. Operations	3-30
	c. Description of Recommended System	3-32
	3-6. Secondary Power	3-35
	a. Vehicle Power Requirements	3-35
	b. Description of Recommended System for Spacecraft, Astrotug and Upcoming Payload	3-36
IV	COST AND SCHEDULE	4-1
	4-1. Communications	4-1
	4-2. Navigation, Guidance and Rendezvous	4-1
	4-3. Attitude Control	4-2
	4-4. Computer	4-2
	a. Tracking Station Computer Subsystem	4-2
	b. Space-borne Computer Subsystem	4-3
	4-5. Checkout Equipment	4-4
	4-6. Secondary Power	4-6

LIST OF ILLUSTRATIONS

Figure		After Page
1-1	Group Area Assignments	1-2
2-1	Operational Schedule	2-3
3-1	Basic Requirements	3-1
3-2	OLF-Satellite Ring - Ground Fence Concept	3-2
3-3	North-South Fence Geometry	3-3
3-4	Typical Space Relay Satellite	3-3
3-5	OLF Equipment	3-3
3-6	Recommended Payload Guidance System	3-10
3-7	Tug/Ferry and OLF Guidance Systems (Primary Mode of Operation)	3-13
3-8	Primary and Secondary Redundancy for the Tug/Ferry and OLF Guidance System	3-13
3-9	Spacecraft Guidance System, Block Diagram	3-16
3-10	Primary and Secondary Redundancy for the Spacecraft Guidance System	3-16
3-11	Alternate Configuration Matrix	3-17
3-12	Stabilization and Control Pitch Axis, Block Diagram	3-18
3-13	Stabilization and Control Yaw Axis, Block Diagram	3-19
3-14	Unmanned Payload Attitude and Vernier-Velocity Control System (No Tug)	3-20
3-15	OLF Equipment for Automatic Control of Unmanned Payloads	3-20
3-16	Space Vehicle Schematic	3-21
3-17	Minimum Attitude-Control System for Manned Astrotug	3-21
3-18	Augmented Attitude and Vernier-Velocity Control System for Manned Astrotug	3-22

LIST OF ILLUSTRATIONS (Cont)

Figure		After Page
3-19	Unmanned Payload Attitude-Control System (With Astrotug)	3-22
3-20	Operational Model	3-23
3-21	Initial OLF Data Processor and Control Computer	3-24
3-22	Central Computer Organization	3-26
3-23	Earth-Based Tracking Station Functional Diagram	3-28
3-24	Tracking Station Block Diagram	3-28
3-25	ADPAC Block Diagram	3-32
3-26	Portable Test Set	3-34
3-37	Troubleshooting With Portable Test Set	3-34
3-28	Transfer Function Measurements	3-34
3-29	Servo Voltage Measurement	3-34
3-30	Hydrogen-Oxygen System, System Schematic	3-36
4-1	Program Schedule - Telemetry and Communications	4-1
4-2	Projected Costs - Telemetry and Communications	4-1
4-3	Program Schedule - Navigation and Guidance System Development	4-1
4-4	Projected Costs - Navigation and Guidance Development	4-1
4-5	Attitude Control Development Schedule	4-2
4-6	Projected Costs - Attitude Control Development	4-2
4-7	Program Schedule - Computer Development	4-3
4-8	Program Schedule - Checkout and Countdown	4-4
4-9	Expenditures for Checkout and Countdown	4-5
4-10	Program Schedule - Secondary Power System Development	4-6
4-11	Projected Costs - Secondary Power System Development	

SECTION I

INTRODUCTION

As a result of the President's message delineating the national goal of placing a man on the moon by 1970, new emphasis has been placed on the NASA Apollo program. The Apollo program was originally intended as a lunar-orbiting vehicle with a three-man crew. This vehicle was to be boosted directly from Earth, orbit the moon, and return to Earth. In line with the course set forth by the President's message, the Apollo program became the logical space program for expansion to include an actual landing on the lunar surface. Accordingly, a NASA industry briefing was held in Washington in July, 1961. At this briefing, reliability and safety were strongly emphasized. These primary demands, coupled with the requirement for an actual lunar landing, increased the total fuel requirements and doubled the weight of the Apollo capsule. These facts aroused grave concern as to whether a booster of the Nova or Super Nova type would be ready in sufficient time to accomplish a lunar landing within the decade.

An alternate to a direct boost from Earth which began to appear attractive was the launch into orbit of major segments of the complete Apollo vehicle, assembly of these segments while in orbit, and then launch from orbit to the moon. Sperry Rand, recognizing the tremendous advances in the space sciences that would be required for such an operation, particularly in astrionics, and also realizing that its corporate structure was ideally composed for the undertaking of a task of this magnitude, initiated a corporate-financed study

of the problems associated with such an orbital launch. This study, in support of the Marshall Space Flight Center efforts, was guided by Vought Astronautics studies under contract NAS 8-853. Under the leadership and direction of the Sperry Rand Systems Group, the overall task was broken down into areas which best suited the particular skills of each of the selected divisions within the Corporation. The overall task was segmented as shown by figure 1-1.

The divisions were instructed to consider the formulation of solutions to the astrionic problems of getting men on the moon through the orbital launch technique as an objective and, in line with this objective, to consider all feasible concepts for accomplishing this goal. These concepts were to be investigated in sufficient detail to select the concept or concepts which had the highest probability of carrying on to fruition the stated task, and finally, to supply Marshall with reports and specific recommendations for further development of the OLO astrionic techniques.

The concept of launching a lunar vehicle from an Earth orbit can be implemented in a number of ways. A discussion of these various ways, which vary according to the booster size available, is contained in Vought reports which have studied the alternatives. It is the purpose of this seven-volume Sperry Rand report to explore the spectrum of Astrionics Equipment that is required to support an OLO program. To a large degree, this equipment will be independent of the number or size of the vehicles involved, or even whether a vehicle will exist or not. As an example, if it is decided that an Orbital Launch Facility is required, this orbital launch facility will require equipment for tracking

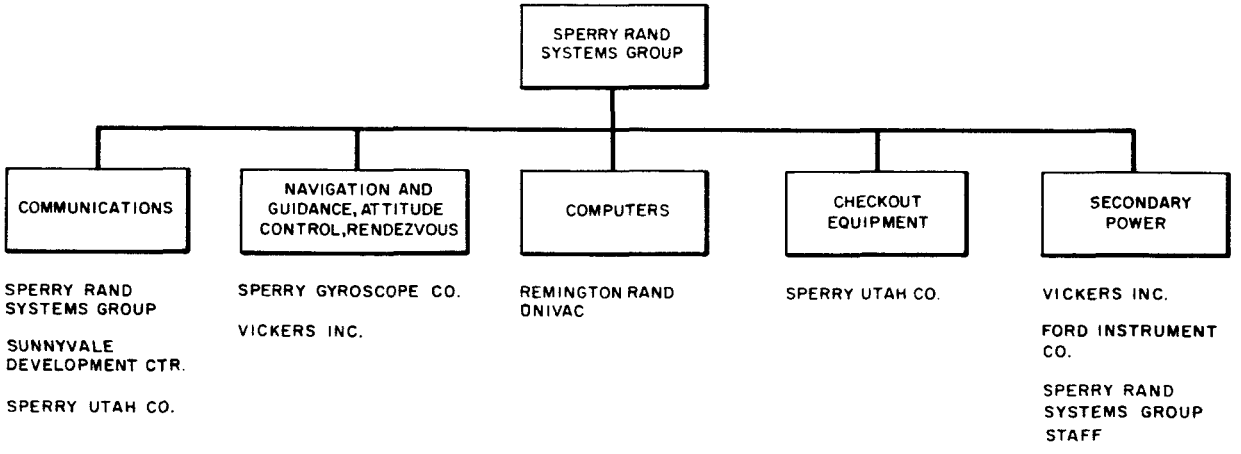


Figure 1-1. Group Area Assignments

and locating upcoming payloads. If there is no orbital launch facility, the task of tracking and locating the upcoming payloads falls to the space craft. It is true that the equipment required will be different because of the difference in interfaces and possible time-sharing functions, but the task of locating and tracking upcoming payloads must still be performed and the overall configuration of the equipment required to perform this function will not change.

The results of the investigations conducted are presented in seven volumes of which this summary is Volume I. Following is a complete list of volumes and the contributing Sperry Rand organizations.

Volume I	Summary	Sperry Rand Systems Group Staff Sperry Gyroscope Company Great Neck, New York
Volume II	Communications and Telemetry	Sperry Rand Systems Group Sperry Gyroscope Company Great Neck, New York supported by: Sunnyvale Development Center Sunnyvale, California Sperry Utah Company Salt Lake City, Utah
Volume III	Navigation, Guidance and Rendezvous	Sperry Gyroscope Company Air Armament Division Great Neck, New York

missing

Volume IV	Attitude Control	Sperry Gyroscope Company Air Armament Division Great Neck, New York supported by: Vickers, Inc. Detroit, Michigan
Volume V	Computer Requirements and Applications	Remington Rand Univac St. Paul, Minnesota
Volume VI	Automatic Checkout Equipment	Sperry Utah Company Salt Lake City, Utah
Volume VII	Secondary Power	Vickers, Inc. Torrance, California supported by: Sperry Rand Systems Group Staff Sperry Gyroscope Company Great Neck, New York

Proprietary

SECTION II

SUMMARY

2-1. OPERATIONAL MODEL

Prior to initiation of the Sperry Rand effort, an operational model had to be assumed. This operational model was generated in the main from the Vought Progress Report No. 4 dated September 27, 1961 and prior reports, and was expanded as necessary to accomodate a more detailed examination of the astrionics problem.

a. OLO Vehicles

Listed below are the vehicles which Sperry Rand considered as forming the total complex of the orbital launch operation.

(1) Orbital Launch Facility (OLF)

The orbital launch facility is the complex which is placed in orbit around the Earth. It can consist, in spectrum, of anything from a tool kit for OLV assembly to a structure housing station keepers, crew members, and the equipment necessary for Apollo mission support, with gradual evolution to an advanced depot for lunar and planetary mission support.

(2) Orbital Launch Vehicle (OLV)

The orbital launch vehicle is considered to be that vehicle which is used as the booster for the space craft for launch from Earth orbit in the immediate vicinity of the OLF.

(3) Space Craft (S/C)

The space craft is the vehicle which is boosted from orbit. In the preliminary stages of OLO this vehicle will probably be the Apollo capsule.

(4) Space Vehicle (S/V)

The space vehicle is the combination of the OLV and the S/C.

(5) Astrotug

The Astrotug is that vehicle which will be used to aid in retrieval and docking of upcoming payloads and/or other parts of the OLO complex. It will be a manned vehicle and, if Apollo derived, can also be used as the escape vehicle in the event of abort.

(6) Upcoming Payload (P/L)

Upcoming payloads consist of those vehicles which will be launched from Canaveral - either to the OLF, or to the desired orbit around the Earth. In general, these payloads will be unmanned, consisting primarily of equipment and/or fuel.

(7) Extra-Vehicular Man

The vehicle assumed for the extra-vehicular man is a space-scooter type vehicle containing practically no equipment other than manually controlled attitude and thrust jets and a radio beacon device for homing in on vehicles or objects

with transmitting beacons. This vehicle is assumed capable of coupling with and retrieving small objects less than 500 pounds in weight.

Since the ultimate quantity and size of vehicles or orbital units required will predicated primarily on booster availability and not on astrionic system weight, this report covers a matrix or combination of vehicles. Where possible, the penalty involved in not having a vehicle such as the OLF or Astrotug was noted.

b. Operational Schedule

The schedule upon which this study was based is shown in figure 2-1. This schedule was chosen from among several proposed schedules available because it required operational orbital-rendezvous capability and manned lunar-landing capability in the shortest period of time. If astrionics can be developed in this time scale, then astrionics systems will not be a pacing item in overall program planning.

c. Operational Concepts

(1) Communications

Communications was considered from the viewpoint of the need for continuous communications and cost. The continuous communications need arose from many considerations. Among these were mission reliability, the large amounts of data that need be transmitted, and cost penalties - including that involved if a very expensive payload were launched from Earth to the OLF complex at a time when, due to some last minute complication, the OLF was not in readiness to receive the upcoming payload. Also, with continuous communications,

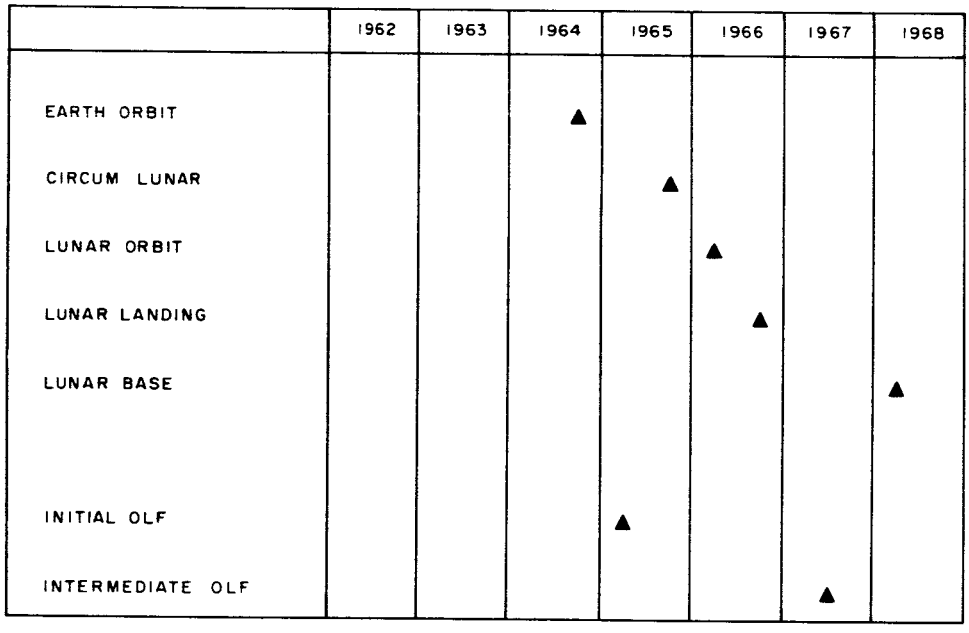


Figure 2-1. Operational Schedule

operational control can be maintained at one ground location until such time as the OLF evolves sufficiently to take over control.

(2) Navigation, Guidance, Rendezvous

Navigation, guidance, and rendezvous considerations, as stated previously, will most probably be decided on the basis of booster availability. Therefore, the ranges of the operational variables within which the best operating concept would probably lie was defined as follows: There will be two families of concepts, one including an OLF and one without an OLF.

Rendezvous was to be accomplished without use of an astrotug, if possible. A vehicle which could serve as an astrotug was to be available, however, on a standby basis. Gross rendezvous was to be accomplished by phase-change ellipses with all vehicles co-planar to within two degrees.

Orbital parameters assumed were a 350-mile orbit, probably non-synchronous, inclined approximately 28 degrees with respect to the Earth's equator.

(3) Attitude Control

Attitude control is a phase of system operation which requires continuous operation. Because of the fatiguing monotony of such a continuous function, man has been placed in the loop only for initiation of an attitude or to change the attitude of the vehicle when required. This concept of using man only for non-routine functions is the basis for attitude-control system selection.

(4) Computers

During the initial stages of OLO operations the ground tracking net may have to perform certain of the necessary guidance and navigation functions and, during certain modes of the operation, may also exercise command and control. The parallel evolution of the spaceborne equipments will tend toward independent operation of the various vehicles, with each being capable of satisfying the guidance, navigation, station keeping, energy management and communications requirements. The ground tracking net will then assume back-up and monitoring functions.

(5) Checkout Equipment

The concept envisaged for the OLO checkout equipment blended the man-machine relationship, utilizing the best capabilities of both. This mixing took advantage of man's ability to exercise judgment, to reason, to be flexible, and to effect repairs, thus improving the overall probability of mission success. Machine's contributions are speed, sequencing, accuracy, and the ability to perform an almost unlimited number of routine operations. The ultimate control was to rest at all times with the man.

(6) Secondary Power

Secondary power systems were not considered as separate entities having nothing in common with the other elements of the total OLO complex, but rather as systems the by-products of which could be used for other purposes. To this end, power systems were considered which could contribute to the life support requirements of the vehicle crews.

Specifically, this involved using combustive fuels, the by-products of which were water and oxygen which could be used by the crew. Also, to simplify the logistic and storage problem, common fuels were considered.

2-2. CONCLUSIONS

The astrionic aspects of the OLO program are within the time scale for state-of-the-art development as conceived by Sperry Rand Corporation. No major technological breakthroughs are necessary to successfully establish OLO operations.

Certain follow-on study work is absolutely necessary to firmly establish plans for successful OLO operations. These follow-on study areas are presented in this section of the Summary report.

Additional follow-on work is also recommended. These items, while not placed in the absolutely necessary category, are still considered necessary from the standpoint of well-balanced planning and proper initiation of work programs necessary to meet the objectives of the national OLO program on schedule.

a. Communications and Telemetry

The principal conclusions arrived at in the area of communications and telemetry are:

(1) Absolutely Necessary Follow-On Study Work

The requirement for continuous communications must be established in depth and fully documented since present

indications are that this communication continuity is a major requirement for successful OLO operations.

- The continuous system utilizes satellites to great advantage. Sperry Rand Corporation proposes to develop the specifications and schedules for the necessary satellites if required.
- Specifications and schedule for the complete communications system should be developed by June 1962 to meet 1965 circumlunar flight schedules.
- Realistic reliability studies are needed for the space-borne communications arrangement. These should include breadboarding and flight testing of key elements.

(2) Other Recommended Follow-On Work Is As Follows:

- The NASA plans for integration of communications, telemetry, ranging and range rate determination through the use of the same equipment using different forms of modulation should be incorporated in the OLO operation, if timely. The work that JPL, MIT and STL have been conducting indicates a strong trend in this direction but considerable study is required to see whether or not any of the concepts will be generally implemented soon enough.
- After the broad communication plan has been par tially detailed, a check should be made on uniformity and interchangeability in order to minimize the logistic problems aloft.
- Work should be continued for further investigation into the use of low-flying satellites as the transmission medium from outlying radar sites to the operation center.

- Methods should be investigated for reducing intentional interference from an enemy and also possible unintentional interference.

(3) Technical Conclusions

Some of the major technical conclusions which are of interest to the Marshall Space Flight Center as an aid in planning are listed.

- Continuous communications and telemetry is of paramount importance to the probability of OLO mission success. An adequate bandwidth, on the order of 75 KC, can be provided by low-flying satellites in the time period required.
- Switchable duplex circuits should be provided from the OLO operations center and computer to various portions of the OLO complex aloft in order to provide the necessary flexibility and allow for expansion if required.
- Low-flying satellites may be used to supply data input paths to the operations center from outlying radar sites.
- The low-flying satellite system is superior to the 22,300-mile high-flying system by a launch-weight ratio of about 21:1.
- R-f equipment should be readily tunable to three or more frequencies in the same band.
- The communications satellites are retrievable and repairable.

- Cost comparison with ground networks for continuous communication coverage show that communications satellites are considerably less expensive.

b. Navigation, Guidance and Rendezvous

(1) Absolutely Necessary Follow-On Study Work

(a) Formulate a detailed development plan for guidance for the OLO vehicles. This plan should have reliability as its primary objective, and should aim for proof of the reliability of each critical guidance element, by successful orbital flight tests, before OLO operations begin.

(b) Prepare specifications and make breadboards of the critical parts of the essential elements of the guidance system, specifically, a multipurpose radar; a manual star tracker mounted on an inertial platform; and the recommended guidance system for the unmanned payloads.

(c) Integrate the breadboards of the critical guidance elements into an operational simulator, to optimize the man-machine relationships.

(2) Technical Conclusions

Some of the major technical conclusions which are of interest to the Marshall Space Flight Center as an aid in planning the above work are as follows:

- The accuracy requirements of the OLO guidance systems can be met by equipment which is available or which can be designed using existing components and techniques.

- Reliability must be a major objective throughout the development, manufacture and test of the guidance equipment for OLO.
- Two specific guidance elements are required for OLO systems:

  A multipurpose radar for measuring altitude, vertical, angle to a rendezvous target, range and range rate; reliability is the primary goal in developing this radar.

  A manual star tracker mounted on an inertial platform. This is essentially an automatic star tracker in which the sensor and signal processing electronics are replaced by the astronaut's eye.

- The primary mode guidance system recommended for the manned vehicles in OLO consists of the two guidance elements just described, plus a digital computer and displays. Sufficient redundancy is added to attain the required reliability with full capability, plus additional back-ups with reduced capability.
- The same basic elements of the system will be used for all of the manned vehicles in the Orbital Launch Operation. This will allow considerable economies in development, production, training, and logistics, compared to the cost if different guidance systems were used in the different manned vehicles.
- The OLO operational concepts are not bounded by guidance. The recommended system can accomodate any of the probable operational concepts discussed in Volume III - existence or non-existence of an OLF, thrust capability of payload, etc.

- The guidance system recommended for unmanned payloads consists of a computer, three gyros three accelerometers, and a horizon tracker.

c. Attitude Control

The principal conclusions arrived at in the area of attitude control are:

(1) Absolutely Necessary Follow-On Study Work

- To establish the relative-attitude sensors and method of relative-attitude control required to operate two OLO vehicles in close proximity to one another.
- To prepare specifications for attitude-control equipment to be used in the initial orbital-launch facility, the spare vehicle and unmanned payloads.

(2) Other Recommended Follow-On Work

- To proceed with further development of storable bipropellant attitude controls.
- To establish the optimum attitude-control technique for use with a coupled tug/payload combination.
- To prepare specifications for the attitude-control equipment required in the intermediate orbital-launch facility, the intermediate unmanned payload, and the astrotug.

(3) Technical Conclusions

Some of the major technical conclusions, of interest to Marshall as an aid in planning the above work, are as follows:

- Development of special relative-attitude sensors is required for use during docking or joining of two vehicles.
- Electrical as well as mechanical interconnection of the astrotug and the incoming payload will be advantageous from the standpoint of fuel economy and equipment weight savings.
- When operating unmanned payloads, the attitude command function can be performed remotely from the orbiting facility. This should allow a significant reduction in the weight of payload attitude-control equipment.
- Bi-propellants provide lowest weight and are the choice for the attitude control systems

d. Computers

The principal conclusions arrived at in the area of computers are:

(1) Absolutely Necessary Follow-On Study Work

- Operations analysis of the Earth-based tracking net coverage requirements as a function of tracking accuracies and prediction methods available.
- Initial investigations of the mission and conceptual designs of an Earth-based Central Operational Control of overall Orbital Launch Operations.

- Extended analysis of the subsystems to be interconnected, information flow, tasks to be mechanized, and associated duty cycles of the OLF-based computer subsystems as a function of the OLO program phasing.

(2) Technical Conclusions

Some of the major technical conclusions which are of interest to the Marshall Space Flight Center as an aid in planning the above work are:

- The navigation and guidance functions of the test and initial operational phases of the Orbital Launch Facility (OLF) should be provided by an Earth-based tracking net. This net could subsequently be employed throughout the evolutionary phases of the program as a monitoring and emergency back-up capability.
- General-purpose digital computer systems should be utilized as the integrating and directional components in this net.
- The initial OLF should be equipped with a computer subsystem capacility adequate for expected additional applications.
- A general-purpose, aerospace digital computer should be employed as an integrating and controlling component of the OLF subsystems.

e. Checkout Equipment

The principal conclusions in the area of checkout equipment are:

(1) Absolutely Necessary Follow-On Study Work

- Establish the firm operational concepts necessary for OLO operations so that detail program schedules, specifications, and funding requirements may be formulated. Specifications for checkout equipment must be prepared outlining complete requirements so that hardware design and development may proceed by the third quarter of 1962 in order that the program schedule be met.
- A more penetrating study of a reliability program must be made so that techniques may be established at the outset of the design program in order to assure maximum Probability of Mission Success.
- Work must begin on prototype hardware in order that common digital building blocks may be evaluated and reliability problems resolved prior to the development of formal system specifications.

(2) Other Follow-On Work Recommended

- A more penetrating study of maintenance and logistic techniques must be initiated to firm up packaging concepts prior to the start of the design program in order to minimize fiscal furdens and schedule problems at later stages.
- The area of human factors must be studied in more depth, including results obtained from Project Mercury and Gemini flight tests. This must be done to establish the degree of tradeoff between operator participation, automaticity and probability of mission s success.

(3) Technical Conclusions

Some of the major technical conclusions which are of interest to the Marshall Space Flight Center as an aid in planning the above work are as follows:

- The scheduled OLO operations can be met with the present state-of-the-art, provided the principal conclusions above are implemented.
- The relative role of man-machine has been established.
- The relative role of Earth-space check-out has been established.
- High reliability is attainable.

f. Secondary Power

The principal conclusions arrived at in the areas of secondary power are:

(1) Absolutely Necessary Follow-On Study Work

- Establishment of the design parameters necessary for OLO operations, so that system design can be initiated.
- Preparation of specifications for the secondary power systems and life-science support, outlining complete requirements for the overall OLO program.
- A more penetrating study of the OLF, S/C, Astrotug, and incoming payload requirements must be made so that maximum performance and reliability can be achieved.

(2) Follow-On Work Related to the OLF

Certain necessary work for the OLF is recommended.

- The shielding requirements for the Snap 8 power system have a considerable influence on the overall configuration of the OLF. A more penetrating study of shielding requirements and its effect on the arrangement and operation of the OLF is recommended.
- The development of a second-choice OLF power plant utilizing solar energy is progressing at a fairly rapid pace. This progress should be continuously monitored to take advantage of any major new developments.

(3) Technical Conclusions

Some of the major technical conclusions which are of interest to the Marshall Space Flight Center as an aid in planning the above work as as follows:

- The only secondary power source which presently fits the operational needs of the OLF is the SNAP-8 nuclear power source
- All other vehicles in the OLO complex can be supplied secondary power with a cryogenic $H_2 - O_2$ internal combustion engine. The advantages of using a similar power source include common spares, single fuel storage area, and considerable interchangeability.

SECTION III

DESCRIPTION OF RECOMMENDED SYSTEM

This section gives a brief description of the systems recommended in the astrionics areas studied. These recommended systems were selected from candidate systems. A more detailed description of selected systems and candidate systems is given in volumes II through VII of this OLO astrionics study.

3-1. COMMUNICATIONS

a. Operational Model

Figure 3-1 shows a model of the desired communications and telemetry system. High quality, reliable, and continuous communications is required between the vehicles in the OLO complex.

This system must provide 2-way data links and voice channels, data detection, processing and storage, and telemetry.

Various concepts and techniques were investigated and a recommended system was derived by considering the trade-off factors: performance, overall cost, reliability and life, maintenance and logistics, weight, size, and power requirements, and schedule.

The resulting system has the following outstanding characteristics

- High-quality, continuous duplex communications between all stations at all times

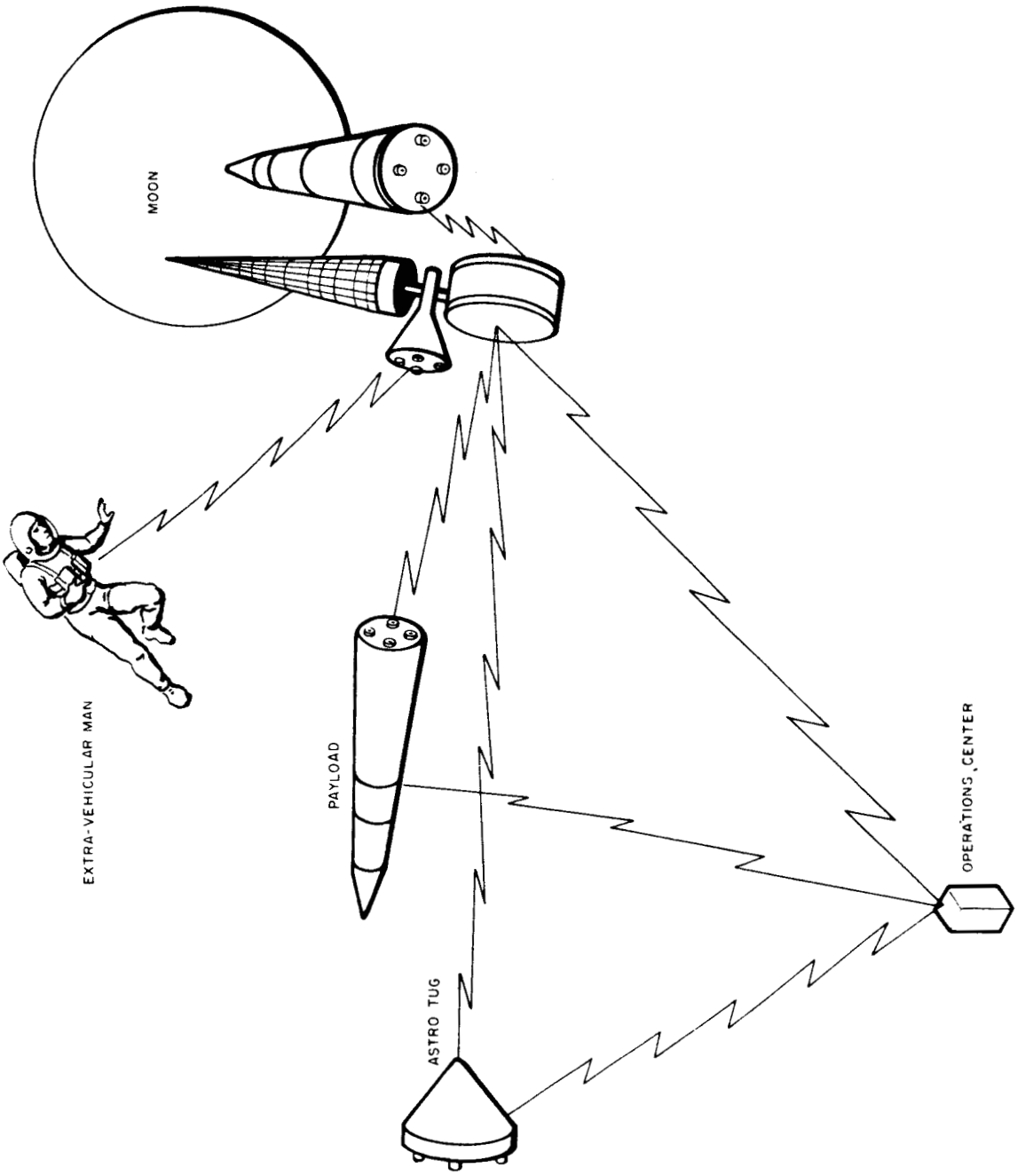


Figure 3-1. Basic Requirements

- Only one frequency band is required for the entire system, thus providing minimum number of antennas, single logistics, and ease of maintenance
- Every communications channel provides 50 kc or more of bandwidth for flexible multipurpose use.

b. Recommended System

(1) Communications Satellites

The communication system generated consists of a ring of 12 communication satellites coplanar with the OLF as shown in figure 3-2. These satellites are launched from the OLF. Each satellite is given a small boost of velocity of about 13.7 ft/sec with one satellite launched every three days. This spaces the satellites every 30 degrees around the earth and forms a ring which slowly rotates with respect to the OLF. Every 36 days each satellite comes sufficiently close to the OLF for retrieval and repair in the event of malfunction or for correction in the event of excessive deviation in position. The total weight of 12 satellites is on the order of 6,000 pounds which could be boosted with other equipment to the OLF. The system selected is shown in figure 3-2.

Directive antennas were used on the satellites, since the complication of orientation is considered well worthwhile in order to be able to achieve relatively wide bandwidth at a good signal-to-noise ratio.

(2) Ground Fence

A ground fence is utilized for ground-to-satellite communication. Although a minimum of three ground stations

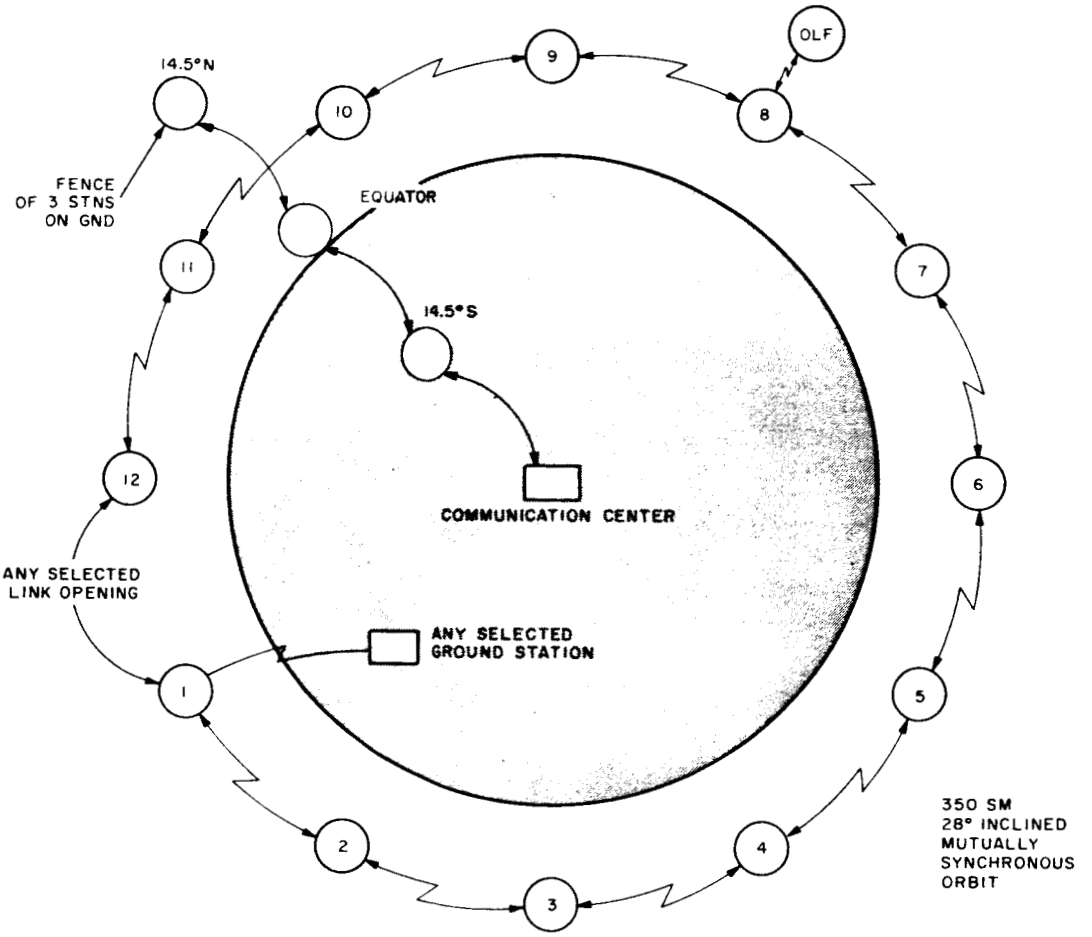


Figure 3-2. OLF-Satellite Ring - Ground Fence Concept

are required, a practical fence may include more than three stations. Figure 3-3 shows a fence using as many existing stations as possible, Antofagasta, Lima, Quito, Panama, Jamaica and the Cape.

Figure 3-4 shows the equipment required in each of the 12 satellites of the ring. There are four sets of duplex radios in each satellite, one looking forward in the ring, one looking backward in the ring, one looking up to the OLF, and one looking down to the ground fence. All the equipment is in the 2300-mc band. Eight-foot dishes are used for the paths around the ring, while a two-foot dish is adequate up to the OLF and a slot antenna is adequate looking toward the ground. The one-watt transmitters for transmission around the ring are operative all the time. The four-watt transmitter looking to the OLF and the two-watt transmitter looking to the ground operate on a 10 to 20 percent duty cycle. The OLF equipment for connection into the satellite ring is shown at the top of figure 3-5. This shows an eight-watt transmitter and a parametric receiver operating with an eight-foot dish.

3-2. NAVIGATION, GUIDANCE, AND RENDEZVOUS

Volume II of this study treats Navigation, Guidance, and Rendezvous. Volume III is concerned primarily with Attitude Control. In actual system design these two areas become much more inter-related. In a first phase of investigation both points of view must be thoroughly explored. This is accomplished in this study by denoting a separate volume to each subject. A more penetrating investigation

Figure 3-3. North-South Fence Geometry

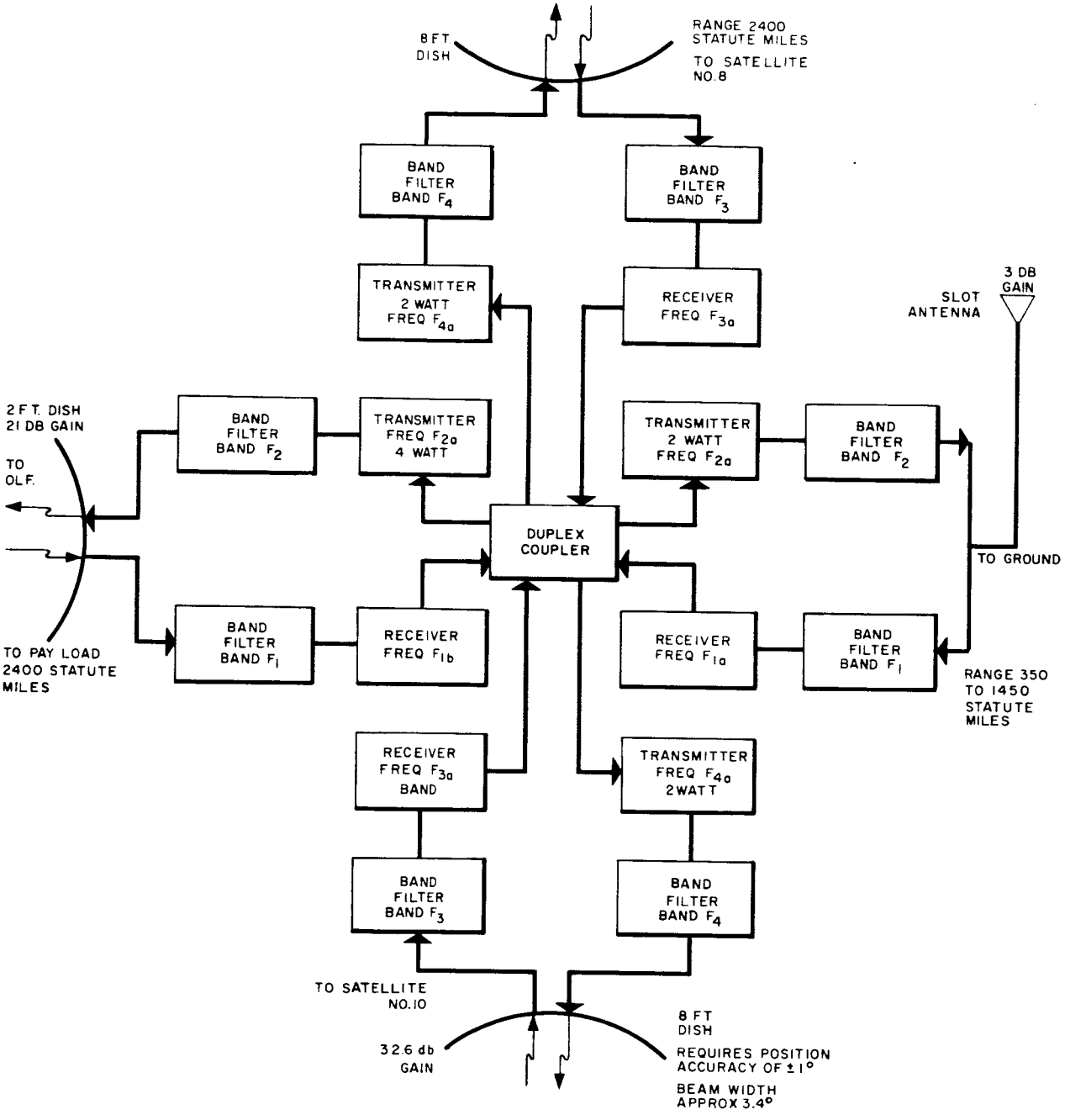


Figure 3-4. Typical Space Relay Satellite

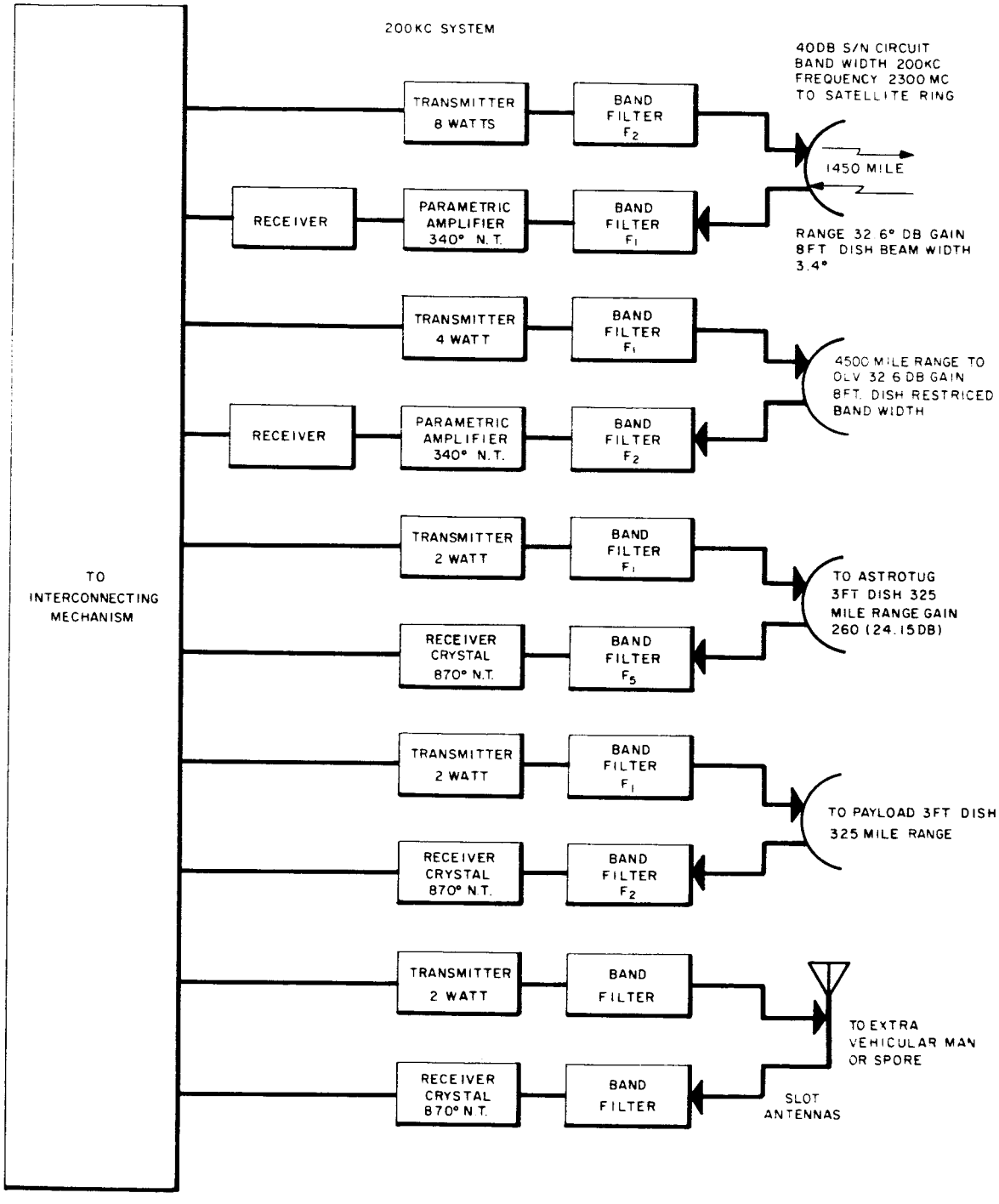


Figure 3-5. OLF Equipment

into reliability and operational requirements will probably result in considerable sub-system backup and time sharing between the two areas.

a. Operational Model

Families of operational concepts within which the best operation would be realized were defined. The factors which influenced the selection of a specific operational concept were, probability of mission success, human safety, rocket booster development and demonstrated space rendezvous involving man.

The study assumed that Saturn C-4 boosters would be used in the buildup of the space vehicle, and the three-man crew would be delivered into orbit by a tug or ferry powered by a Saturn C-1. The tug and ferry are assumed to be similar vehicles insofar as guidance is concerned.

(1) Operational Variables

The three operational variables which affect guidance are the number of rendezvous dockings which must be accomplished, the thrust capability of the payloads, and the allowable earth launch time deviations. When each of these variables is fixed, the operating concept is largely specified, insofar as guidance is concerned.

The number of dockings required for a mission serve to identify two families of operational concepts, those which include an OLF, and those which have no OLF.

The thrust capability of the payloads were divided into the three categories, fine vernier thrust control, gross thrust capability, and no thrust capability.

Deviations in launch time cause problems of two kinds. First, the launching point on earth moves away from the orbital plane of the target, and secondly, the target moves along its orbit, away from its planned location at the ideal instant of launch. In the first launch time deviations of no greater than eight minutes were assumed. The second effect is significant only if the target (OLF or S/V) is in a synchronous orbit. It was assumed that the launching was delayed from the ideal time by no more than two seconds. With this assumption the target will have moved no more than ten miles away from the point of arrival of the payload.

(2) Families of Operational Concepts

Two families of concepts were considered one including an OLF and one without an OLF. Within each of these, there are three possible degrees of payload thrust capability. Table 3-1 shows the vehicles required for each possibility, particularly the relation of the tug to the spacecraft and payload. In each case, it was assumed that the guidance would be sufficiently accurate to keep the OLF or spacecraft in a synchronous orbit if desired, and have the capability to bring the payload in from a considerable distance if it does not arrive close to the OLF.

(3) Sequence of Operations

(a) No OLF

Ascent

1) Fuel payload (#1) launched into orbit along preprogrammed path while tracked by the ground facility.

TABLE 3-1

OPERATIONAL CONCEPTS

Family	Payload Thrust Capability		
	None	Gross	Vernier
* NO OLF	• S/C must be the active vehicle in both the rendezvous and docking maneuvers with P/L. • Ferry vehicle may assist in docking. • Propellant transfer technique used in orbital buildup of space vehicle.	• S/C commands P/L to within 1 nm of S/C. • S/C provides the fine vernier thrust for docking with the P/L. • Ferry vehicle may assist in docking.	• S/C commands P/L to within docking range. • S/C need not conduct any thrusting prior to lunar launch. • Ferry vehicle may assist in docking.
* OLF	• Tug is required to retrieve and assist P/L in docking with OLF. • Tug must accomplish all thrusting required to move P/L.	• P/L can be commanded to within 1 nm of OLF. • Tug is required to assist in docking P/L with OLF. • Tug can command payload to thrust and can provide vernier control.	• P/L can be commanded to dock with the OLF with no assist from Tug. • Tug will be required for backup and emergencies. • Vehicle assembly technique used in orbital buildup of space vehicle.

NOTES:
1) * indicates concepts to be studied.
2) Assume phasing ellipse maneuver brings payloads to within a few miles of target.
3) 3 Man Crew is delivered into orbit in C-1 ferry vehicle.
4) S/C - space craft
5) P/L - pay load

2) Ground facility makes remote confidence checks of P/L #1 after injection into desired orbit.

3) Spacecraft and S/C payload (#2) are launched into compatible rendezvous orbit to within ten miles of P/L #1.

4) Ground facility makes remote confidence checks of P/L #2.

5) The Saturn C-1 crew ferry vehicle is launched into orbit approximately 24 hours after P/L #2 and follows the same ascent procedures to arrive in the vicinity of the spacecraft payload.

Terminal Rendezvous and Docking

1) The crew ferry vehicle determines and executes a thrust program to bring it within 50 feet of the spacecraft.

2) Docking is accomplished and one man enters the spacecraft to initiate detailed checkout.

3) After spacecraft is checked out it is stabilized and its' position and motion relative to the fuel payload are determined.

4) The spacecraft determines the thrust schedule required to rendezvous with the fuel payload within 50 feet. Spacecraft executes the schedule first checking the thrust control system.

5) Ferry vehicle may assist in docking by supplying vernier thrust control to the spacecraft or by serving as a remote observation point.

Propellant Transfer

1) Propellant transfer lines and the transfer would be controlled automatically by the man in the spacecraft. When completed, the remaining crew members enter the spacecraft.

2) Final preparations and checkouts are made in the spacecraft and the ground is informed of the status.

Launch Positioning

1) Crew ferry and empty fuel vehicles are moved away under command from the spacecraft.

2) Ground informs space vehicle of the precise launch time.

3) Space vehicle is oriented to proper attitude and is launched by the crew on command from the ground.

(b) OLF

This sequence of operations assumes that an OLF has been launched and staffed by the OLF crew. A manned long range (300 nm) tug is on standby at the OLF to assist in the rendezvous operation if the payloads fail to respond to the OLF commands.

Ascent

The payload vehicles follow essentially the same sequence developed for the NO OLF case except that the vehicle comes within range of the OLF's radar, the OLF verifies the vehicle's actual trajectory. If corrective action must be taken, the OLF determines new thrusting information and commands the vehicle accordingly.

Terminal Rendezvous

1) The OLF determines a thrust program to bring the vehicle cut from a maximum of 10 miles to within 50 feet of the OLF along a line-of-sight path, rather than using an orbital transfer. The thrust program calls for an initial maneuver to check out the vehicle's attitude stabilization and precision thrust control systems.

2) The vehicle is commanded along this path by the OLF to within about 50 feet and essentially zero relative velocity.

Docking

1) Suitable tow bars or cables are remotely attached to the vehicle and the vehicle is towed into the hanger.

2) Changes in the OLF attitude and station resulting from the docking reaction forces are sensed and corrected by the OLF.

(1) Recommended Guidance System for the Payload

Various types of payloads will be put into orbit during the OLO operational phase. Among these will be fuel payloads and portions of lunar vehicles.

The primary function of the payload system is to align the thrust vector of the vehicle along a line in inertial space and then to command, measure and cutoff the thrust once the required ΔV has been added. To perform this function, the payload guidance system must establish the inertial attitude of the payload. This is determined by using a horizon scanner to establish a vertical and three orthogonal gyros used in a gyro compassing mode to determine the reference in the orbital plane. The velocity-to-be-added determination is performed using three accelerometers along the three major axes of the payload and a digital computer performing the integration and computation functions. Figure 3-6 is a block diagram of the complete payload vehicle guidance system.

Table 3-2 is a tabulation of the weight, power required and volume for the recommended system. Table 3-3 summarizes the system functional elements.

(2) Recommended Guidance System for the OLF

The guidance system for the OLF must determine and control the attitude of the vehicle, determine the orbit of the vehicle, establish a line of sight to a target, and control ΔV's.

The guidance system recommended consists of the following elements:

1. Digital computer

Space Vehicle Assembly

1) The first payload to arrive at the OLF is the spacecraft and adapter. Fairly extensive checks are made immediately to uncover any malfunctions that cannot be repaired in the OLF.

2) As the remaining payloads arrive at the OLF and are checked out, the space vehicle is assembled and pre-launch checks are performed.

3) The flight crew transfers to the command module and preparations are made to move the space vehicle from the OLF to the orbital launch area.

Launch Positioning

1) Space vehicle is moved under control by the OLF to the remote orbital launch area.

2) Count down sequence is the same as the NO OLF case.

b. Recommended Guidance Systems

The guidance system configurations which are recommended for the four types of vehicles of the OLO complex are unmanned upcoming payloads, an Orbital Launch Facility, a tug/ferry and the spacecraft.

The recommended guidance systems fall into two classes: guidance for unmanned vehicles, and guidance for manned vehicles. For both classes of systems, the accuracy requirements are in general easily attainable with present day guidance equipment, while reliability is a major problem.

TABLE 2-4. OLF, TUG/FERRY, SPACECRAFT FUNCTIONAL REQUIREMENTS (Cont)

Primary Mode and Redundancy with Full Capability			Further Redundancy, Usually with Reduced Capability			
Guidance Function	Primary Equipment	Backup Equipment	Equipment	Remarks	Equipment	Remarks
Re-entry						
1) Attitude Reference	1) Gyro Stabilized Platform	1) Redundant gyro and Platform Elect.	1) Horizon Tracker and Periscope	1) This Back-up can be used during the initial phase of Re-entry. Can not be used in the atmosphere.	1) Vehicle Attitude Stabilization System	1) Coarse information
2) ΔV Control Computation Sensing	2) Digital Computer Accelerometers		2) Manual computation and control	2) Accuracy limited by tolerance on engine vernier thrust		
3) Acceleration (g load)	3) Accelerometers plus Computer					
4) Vertical Velocity	4) Radar Inertial Mixing	4) Redundant radar electronics	4) Radar or Accelerometer	4) Radar not usable during large period of reentry due to Plasma effect. Accelerometer has build-up error in pure inertial operation.	4) Ground Tracking	4) Ground can not supply complete coverage
5) Attitude	5) Radar Inertial Mix.	5) Redundant radar elect.	5) Radar or Accelerometer plus computation	5) Same as 4) above.	5) Ground Tracking	5) Ground can not supply complete coverage
Payload Functions						
1) Wake-up and Stabilize	1) Inertial Platform	1) Redundant Gyro Platform Electronics				
2) Establish Two Lines	2) Horizon Tracker Platform gyros (gyro compassing)	2) Redundant Gyro Platform Electronics				
3) Maintain Attitude	3) Inertial Platform	3) Redundant Gyro Platform Electronics				
4) Align thrust and measure.	4) Inertial Platform accelerometers and Digital Computer	4) Redundant Gyro Platform Electronics				
Midcourse Guidance Functions						
1) Determine Trajectory number of pos. fixes)	1) Multiple Image Optics plus manual aids		1) Manual Platform mounted Star Tracker	1) Platform must be energized. Since only one Star can be tracked at a time, considerable computation is required.	1) Periscope	1) Limited accuracy, requires computer since only one star may be tracked at a time.
2) Compute incremental velocities to correct Trajectory	2) Digital Computer		2) Manual Computation	2) Limited accuracy. Adequate to permit return to earth vicinity		
3) Sense Incremental Velocity	3) Accelerometers on Inertial Platform	3) Redundant Platform Electronics	3) Manual Computation and Control	3) Accuracy limited by tolerance on engine vernier thrust		

TABLE 3-4. OLF, TUG/FERRY, SPACECRAFT FUNCTIONAL REQUIREMENTS

Primary Mode and Redundancy with Full Capability			Further Redundancy, Usually with Reduced Capability			
Guidance Function	Primary Equipment	Backup Equipment	Equipment	Remarks	Equipment	Remarks
Orbit Determination						
1) Local Vertical	1) Multi-purpose Radar	1) Redundant Electronics	1) IR Horizon Tracker	1) Horizon Tracker can determine vertical to required accuracy	1,2,3 Ground Tracking	Number of ground stations presently available can not track continuously. A number of vehicle orbits must be permitted to achieve required accuracy.
2) Altitude	2) Multi-purpose Radar	2) Redundant Electronics	2) IR Horizon Tracker	2) Horizon Tracker requires considerable smoothing to achieve required altitude accuracy		
3) Two Lines in Space	3) Manual Star Tracker	3) Redundant Electronics and servos	3) Manual positioning and readout			
Attitude Determination and Control						
1) Two Lines in Space	1) Manual Star Tracker	1) Redundant Electronics and servos	1) Manual positioning and readout			
2) Maintain References	2) Gyro stabilized Platform	2) Redundant Gyro and Platform Elect.	2) Vehicle Stabilization System	2) Vehicle Attitude will be maintained to a few degrees		
ΔV Control						
1) Computation	1) Digital Computer		1,2 Manual computation and control	1,2 Accuracy limited by tolerance on engine vernier thrust		
2) Sensing	2) Accelerometers					
Line of Sight						
1) Acquisition	1) Multi-purpose Radar and Payload Beacon	1) Redundant radar elect. and payload beacon	1) Periscope	1,2,3,4,5, periscope requires that target be illuminated by Sunlight.		
2) Bearing angles	2) Multi-purpose Radar and Payload Beacon	2) Redundant radar elect. and payload beacon	2) Periscope	Usable range 30 NM	2,3,4,5 Ground Tracking plus visual observation at close range	
3) Range	3) Multi-purpose Radar and Payload Beacon	3) Redundant radar elect. and payload beacon	3) Periscope with Stadiametric ranging	3,4 Stadiametric ranging usable range of 5 NM		
4) Range rate	4) Multi-purpose Radar and Payload Beacon	4) Redundant radar elect. and payload beacon	4) Periscope with Stadiametric ranging			
5) Angular rate	5) Multi-purpose Radar Payload Beacon plus computer	5) Redundant radar elect. and payload beacon	5) Periscope			

TABLE 3-4. OLF, TUG/FERRY, SPACECRAFT FUNCTIONAL REQUIREMENTS (Cont)

Primary Mode and Redundancy with Full Capability			Further Redundancy, Usually with Reduced Capability			
Guidance Function	Primary Equipment	Backup Equipment	Equipment	Remarks	Equipment	Remarks
Terminal Guidance and Lunar Approach (Add. Requirements) 1) High Altitude Det. a) Set Radar Range Gate b) Determine Altitude range, range rate 2) Moon Observation	1a) Horizon Tracker 1b) Radar Inertial Mixing 2) Periscope	1a) Periscope with Stadiametric Ranging 1b) Redundant Gyro, platform and Radar elect.	1b) Horizon Tracker plus computer	1b) Limited Accuracy.	1b) Periscope with Stadiametric ranging	1b) Very limited accuracy

2. Multipurpose radar
3. Platform mounted manual star tracker
4. Horizon scanner (used for back-up only)
5. Transponder mounted on the payload

TABLE 3-2

SYSTEM CHARACTERISTICS
(not including digital computer or displays)

Vehicle	Wgt (lbs.)	Power (watts)	Vol (in^3)
Upcoming payload	33.8	85.2	999
OLF	91	260	4,770
Astrotug/ferry	91	260	4,770
Spacecraft	133	268	6,470

TABLE 3-3

UPCOMING PAYLOADS FUNCTIONAL REQUIREMENTS

Guidance	Primary Equipment	Back-up Equipment
1. Wake-up and stabilize	3 Gyros (3 strapped down)	1 spare gyro
2. Establish 2 lines in space	(1) Horizon tracker 3 scanning heads (2) Gyro compass using 3 gyros (strapped down)	fourth head if desired spare gyro

TABLE 3-3 (Cond)

UPCOMING PAYLOADS FUNCTIONAL REQUIREMENTS

Guidance	Primary Equipment	Back-up Equipment
3. Maintain Attitude	3 Gyros (strapped down)	spare gyro
4. Align thrust and compute	3 Accelerometers (strapped down)	-

The estimated reliability of the system is 0.086 for 20 hour operation. In order to achieve the estimated reliability, redundancy is provided for the platform, the pulse radar, and the beacon on the payload. Table 3-2 summarizes the weight, power, and other characteristics.

Table 3-4 lists the major functions performed by the OLF and the portions of the recommended system performing these functions. In addition, inherent or easily added back-up operation is indicated. Briefly each major function is performed as follows:

(a) Orbit Determination. A horizon scanner will be used to establish a vertical reference to the earth. The pulse radar has an altimeter mode of operation which will be used to determine orbit altitude. The additional required information will be supplied by tracking two stars with a manual star tracker.

(b) Attitude Determination and Control. The attitude determination function is achieved as a by-product of the

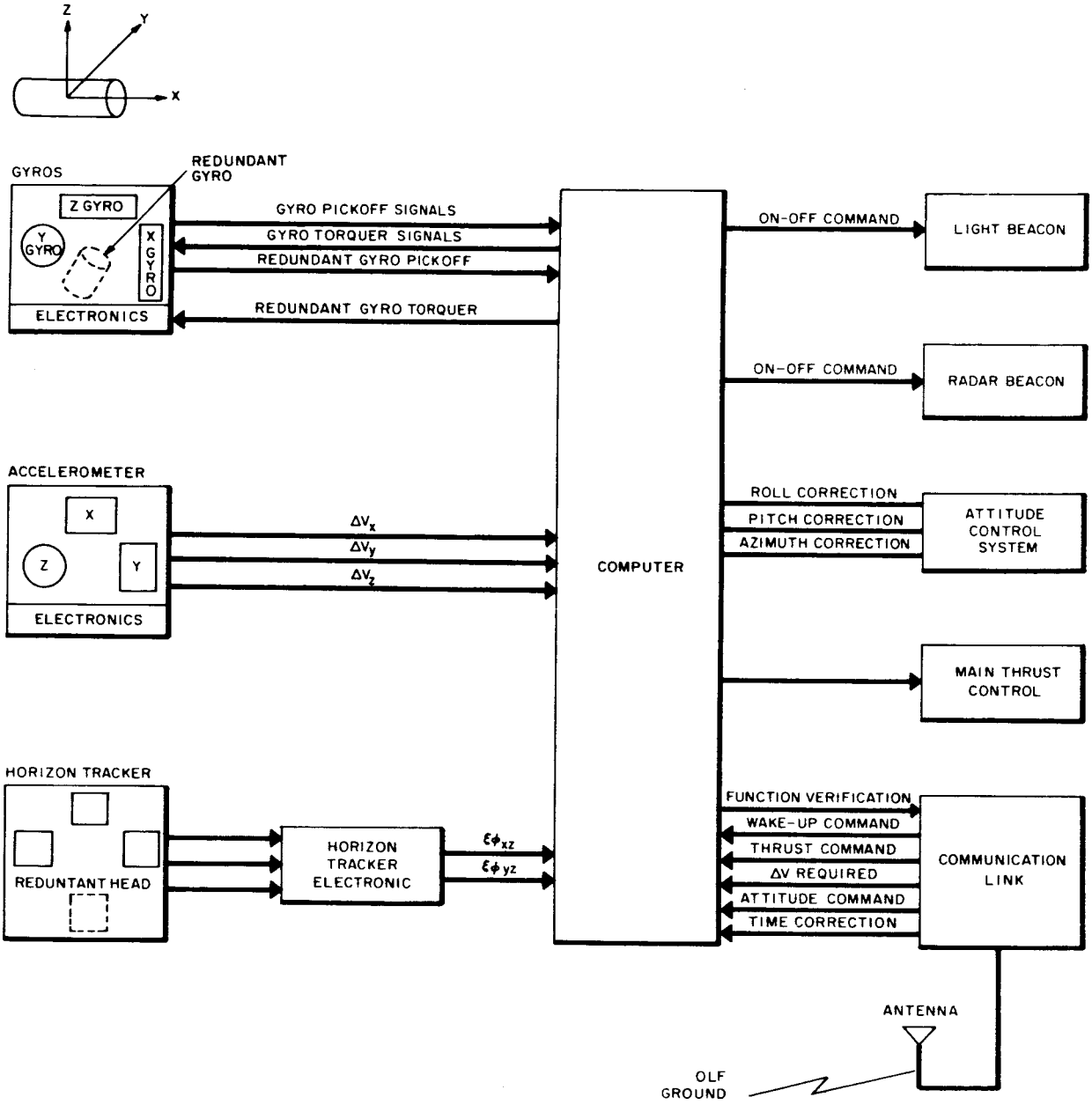


Figure 3-6. Recommended Payload Guidance System

orbit determination procedure, because of the system selected. The established references are maintained by an inertial platform.

(c) Control ΔV. The inertial platform will also contain the three accelerometers needed to measure the required velocities to be added for orbit corrections or changes. The accelerometer output will be integrated by a digital computer which will perform the required computation and control the thrust system of the vehicle.

(d) Line-of-Sight. Line-of-sight to a payload or manned vehicle will be accomplished by the pulse radar. The sighting angles, angle rates range and range rates will be continuously displayed to assist in the docking maneuver. Figure 3-7 is a block diagram of the primary mode of operation. Figure 3-8 is a block diagram showing the built-in redundancy.

(3) Recommended Guidance System for the Tug/Ferry Vehicle

The primary purpose of the tug is to assist in the rendezvous operation of two vehicles in the OLO comples, while that of the ferry is to deliver crew members to either the OLF or spacecraft. In order to accomplish these tasks, however, both vehicles must perform virtually the same guidance functions, so that the guidance system recommended is applicable to both the tug and ferry vehicles.

The tug guidance system must determine and control the attitude of the tug, determine the orbit of the tug, establish a line-of-sight to a target, control ΔV's, and determine and control the re-entry path. All but the last

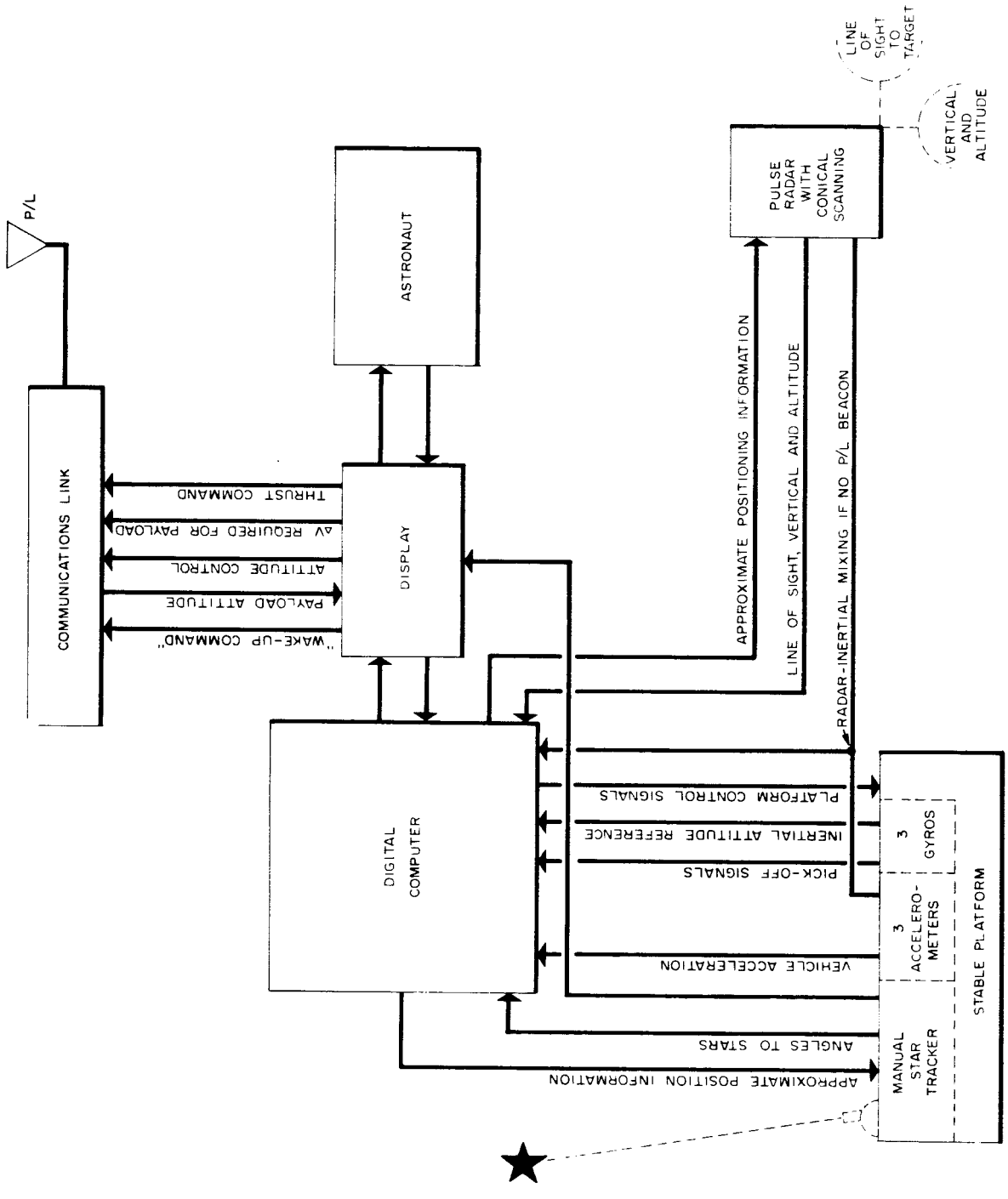
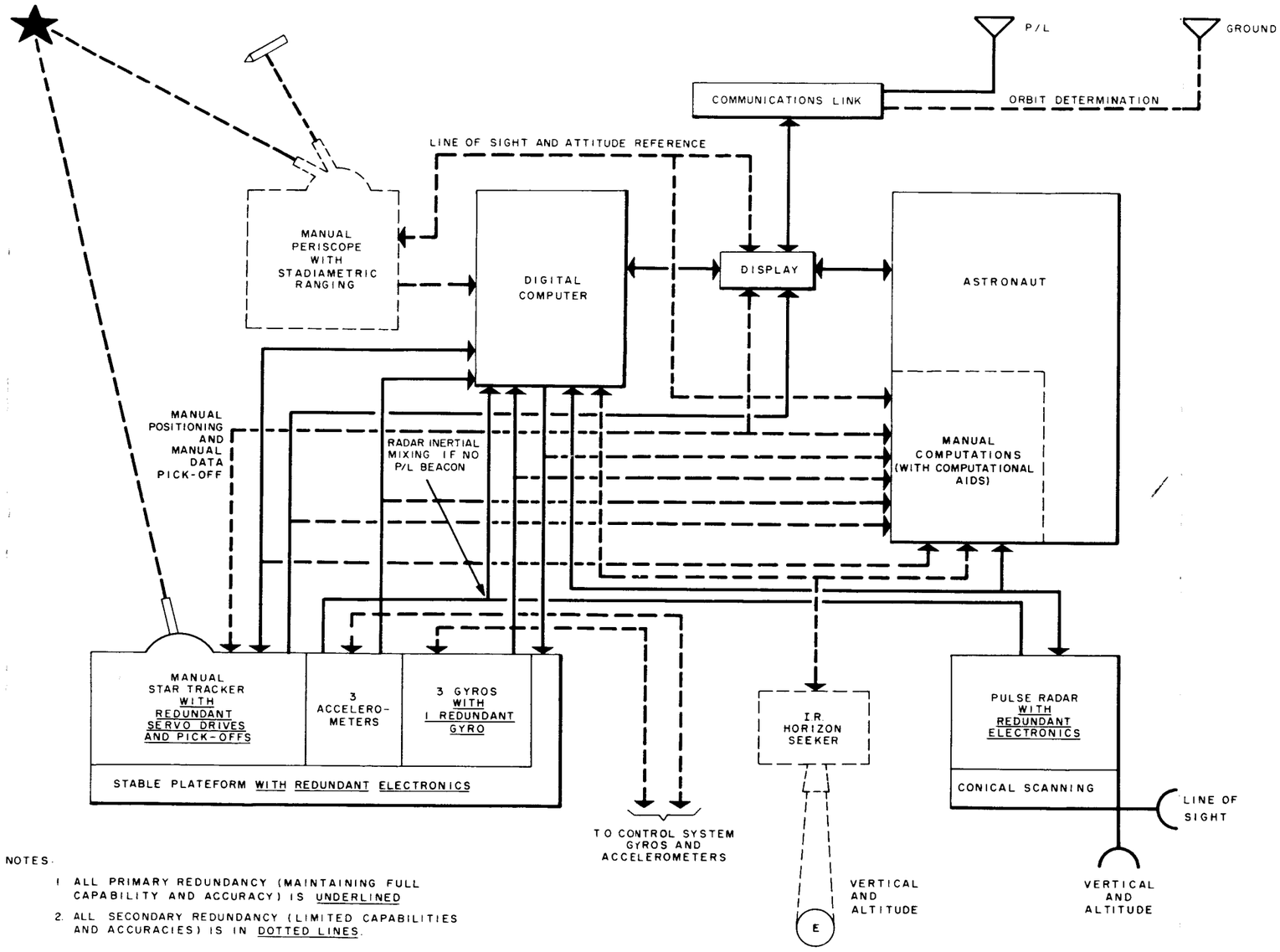


Figure 3-7. Tug/Ferry and OLF Guidance Systems (Primary Mode of Operation)



NOTES:

1. ALL PRIMARY REDUNDANCY (MAINTAINING FULL CAPABILITY AND ACCURACY) IS UNDERLINED
2. ALL SECONDARY REDUNDANCY (LIMITED CAPABILITIES AND ACCURACIES) IS IN DOTTED LINES.

Figure 3-8. Primary and Secondary Redundancy for Tug/Ferry and OLF Guidance System

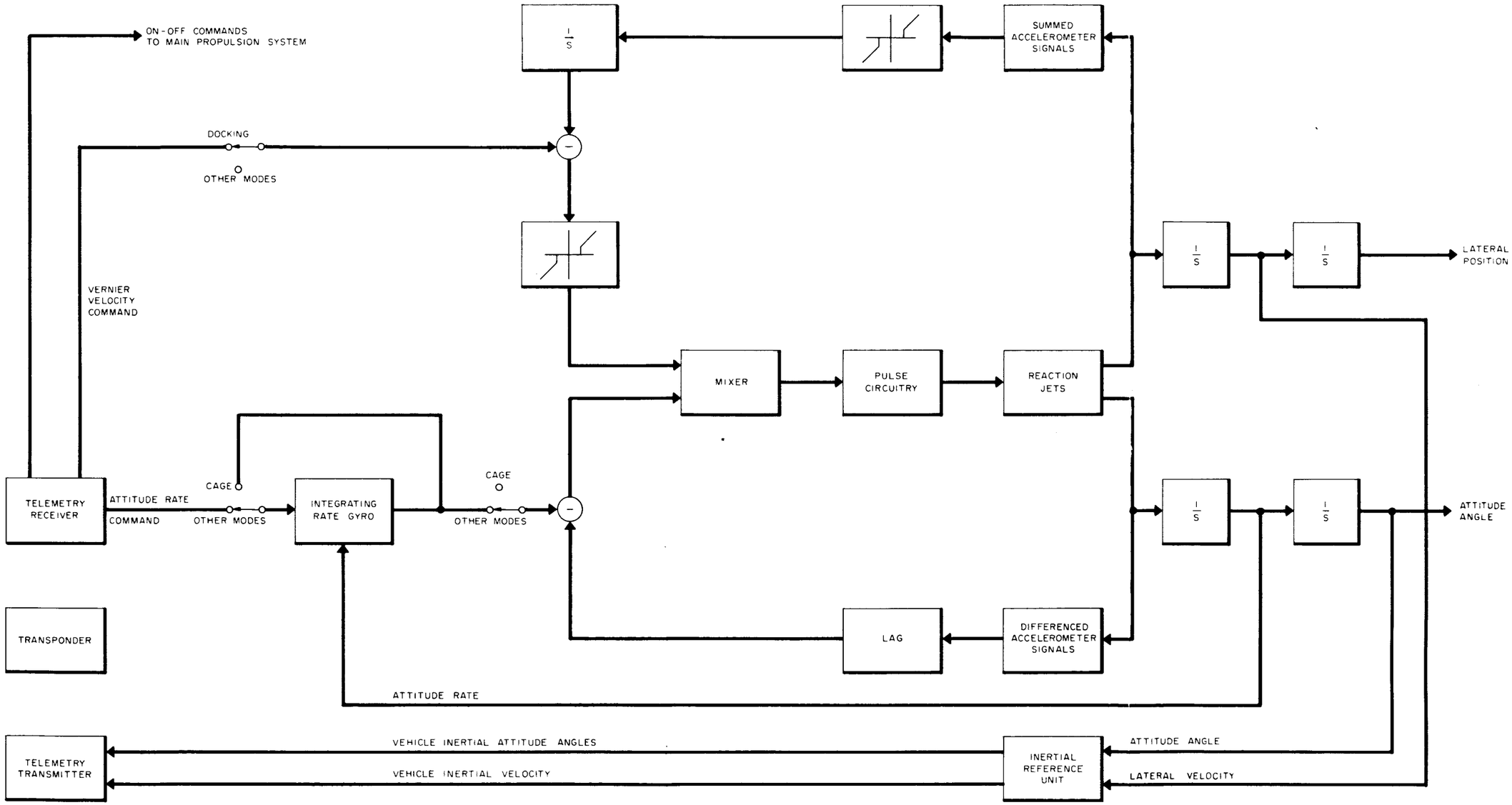


Figure 3-14. Unmanned Payload Attitude and Vernier-Velocity Control System (No Tug)

function are identical with those required of the OLF guidance system. When a tug is sent out to rendezvous with a payload, it essentially performs as an OLF.

The equipment required to perform the first four functions is sufficient to accomplish the last function (re-entry) with only minor additions to the computer and display requirements.

The re-entry function may be considered to consist of three separate phases: Computation of a de-boost program from a knowledge of the orbit parameters and the desired landing location, attitude control and measurement of ΔV during retro-thrusting, and control of the vehicle's flight path in the atmosphere.

Figures 3-7 and 3-8 are block diagrams of the tug/ferry recommended guidance system. Table 3-4 lists the functions performed by the tug/ferry and indicates those elements of the system required to accomplish each of the functions.

(4) Recommended Guidance System for the Spacecraft

The spacecraft vehicle will be assisted in its orbital maneuvering prior to orbital launch by either the crew ferry vehicle or the OLF and/or tug. For this orbital maneuvering the spacecraft's guidance system is essentially the same as the role for the unmanned payload plus the capability of precise line-up. For crew safety considerations however, the operational flexibility of orbit determination, the ability to establish line-of-sight to other vehicles, and the capability of re-entry into the earth's atmosphere

must be added. This gives the spacecraft the ability to perform the functions of the crew ferry vehicle as well.

The spacecraft guidance system must therefore perform the following functions:

(1) Determine the spacecraft orbit, approximately (accurate orbit information prior to orbital launch will be supplied from the ground).

(2) Determine the line-of-sight to other vehicles in the rendezvous operation.

(3) Perform the guidance functions of the unmanned payload during ascent into orbit.

(4) Determine and control attitude.

(5) Measure and control the velocity increment for orbital launch.

(6) Midcourse guidance

(7) Lunar approach and terminal guidance

(8) Earth approach and reentry guidance.

These functions include all of the functions required for the OLF plus a number of others. The recommended guidance system for the spacecraft was determined, then, by commencing with a system similar to that recommended for the OLF and examining two further considerations:

- What must be added to the OLF guidance system to enable it to perform the remaining functions required of the spacecraft in the Orbital Launch Operation?

- Can the resulting guidance system be efficiently combined with elements necessary to perform the midcourse and terminal phases (non-OLO) of the spacecraft mission?

Referring to the required guidance functions which are enumerated in this paragraph, the following equipment must be added to the basic OLF guidance system for each function:

1, 2	None
3	Convert IR horizon tracker to primary usage (presently used for back-up operation)
4	None (manual star tracker on inertial platform is sufficient)
5	Accelerometers on inertial platform must be more accurate and have greater range than in OLF guidance system.
6	An optical sensor must be added to the basic OLF guidance system.
7	Minor; additional function of multipurpose radar
8	Minor; similar to tug/ferry guidance system.

Table 3-4 gives the functions performed by the spacecraft and the components used for each function. Figures 3-9 and 3-10 are block diagrams of the selected system, both primary and redundancy or back-up modes shown, respectively.

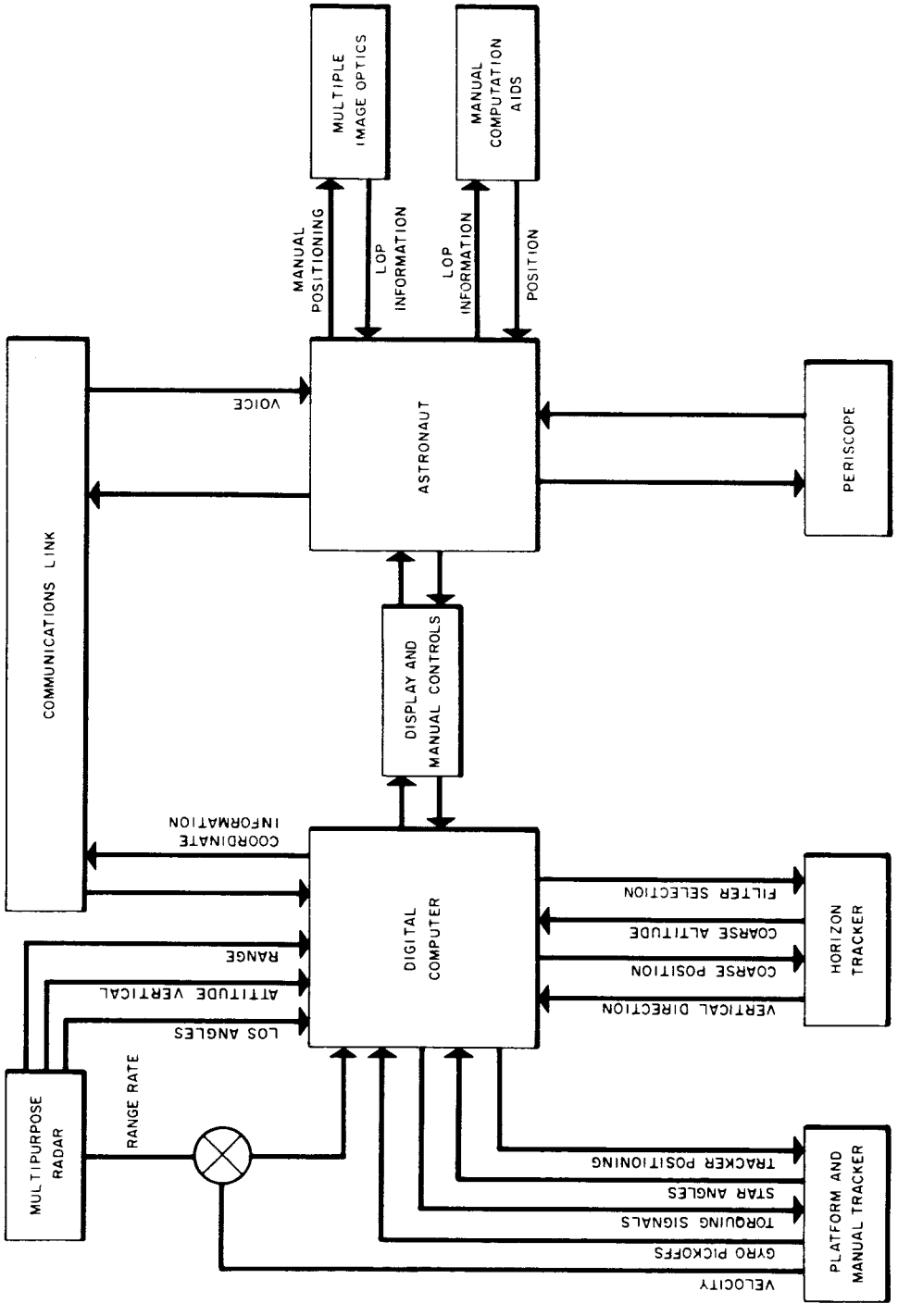


Figure 3-9. Spacecraft Guidance System, Block Diagram

3-3. ATTITUDE CONTROL

a. Operational Model

Three basic OLO complexes were considered: Systems without a permanent orbiting facility, systems with a permanent facility and no astro-tug, and systems with both a permanent facility and an astro-tug.

Figure 3-11 shows a matrix of the vehicles required in each of the alternate configurations. The checked combinations were selected for detailed study. At least one example of every control scheme under consideration is included.

The areas investigated in synthesizing attitude control systems for these vehicles were:

- Manual and/or automatic attitude and vernier-velocity controls
- Stability augmentation
- Reaction-control torquers and vernier-velocity thrusters.

b. Recommended System

(1) Orbital Launch Facility

The control system, completely contained in the OLF, starts to function at booster separation. The vehicle is aligned and controlled to the local vertical. Manual control and remote guidance input are provided to allow vehicle reorientation for rendezvous, payload docking and spare vehicle launching.

The system operating modes considered are initial stabilization and alignment, coarse attitude hold, docking alignment, fine attitude hold, and emergency manual control.

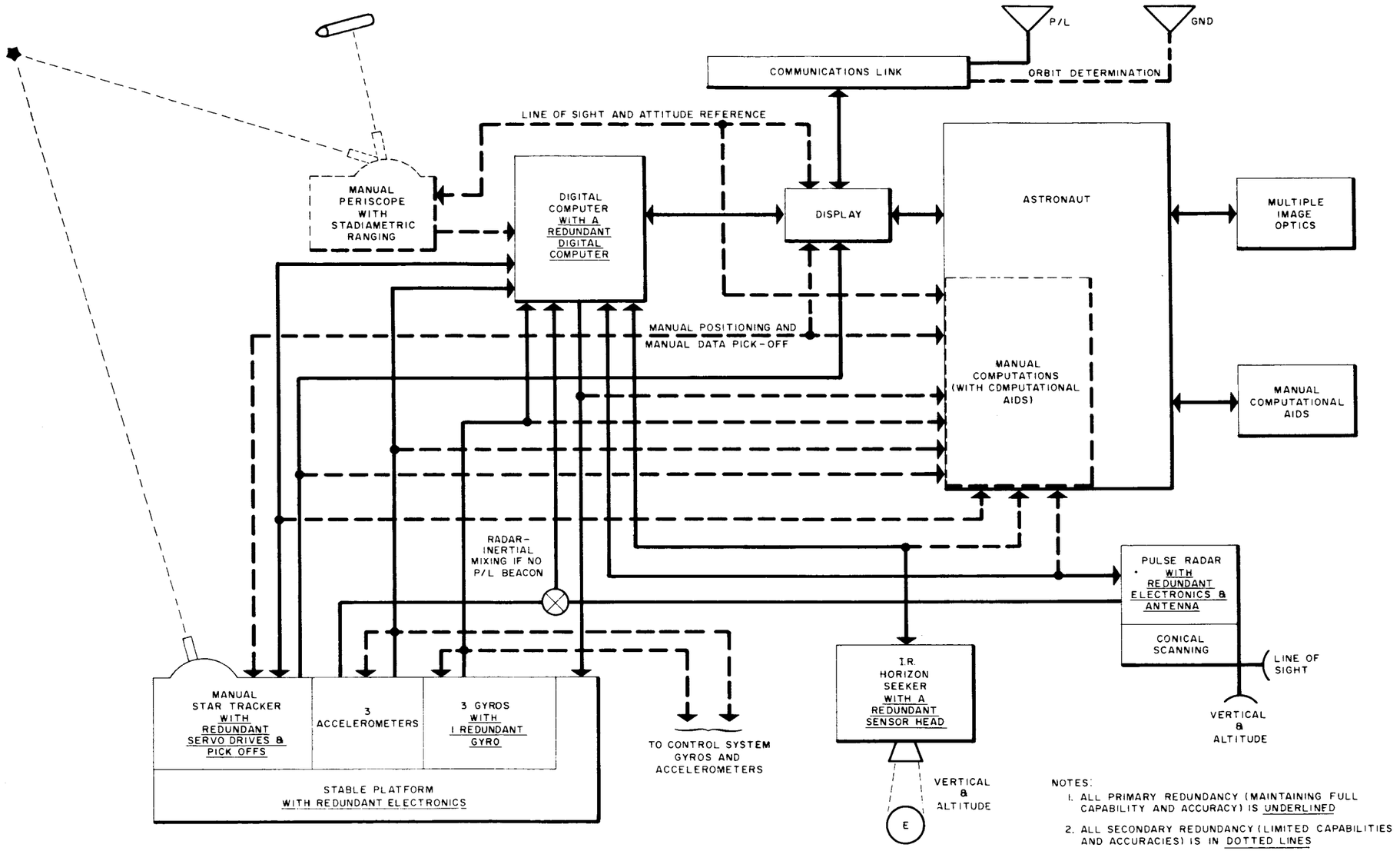


Figure 3-10. Primary and Secondary Redundancy for the Spacecraft Guidance System

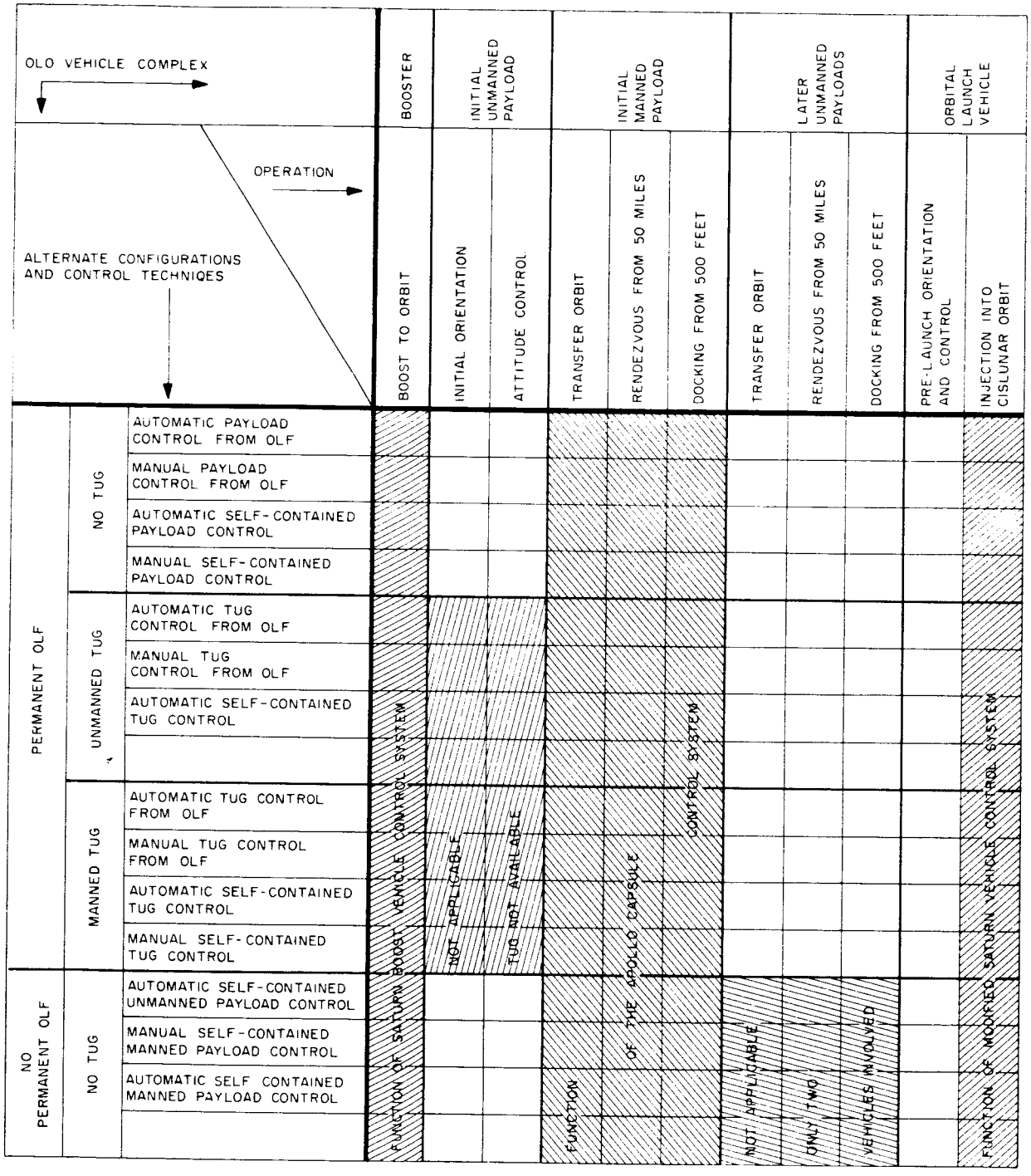


Figure 3-11. Alternate Configuration Matrix

(a) Pitch Axis (Figure 3-12) - The OLF's pitch axis is required to be automatically aligned to the local vertical. In addition, the system must be capable of manual or remote attitude reorientation. Figure 3-12 is a block diagram of the basic approach to pitch-axis stabilization and control.

- Initial Stabilization and Alignment - The OLF-booster separation will impart torques to the OLF that produce vehicle tumbling. The tumbling rates must be counteracted by the stabilization and control system to align the vehicle to the local vertical. The control signal for initial alignment is derived from body-mounted horizon seekers.

- Coarse Attitude Hold - The system configuration for coarse attitude hold is the same as for the initial alignment.

- Docking Orientation and Fine Attitude Control - During rendezvous and docking of a payload with the OLF, the OLF's attitude will be required to be automatically held to one degree or less, and the vehicle must be capable of being reoriented, remotely or manually, to facilitate the mating of the OLF to the incoming vehicle.

  By remote or local selection, the system's attitude and rate thresholds will be tightened to 0.5 degree and 0.05 degree per second, respectively, to provide a fine attitude control mode. The remote selection is required for rendezvous and docking of the initial incoming payload with the unmanned OLF. For a manned OLF, a manual controller is provided with a $\pm$ 30 degree attitude command capability. Manual gain programming is proposed in lieu of a more complex adaptive control.

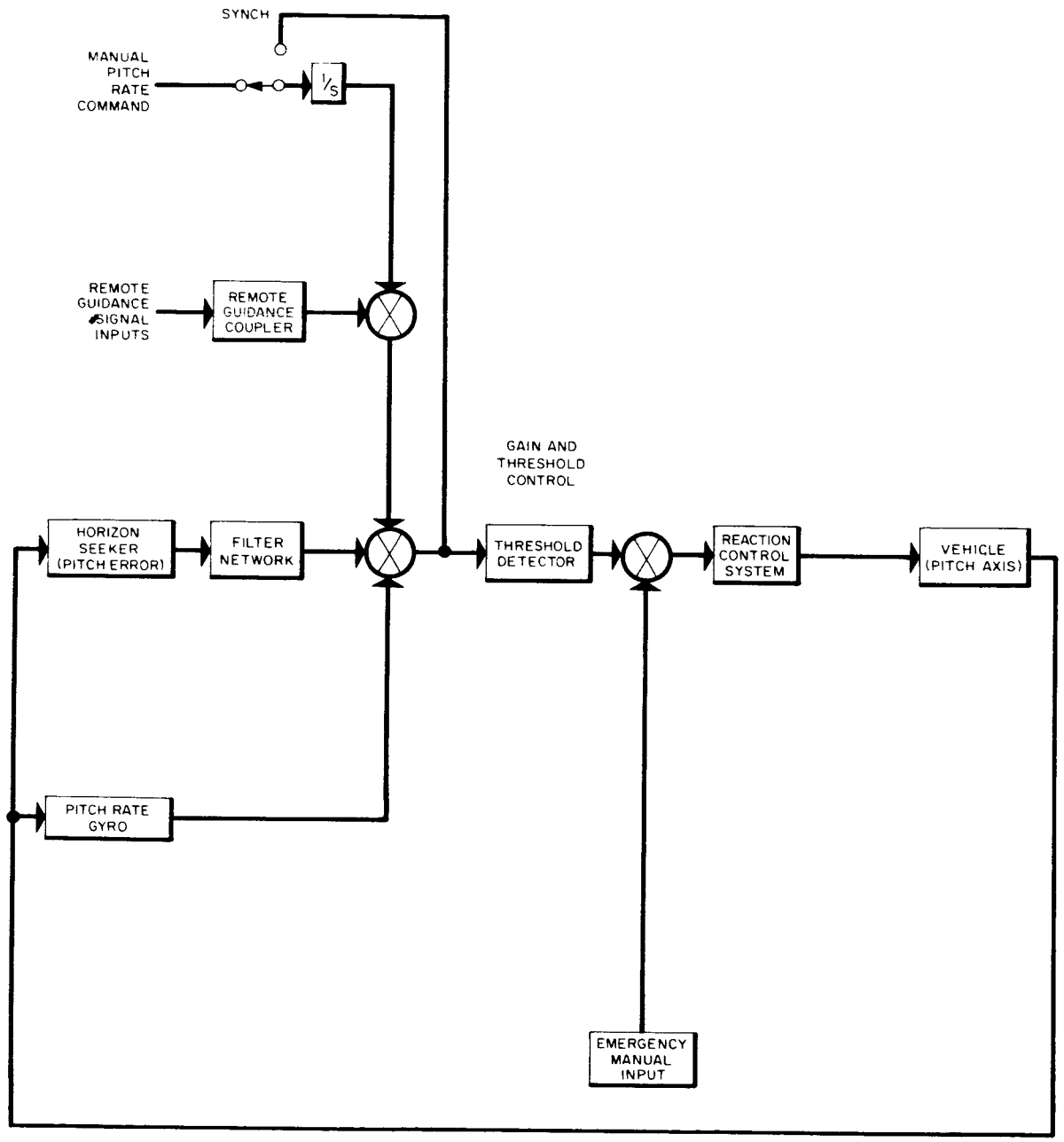


Figure 3-12. Stabilization and Control Pitch Axis, Block Diagram

- Emergency Manual Control - An emergency manual control capability is provided. Provision is made for direct coupling of an emergency control to the reactionjet system to enable manual override for OLF attitude control.

(b) Roll Axis - The roll axis is functionally the same as the previously described pitch axis.

(c) Yaw Axis - The yaw axis, shown in figure 3-13, functions in a manner similar to the roll and pitch axes except for its control reference(s) and method of manual control. The control reference for the yaw axis during initial alignment andperiodic realignment phases of OLF operation, is the forward velocity vector (orbit direction). Vehicle alignment to the velocity vector is desirable in order to provide proper pitch axis orientation with its attendent fuel efficiency.

Full-freedom (360 degree) yaw-attitude reorientation is available to facilitate payload rendezvous and docking. The vehicle should be manually reoriented to the velocity vector after completion of the docking maneuver.

(2) Unmanned Payload (Without Tug)

The unmanned payload rendezvous and docks with the OLF by means of commands generated in the OLF. During rendezvous, the attitude control system must position the payload in attitude for firing the main propulsion rocket. For docking, the attitude control system must be capable of translating the vehicle laterally without rotation and be capable of positioning the vehicle accurately with respect to the OLF. The control system must, therefore, accept attitude, lateral velocity, and thrust commands, and produce torques and lateral

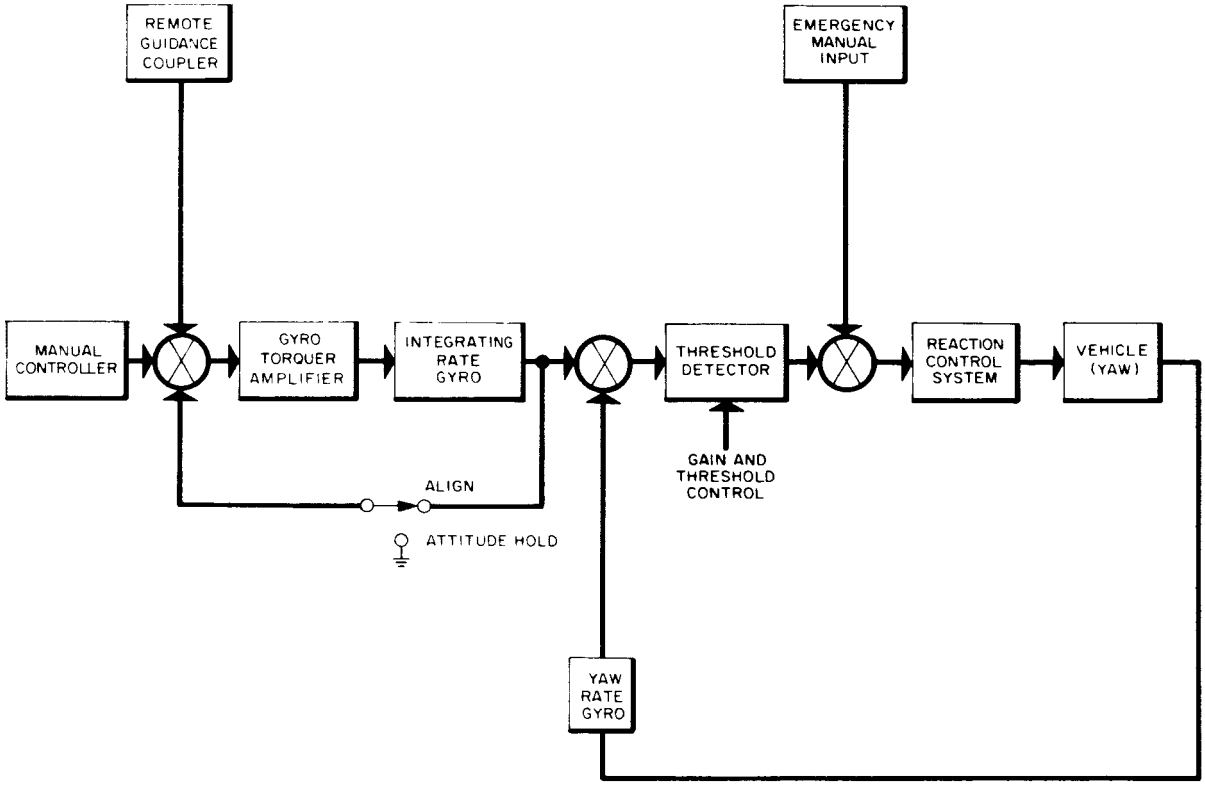


Figure 3-13. Stabilization and Control Yaw Axis, Block Diagram

accelerations. In addition, inertial attitude angles and velocities must be measured in the vehicle and transmitted to the OLF so that the proper attitude and lateral velocity commands may be computed.

One axis of the recommended attitude control system configuration for the unmanned payload is shown in figure 3-14. Both attitude and lateral vernier velocity control is provided, but the velocity control is used only for docking. The attitude control loop will produce an angular rate proportional to the input signal telemetered from the OLF, and will hold the existing attitude in the absence of an input. The lateral velocity control loop will produce a lateral velocity proportional to the input from the OLF, and will hold essentially zero velocity in the absence of an input. An inertial reference unit is used to measure the inertial attitude, and velocity of the vehicle for transmission to the OLF. A transponder is used to aid radar tracking of the payload from the OLF.

The equipment required in the OLF to derive commands for rendezvous and docking of the payload, is shown in figure 3-15. The selection of a specific control method will be based on how well the human operator can perform the rendezvous and docking maneuvers in terms of time, extra fuel, and the amount of automatic equipment replaced by him in the OLF.

If manual control is used, the operator in the OLF will have a display of the payload's position, orientation, and angular rates with respect to both the OLF and inertial space. Using these displays, he will determine the proper attitude rate and thrust on-off commands to accomplish a rendezvous.

If automatic control is used, the human operator will be replaced by a computer operating on essentially the same input information.

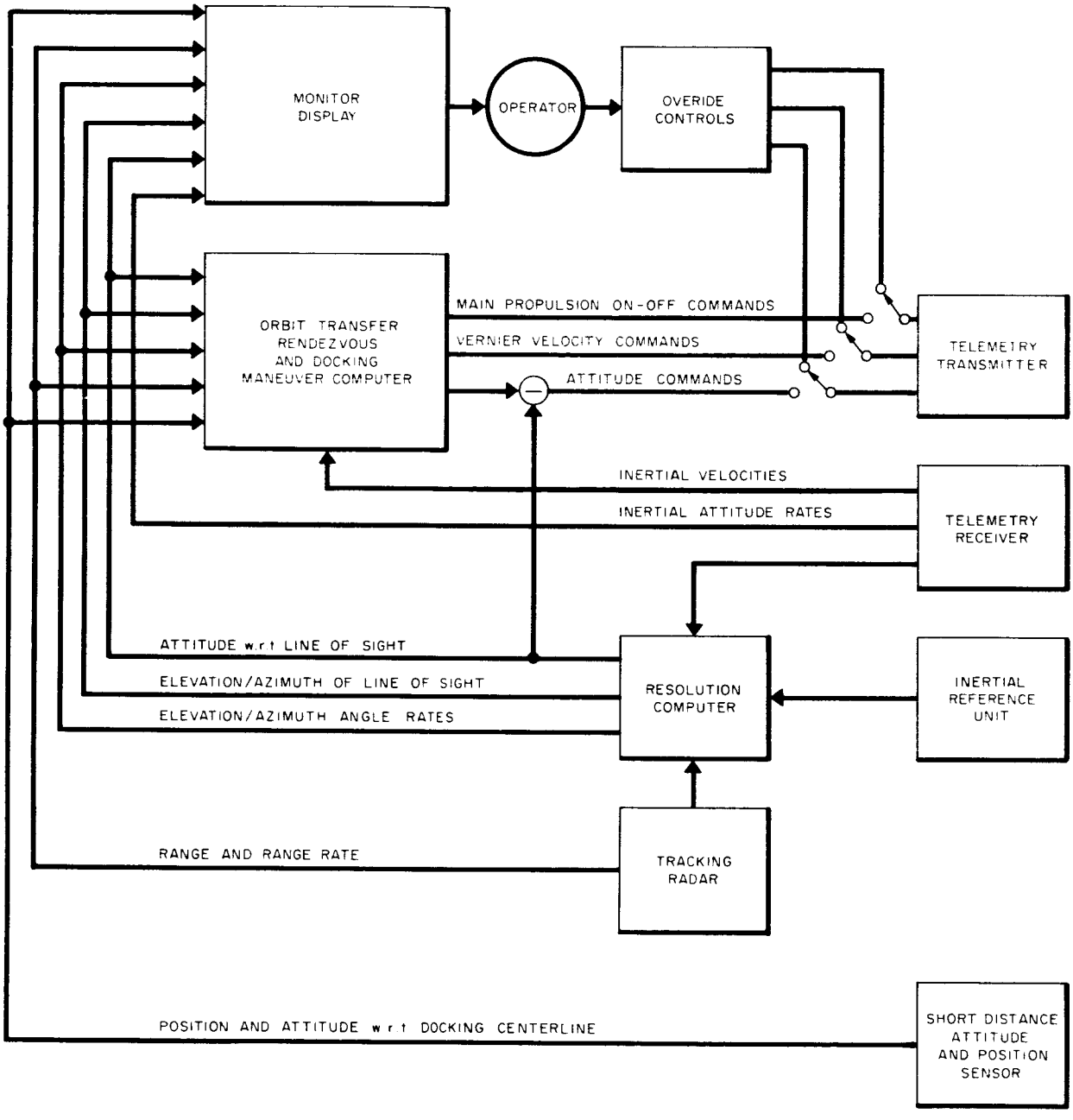


Figure 3-15. OLF Equipment for Automatic Control of Unmanned Payloads

(3) Space Vehicle (No Tug)

In the absence of tug assistance, the space vehicle must be taxied to a launch site some distance from the OLF and oriented for launch. This involves orientation of the vehicle to thrust away from the OLF, coast to the launch site, reorientation for braking at the launch site, and orientation for launch. The latter maneuver may be controlled from the OLF in the same manner as the unmanned payload in order to take advantage of its centralized sensing and computing facilities. The control system requirements will be similar to those of the unmanned payload.

The torque generators for the space vehicle will operate on hydrogen from the main propulsion tanks. This is the same system as is recommended for the astrotug. The system is described in the following paragraph (4) and shown in figure 3-16.

(4) Manned Astrotug

The astrotug's functions include rendezvous and docking with the incoming payload, followed by rendezvous and docking at the OLF with the payload in tow. Control of tug attitude will be required for rendezvous, while control of both attitude and lateral velocity will be required for docking. During payload return, the payload attitude control system may or may not be coupled to the tug control system. The system configurations discussed cover both possibilities.

One axis of the recommended system configuration for the manned astrotug is shown in figure 3-17. The system provides control of attitude rate and lateral acceleration. The attitude control loop will produce a vehicle rate proportional

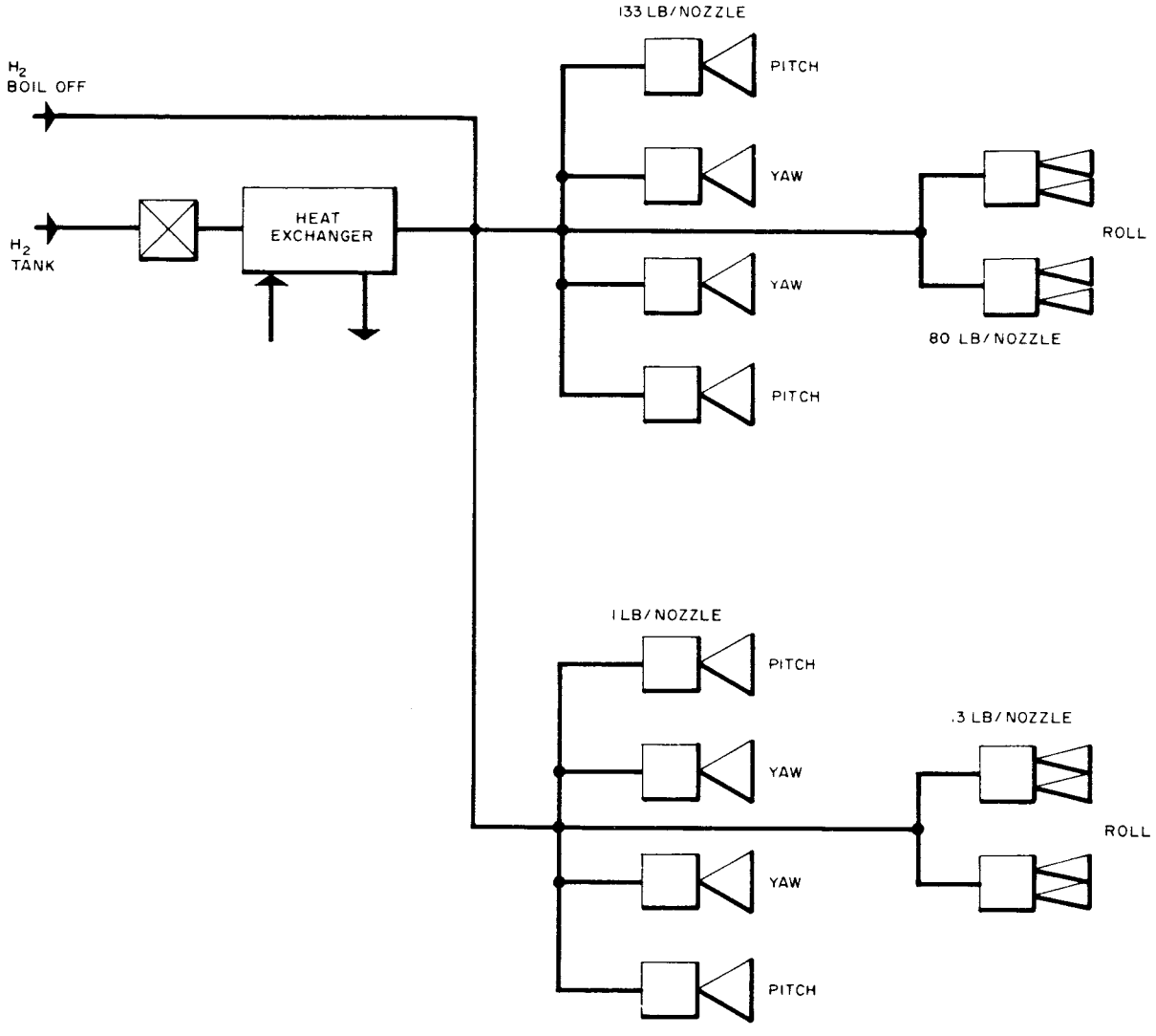


Figure 3-16. Space Vehicle Schematic

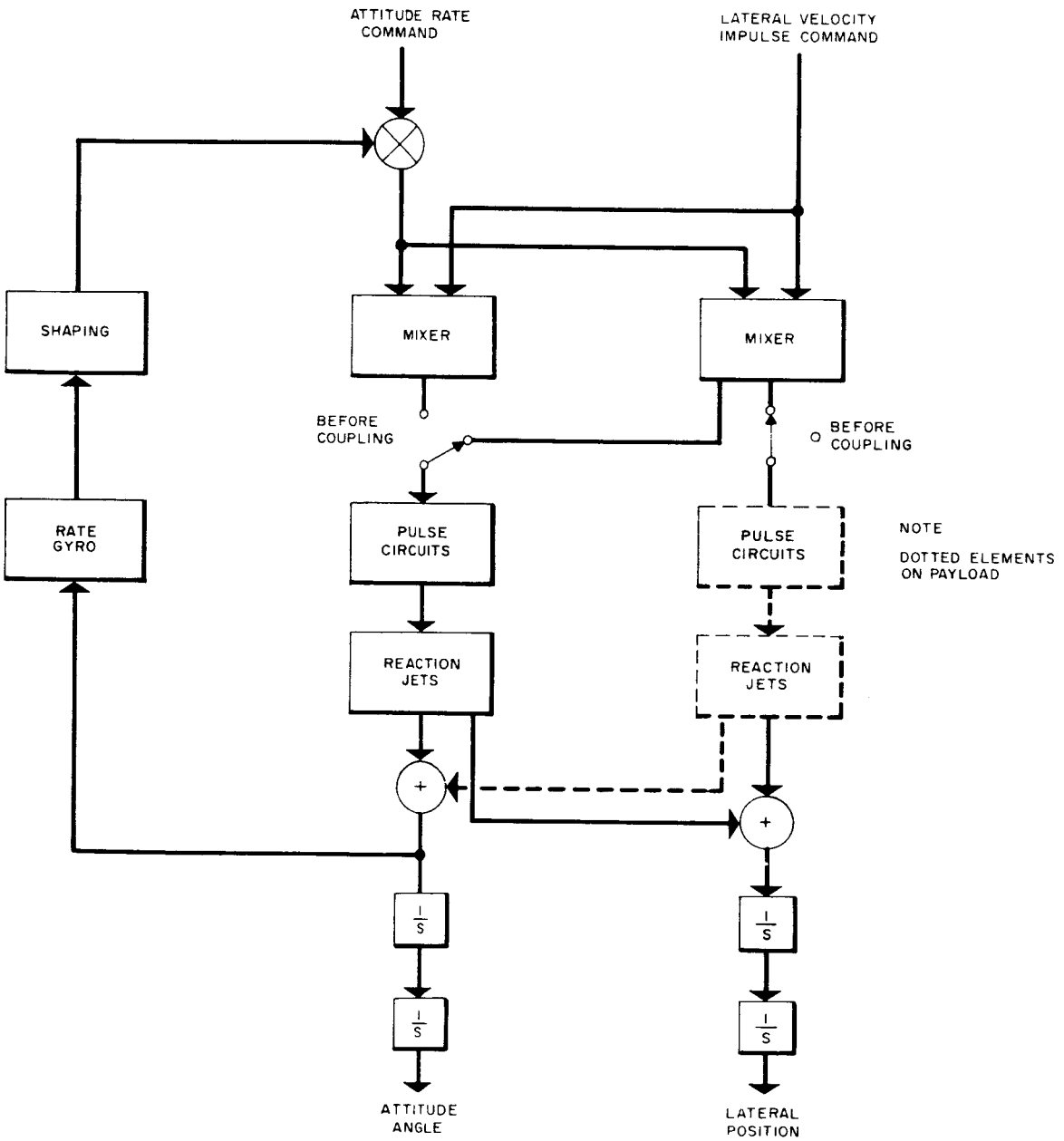


Figure 3-17. Minimum Attitude-Control System for Manned Astrotug

to the pilots input command and will provide short period attitude hold in the absence of an input. Optical references will be used in docking on the payload and OLF.

An agumented tug control system is shown in figure 3-18 which will provide pilot relief as well as more precise control by using additional equipment. Both attitude angle and lateral velocity control loops are provided.

(5) Unmanned Payload (With Tug)

When tug assistance is available, the unmanned payload control system is required to damp the initial tumbling and upon command from the tug, hold its attitude while the tug attaches. The control system may also be required to accept thrust commands from the tug in the event that the payload has to provide all or part of the attitude and/or propulsion forces required for the return trip and docking..

The recommended control system configuration for the payload is shown in figure 3-19. The torque generators and the associated control circuitry is similar to that described for the other vehicles. Integrating rate gyros are used as the attitude reference.

(6) Space Vehicle

The recommended system for the OLV, when a tug is available, is the same as that used for the unmanned payload.

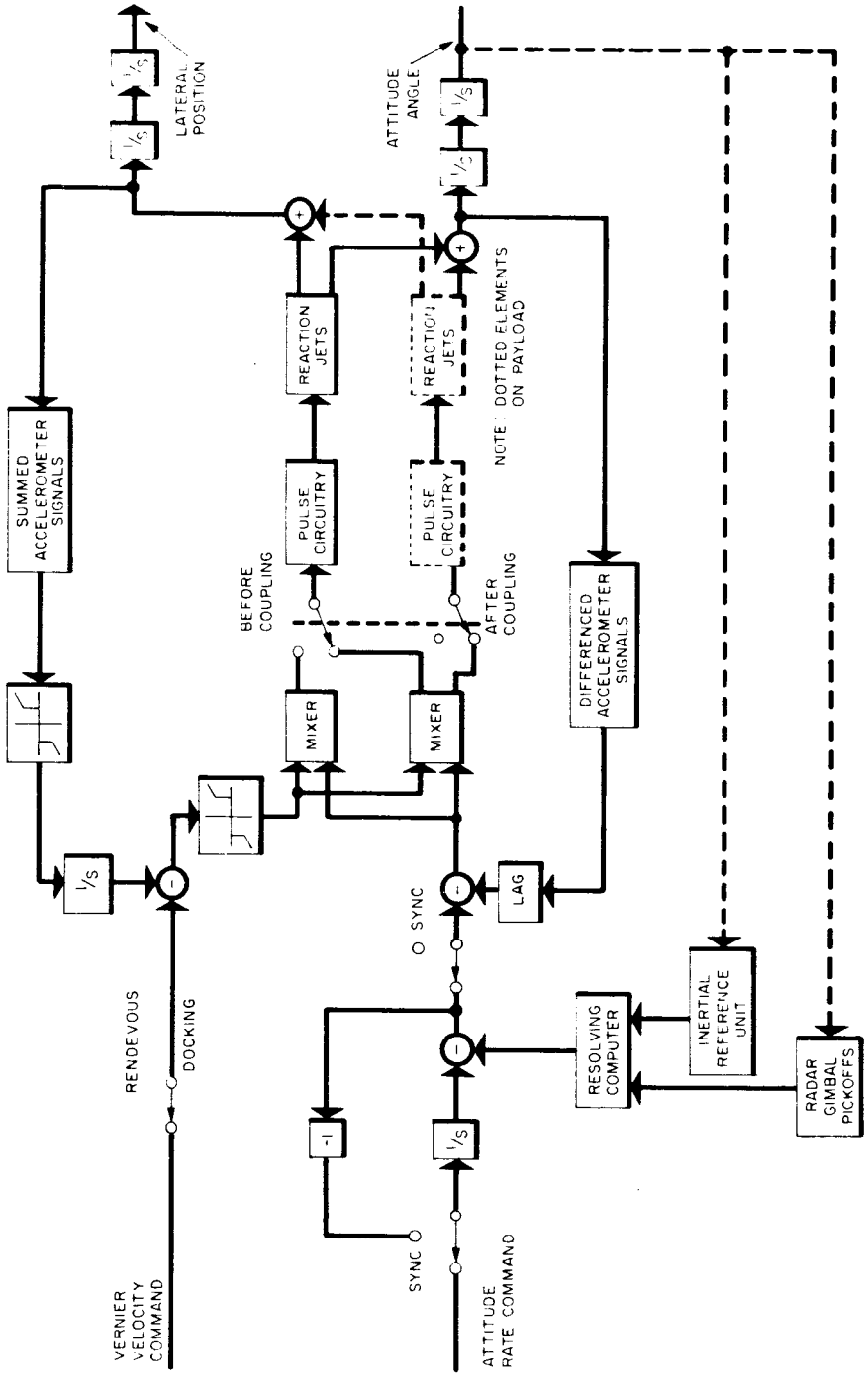


Figure 3-18. Augmented Attitude and Vernier-Velocity Control System for Manned Astrotug

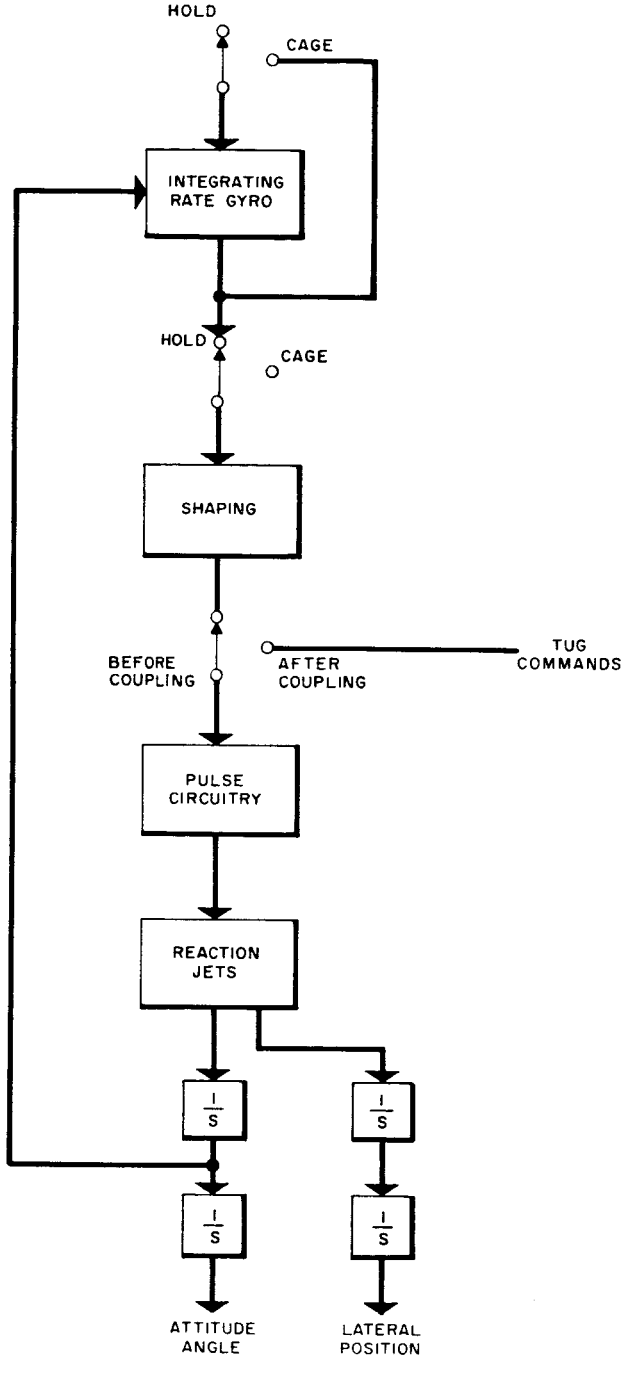


Figure 3-19. Unmanned Payload Attitude-Control System (With Astrotug)

3-4. COMPUTERS

a. Operational Model

The operational model for determining computer requirement is described in figure 3-20 with the primary elements being the OLO vehicles, the earth based central operational control (COC), and the command and control supporting links.

The model was conceived and restrained so that there was a steady evolutionary progression from the initial phases of operation to ultimate capability. The motivation of this evolution is a vehicle concept which is completely independent of earth-based control. To begin with, the ground tracking net may actually perform certain of the guidance and navigation functions necessary within the system and may even exercise command and control over certain modes of the operation. As further testing and utilization proceed, it is expected that this tracking net will, at most, be utilized as a back-up system. More specifically this means that the spaceborne computers aboard the vehicles shall not be interdependent, one upon the other, and that each shall be capable of satisfying the guidance, navigation, station keeping, energy management and communications requirements for its respective type of vehicle.

On the other hand, the ground based computers should have the capability of performing all of these functions for all of the vehicles, as well as performing the normal tracking enhancement functions for each of the tracking stations, and the additional navigation and stabilization requirements of any recommended ship stations. The study was separated into the requirements for, space-borne computers and, earth-based computers.

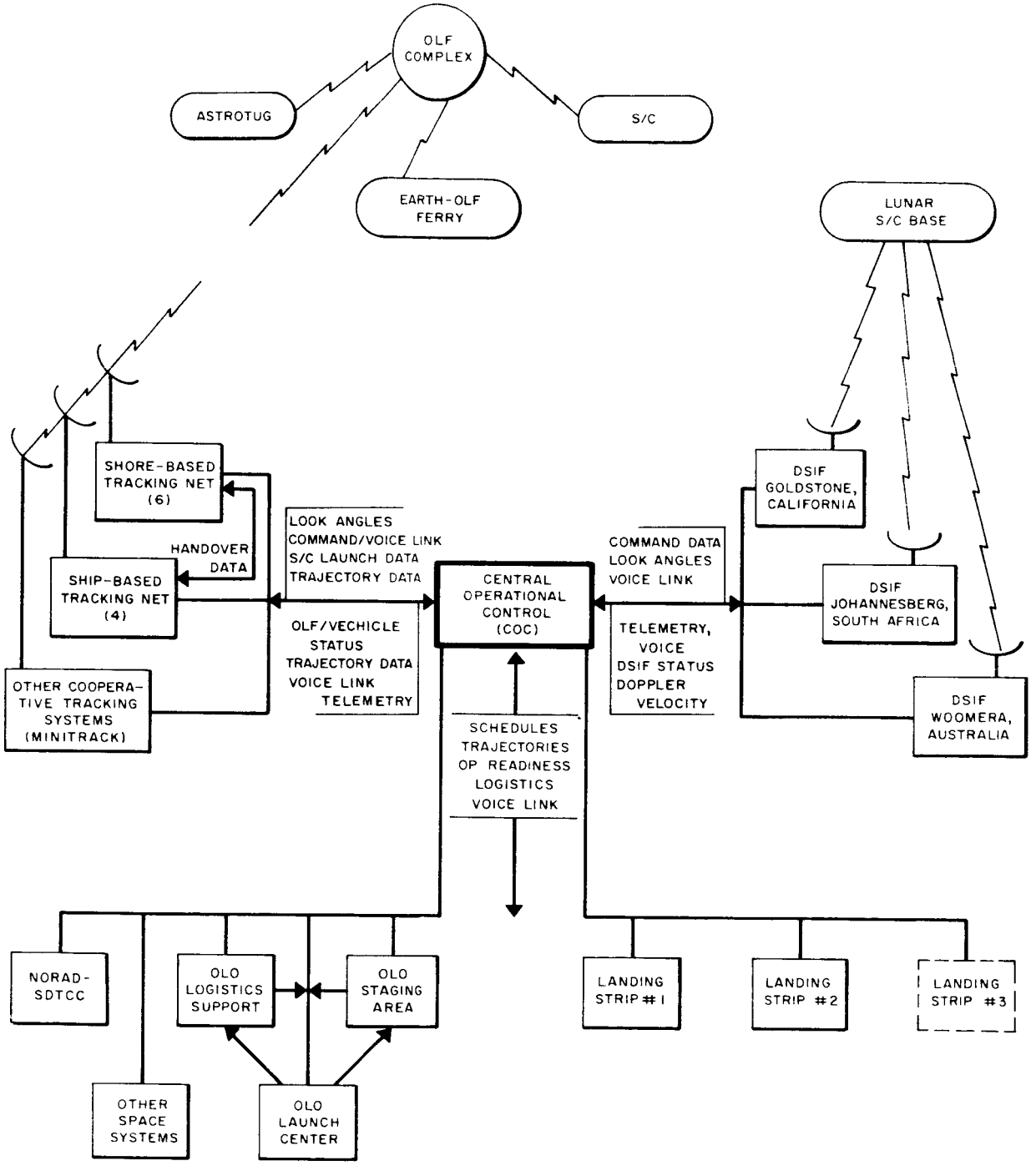


Figure 3-20. Operational Model

b. Space-borne Computer

(1) Requirements

It is assumed that the final configuration of the OLF will incorporate the simplest versions of instrumentation. In most cases this instrumentation although accurate, will be an unsophisticated manually operated mechanical and optical device. It is anticipated, however, that there will be a necessity for considerable computational capability, particularly for data enhancement for increased reaction time, and as an aid in presentation of data especially when "quickening" and higher order inputs are desirable. A first list of the primary computer requirements for the OLF is given as being:

- Guidance
- Navigation
- Own vehicle control data enhancement
- Communication (data) control
- Launch facility control
- OLV targeting
- Checkout
- Simulation
- Countdown.

These areas were considered so that the system characteristics of a digital data processing computer aboard the OLF could be determined.

(2) Selected Space-borne Computer System

Based on the analyses performed of the space-based operations, a possible system configuration for an OLF computer system was determined. Figure 3-21 shows the overall relationships between OLF and the computer system.

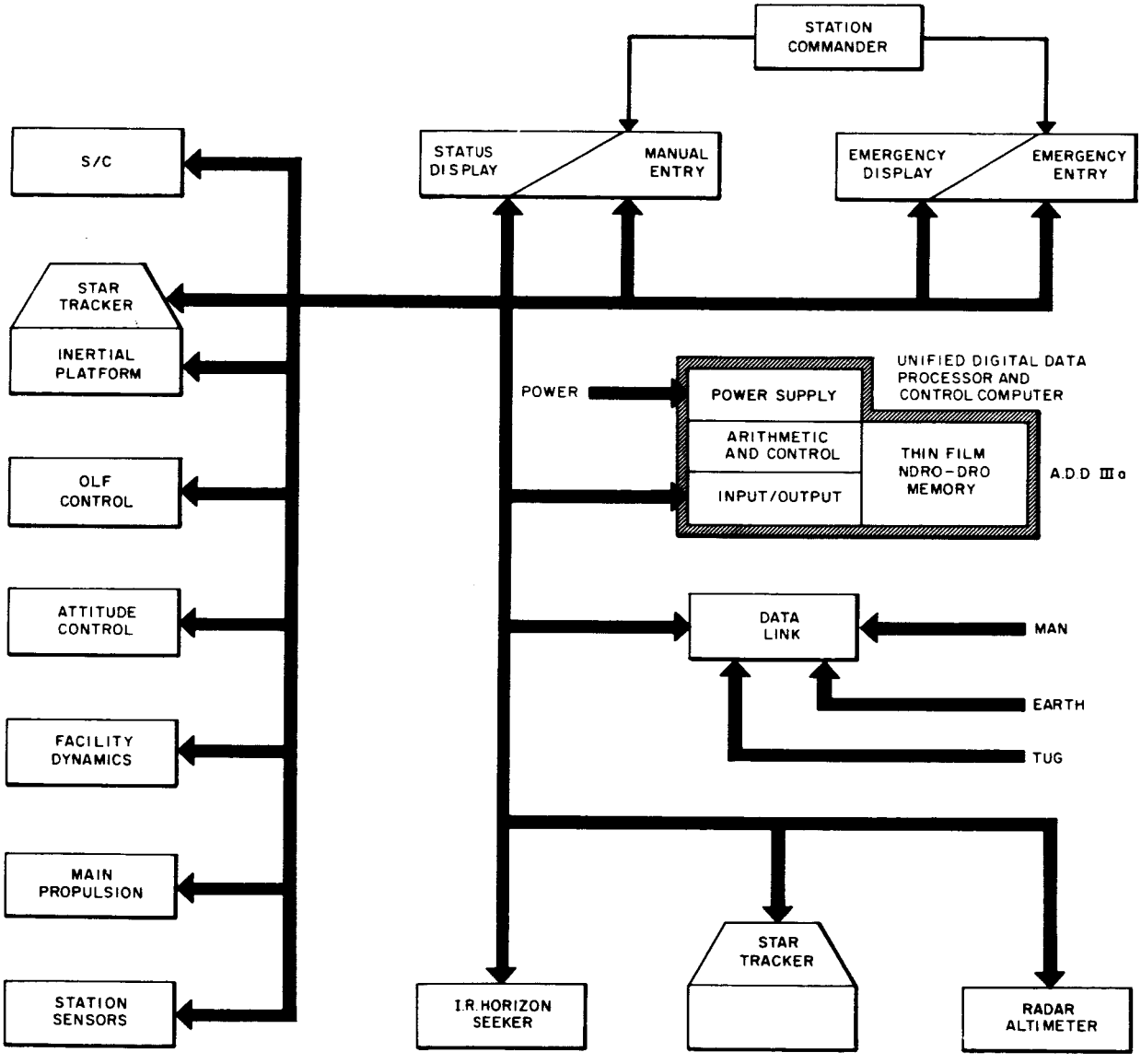


Figure 3-21. Initial OLF Data Processor and Control Computer

The memory capacity required of the on-board computer in the minimal OLF system as determined by this study, is estimated to be 8,000 24 bit words - 1000 words of variable (DRO - Destructive Read-Out) memory and 7,000 words of electrically alterable, perminent (NDRO - Non destructive read-out) program memory. A relatively small amount of memory is required for the program storage and computation of the equations of motion, the orbit determination, and orbit corrections as shown in table 3-5. The time required to perform these computations on a typical computer* is also given in table 3-5. With the exception of the in-plane orbit computation, exact square roots are assumed rather than a simple Newton-Raphson approximation.

TABLE 3-5

ESTIMATES OF ORBITAL PROGRAM COMPUTATIONS

	Program Memory (NDRO)	Variable Memory (DRO)	Time (milisec)
1. Equations of motion	100	27	11.9
2. Orbit determination	200	88	20.0
3. Orbit correction	110	15	13.5
Total	410	130	45.4
Adjusted Total	820	143	56.8

The memory requirements for the processing and storage of the sensor data and the control of the OLF's attitude orientation is several times that required for the orbit computations. Another 1000 words of permanent memory and several hundred words of variable memory would be required for the

*UNIVAC ADD-2a

earth data link communications, the system displays, and the orbit determination of a rendezvous vehicle.

(3) Computer Organization

The organization of the initial OLF Computer is shown in figure 3-21. This is the block diagram of the probable sub-units and the computer functional units. The central computer organization is shown in figure 3-22. The number of phases which are executed to perform an instruction was increased to give added flexibility. The word organization gives a very versatile and powerful instruction repertoire of over 960 useful instructions.

The computer design allows for direct access to the memory to effect input and output. This eliminates the need to buffer many external sensors since they can be accessed directly in and out of storage in real time with control by external request or by a program initiated step.

A study was made of the short term goals of computer development and the longer range expected improvements as related to the typical increase in complexity of systems as the problem and system size expand. The results of this study are presented in table 3-6.

TABLE 3-6

COMPUTER SUBSYSTEM GROWTH

	Weight	EOR*	1/O Channel	1,000's words of memory
Initial OLF	17	71	44	9K
Intermediate OLF	8.5	88	57	25K
Final OLF	4.5	121	88	47K

*Execution Operation Rate 100's Operations Per Second.

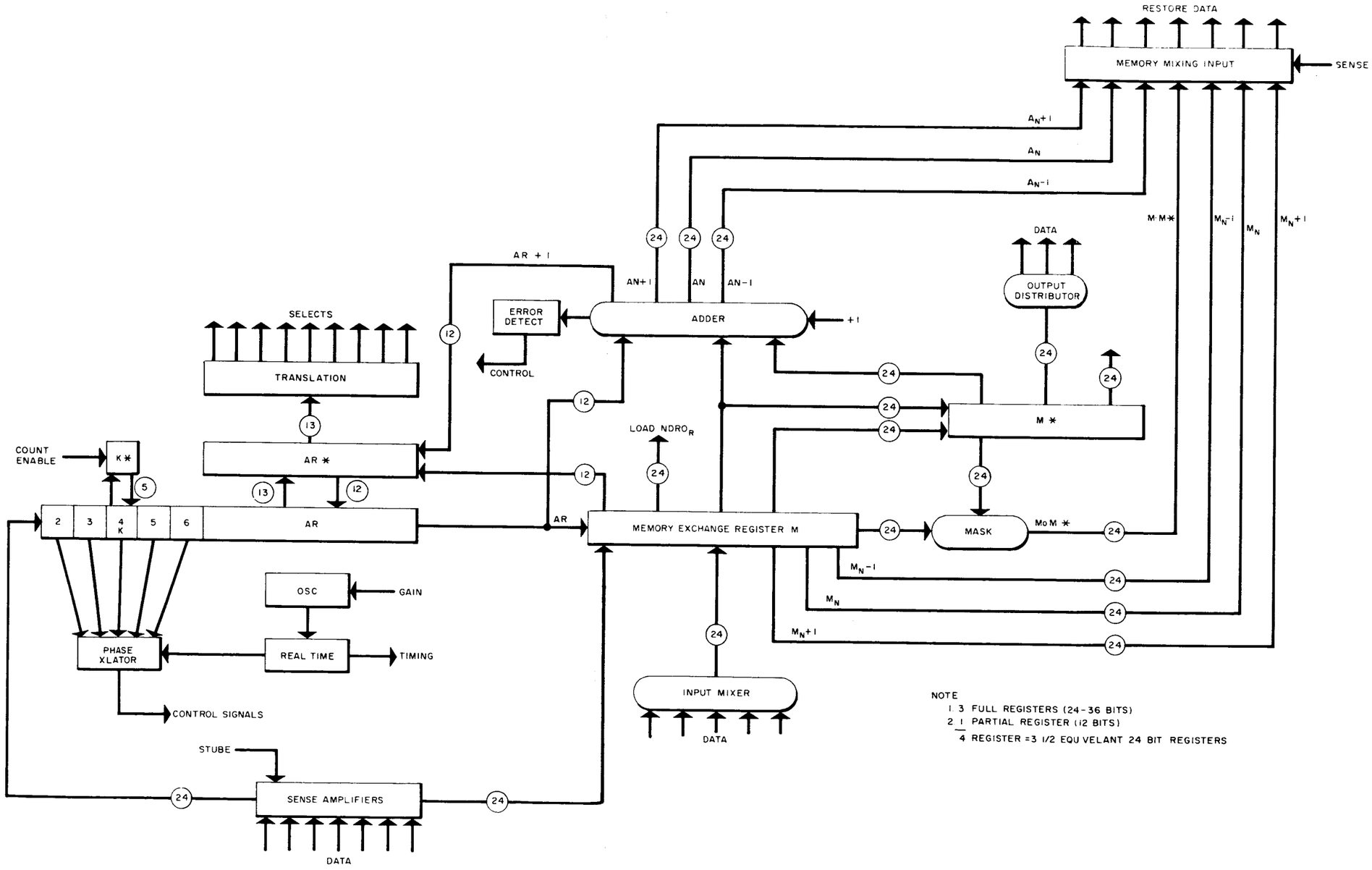
c. Earth-based Operations

(1) Requirements

The evolutionary nature of the OLO program required that the initial phases of navigation and guidance be supported by an extensive earth-based tracking and control complex. Furthermore, the continuing operational requirements for human safety and mission success indicated that earth-supported navigation and guidance of at least the OLF flight path, is a primary system function. Finally, it is anticipated that this widely dispersed earth-based complex of tracking stations, launch pads, landing strips, ship nets, supply depots, and assembly areas, must be under direct command and control of a centralized operational authority.

Investigations of the earth-based operations of the OLO system were confined to the tracking complex and its supporting role to the Central Operational Control and to support of the OLF navigation-guidance functions. The operational concept of centralized supervisory control (at the COC) is the basic premise of this ultimate design. The tracking nets have tracking responsibility of all traffic to/from the OLF for all phases of the OLO program. Initially this responsibility is primary in that navigation-guidance would depend on inputs from this net. As OLO evolves into a more self-sufficient guidance, this primary responsibility would be reduced to a back-up capability for emergency situations.

The remote tracking stations must be capable of a broad spectrum of functions. This requirement will diminish as more and more confidence is acquired in the capability of each of the vehicles to operate completely on their own.



NOTE
1. 3 FULL REGISTERS (24-36 BITS)
2. 1 PARTIAL REGISTER (12 BITS)
4 REGISTER =3 1/2 EQUIVELANT 24 BIT REGISTERS

Figure 3-22. Central Computer Organization

An approximation to the functional equipment for a tracking station computer subsystem (enclosed by dotted lines) and the connected subsystems is presented in figure 3-23. With the exception of the communications and telemetry subsystems, the configuration shown is similar in concept to that developed by Univac for AMR Stations 12, 13, and the MARS ships as shown in figure 3-24.

(2) Selected Ground Computer System

Experience with computer subsystems for tracking stations indicates that a number of major computers are capable of performing the necessary functions in real time. Among these are the IBM 7090, Philco 2000 (several models), the CDC 1604, RCA 601, and Univac 1206, 490 and 1107. Inasmuch as the Univac 1206 has been proven suitable for shipboard use, this machine was used as the typical ground-based computer for use in the OLO system. For convenience, the salient characteristics of this machine are presented.

The Univac 1206 is identical with the CP-652A/USQ-20 computer developed and now in production for the Naval Tactical Data System. It is a ruggedized (MIL-E-16400) machine designed for rapid processing of large quantities of complex data. This design emphasizes random-access storage and ease of communication with a wide variety of connected subsystems. Major features include:

a) Internal high-speed storage and a capacity of 32,768, 30-bit words.

b) Repertoire of 62 instructions.

c) Average instruction execution time of 13 microseconds (including memory access time).

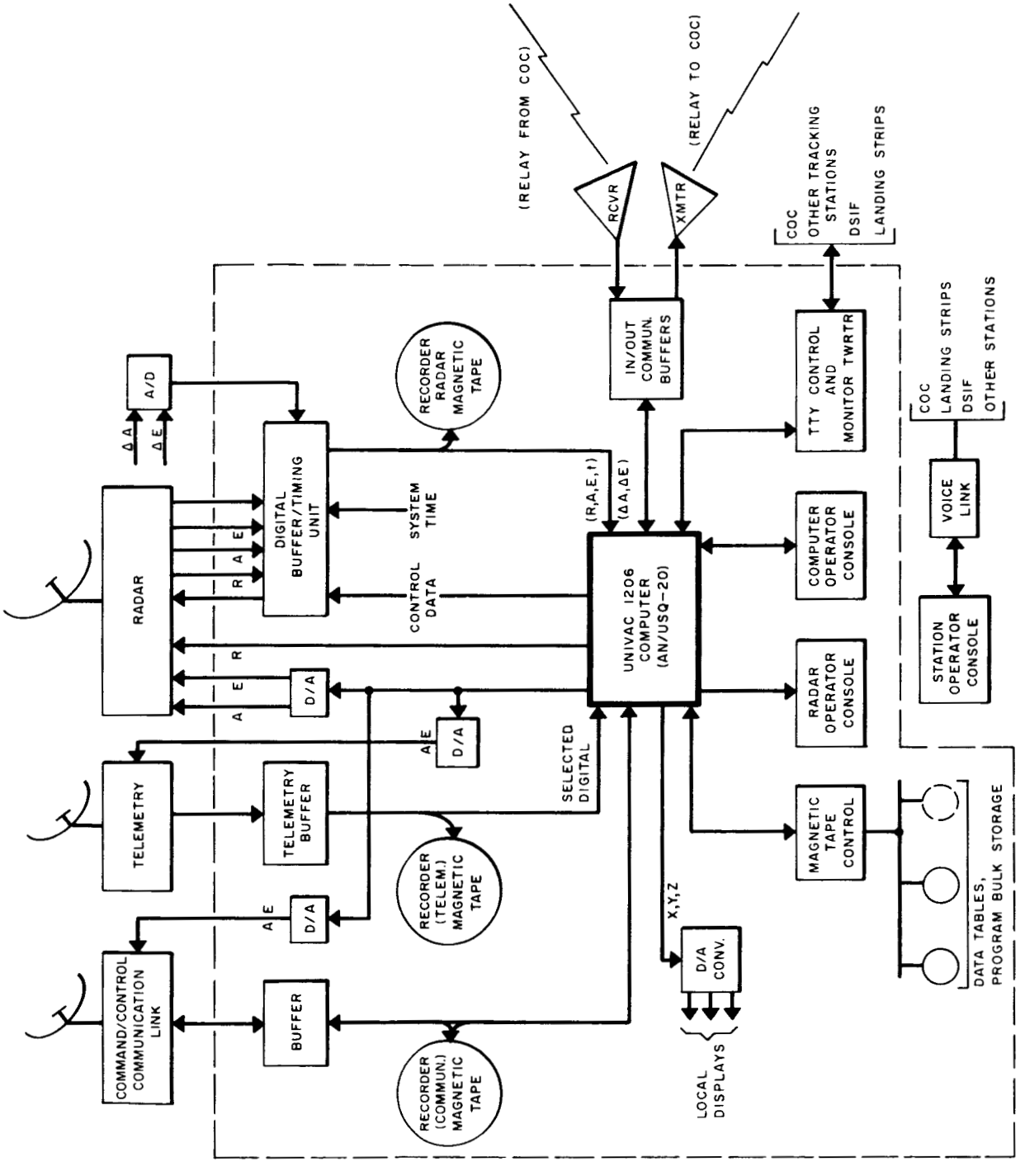


Figure 3-23. Earth-Based Tracking Station Functional Diagram

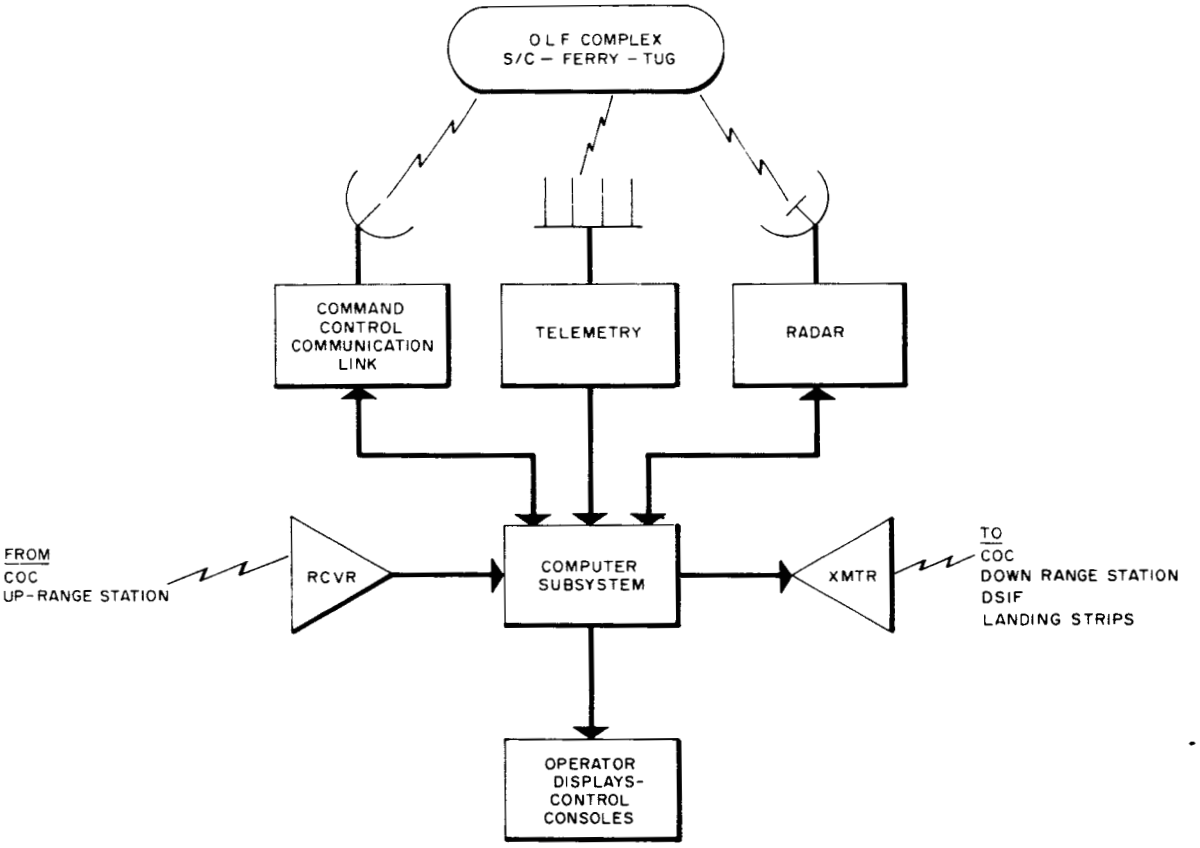


Figure 3-24. Tracking Station Block Diagram

d) Optional operation with 15-bit half-words.

e) Single-address instructions.

f) Internal seven-day real-time clock.

g) 12 input and 12 output channels.

h) 16-word wired auxillary memory.

i) Parallel one's compliment, subtractive arithmetic.

Some of the salient physical characteristics of the Univac 1206 are as follows:

a) Dimensions: 33" deep x 37" wide x 65" high.

b) Primary power supply: 60 cycle input, 400 cycle output motor alternator.

c) Total power consumption: 2400 watts.

d) Forced-air cooling; heat exchanger optional.

3-5. CHECKOUT EQUIPMENT

a. Operational Model

The operational model considered represents a nearly ideal relationship between man and machine. The function of command should be reserved for the human operator. Man's abilities to reason and exercise judgement should be exploited. Routine decisions, data processing, and related tasks which must be accomplished in great quantity and with extreme speed and accuracy should be relegated to processes under control of a computer.

b. Operations

(1) Checkout

The checkout philosophy adopted is that an assembled OLV in orbit is presumed to be operable. Checkout will then consist simply of demonstrating that the OLV is indeed operable and capable of performing certain required tasks. After cable connection has been made, a confidence level checkout program will be initiated. Demonstration that the spacecraft is operable will be made at the subsystem level and will proceed in the following order:

- Power
- Communications
- Central computer
- Navigational aids (radar, stellar tracker, stable platform, etc.)
- Autopilot (including attitude and thrust control and the propulsion system).

(2) Countdown and Launch

The OLO program must be an integrated program with each phase leading smoothly into the succeeding phase. In the MALLAR mission, countdown must be controlled by the spacecraft for the following maneuvers: midcourse guidance, deboost to lunar orbit, lunar landing, and lunar launch. It is reasonable to assume that countdown for launch from earth orbit will also be controlled by the spacecraft. Countdown and launch under spacecraft control will constitute a final check on the performance of the spacecraft and its crew. During countdown, all

parameters required for prediction of the launch maneuver will be self-derived by the spacecraft. The derivation of these parameters will provide a check on all navigational aids and on the computer and its program. Verification of parameter derivation will be accomplished by the OLF where a redundant parameter derivation is made. Actual launch of the OLV from earth orbit is expected to take place at some distance from the OLF, contact being maintained between spacecraft and OLF by means of an RF link.

Following a successful checkout and the completion of certain required manual operations, the spacecraft crew will enter the spacecraft and the spacecraft will be separated from the OLF. The countdown will then proceed under control of the spacecraft. All spacecraft countdown steps will be monitored by the OLF. A hold or abort should be callable by the OLF to within tens of milliseconds of launch.

Countdown for midcourse guidance, lunar orbit deboost, lunar landing and lunar launch must proceed under complete control of the spacecraft. Backup for these areas must be supplied from earth via the DSIF.

(3) Troubleshooting

The area which will make the greatest demands upon the checkout equipment is that of troubleshooting. Once trouble has been identified as the failure of a major subassembly to function properly, the trouble must be traced to a replaceable module. This troubleshooting is accomplished by an OLF crew member with sophisticated aids and in conjunction with the OLF checkout equipment.

During troubleshooting, the positions of dials, switches, and knobs of the PTS should be automatically

monitored aboard the OLF. When desired, this data can be relayed to the ground command center, which can then monitor and guide the troubleshooting operation.

c. Description of Recommended System

A semi-automatic checkout was selected for use in the OLO program. The system consists of the ADPAC Computer, (Adaptable Digital Process and Control) computer controlled peripheral equipment, and a portable test set complete with probes and cabling. All test programs and subroutines required for checkout, troubleshooting, and countdown are available within ADPAC. The peripheral equipment is an array of computer selectable signal and function generators which are available to serve as stimuli in system tests and to assist in the measurement of such characteristics as power, frequency, voltage, and impedance. The portable test set is the primary means employed to reduce the number and length of test leads, reduce required automatic switch matrices, and reduce test equipment loading on sensitive circuits in the system under test.

(1) Central Computer (Figure 3-25)

OLF testing necessitates flexible and versatile test equipment. The selection of a programmable, digital computer as the OLF checkout and countdown controller not only provides the adaptability desired of a space-borne testing system, but also provides greater accuracy in testing by basing decisions concerning faulty units upon a sequence or series of tests rather than upon a single test. The computer can store the results of several tests and draw logical conclusions from these results. Contained within this central computer is a

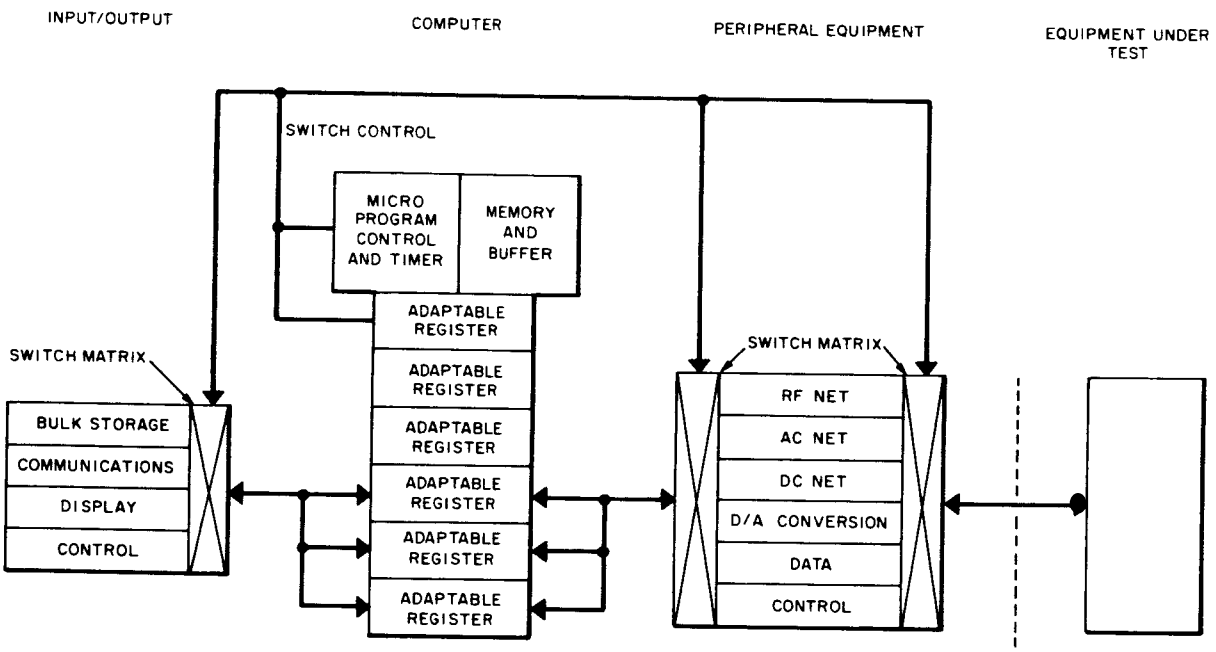


Figure 3-25. ADPAC Block Diagram

collection of automatic test programs and subroutines which are employed during the checkout, troubleshooting, and countdown operations. Additional test programs and subroutines can be added as required to support system growth and mission expansions.

A microprogrammable computer requires five first rank registers to perform the logical operations. Only very simple and extremely few logic control circuits are required outside the registers themselves, which results in extremely fault diagnosis techniques to isolate failures within the computer. Because the computer must control a significant amount of peripheral equipment, the logic and registers which control that external equipment have been integrated directly into the computer logic. These integrations have added more universal registers and have thereby contributed to the selfhealing capability.

The proposed computer has the following characteristics:

Weight (including random access memory)	20 pounds
Average power required (when calculating)	90 watts
Average power required (when controlling)	40 watts
Add time	6 microseconds
Volume (including random access memory)	0.70 cubic feet
Multiply (microprogrammed)	260 microseconds

Standard routines such as sin x, cos x, e^x, etc., can be either programmed or microprogrammed, depending upon the time which may be allotted for their generation.

The computer is programmable as a standard single address, single command, sequentially addressed machine, although the operations of the commands themselves may be varied by reprogramming the microcommands by means of internal routines.

The estimated 100,000 words of test programs dictate the need for a large bulk storage capability. For this purpose, a tape storage unit has been selected. To provide rapid access buffer storage compatible with the high speed computer operation, a 1,000 word parallel readout, random access memory is provided.

(2) Peripheral Equipment

Peripheral equipment for the recommended system will consist of the following: a portable test set, an array of signal and stimuli generators, a switching matrix, a display-control panel, an OLV-OLF digital data transfer unit, and a communications tie-in unit.

(3) Portable Test Set

The portable test set (PTS) will be the primary means of communication between man and machine during maintenance. The PTS is shown in figure 3-26. It is a single unit which can be strapped to the shoulders and belt of a crewman or can be fixed to the OLV.

The microfilm viewer provides ready access to OLV prints, drawings, and test plans. Referrals to specific documents can be part of an ADPAC subroutine for any given test.

The digital data and control section consists of a buffer register, a display light panel, a manual digital input device, a mode select, and a test-initiate control.

Example of uses of the PTS are shown in figures 3-27, 3-28, and 3-29.

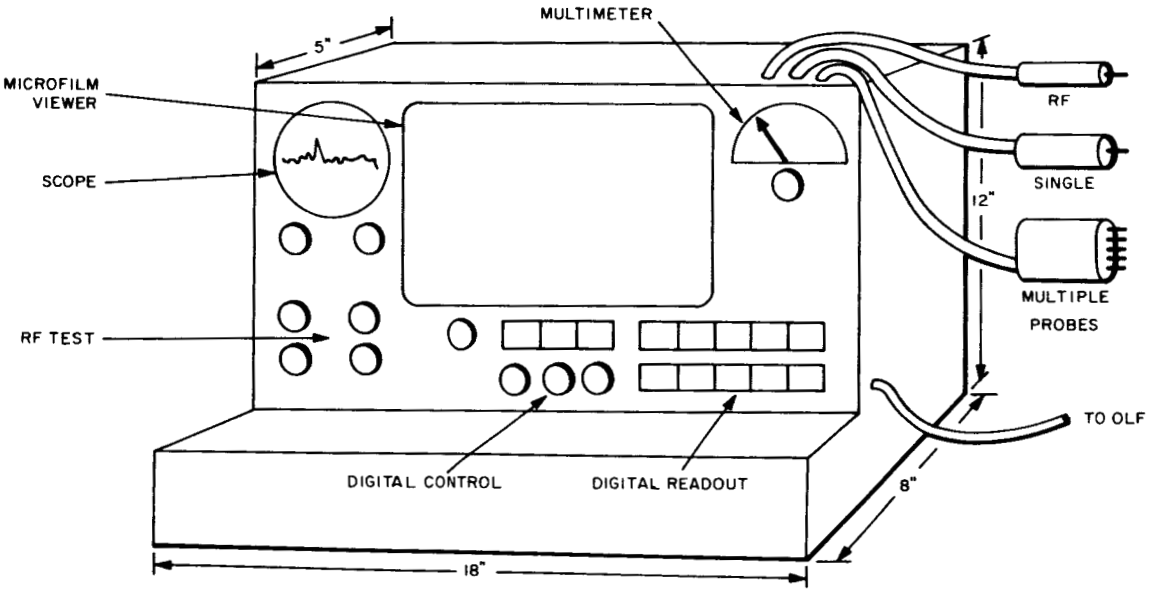


Figure 3-26. Portable Test Set

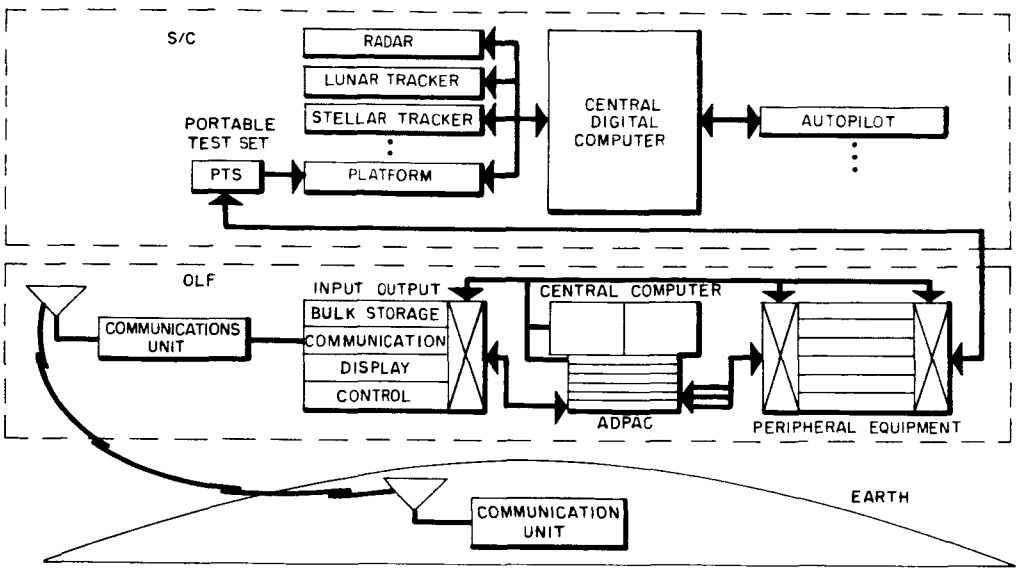


Figure 3-27. Troubleshooting With Portable Test Set

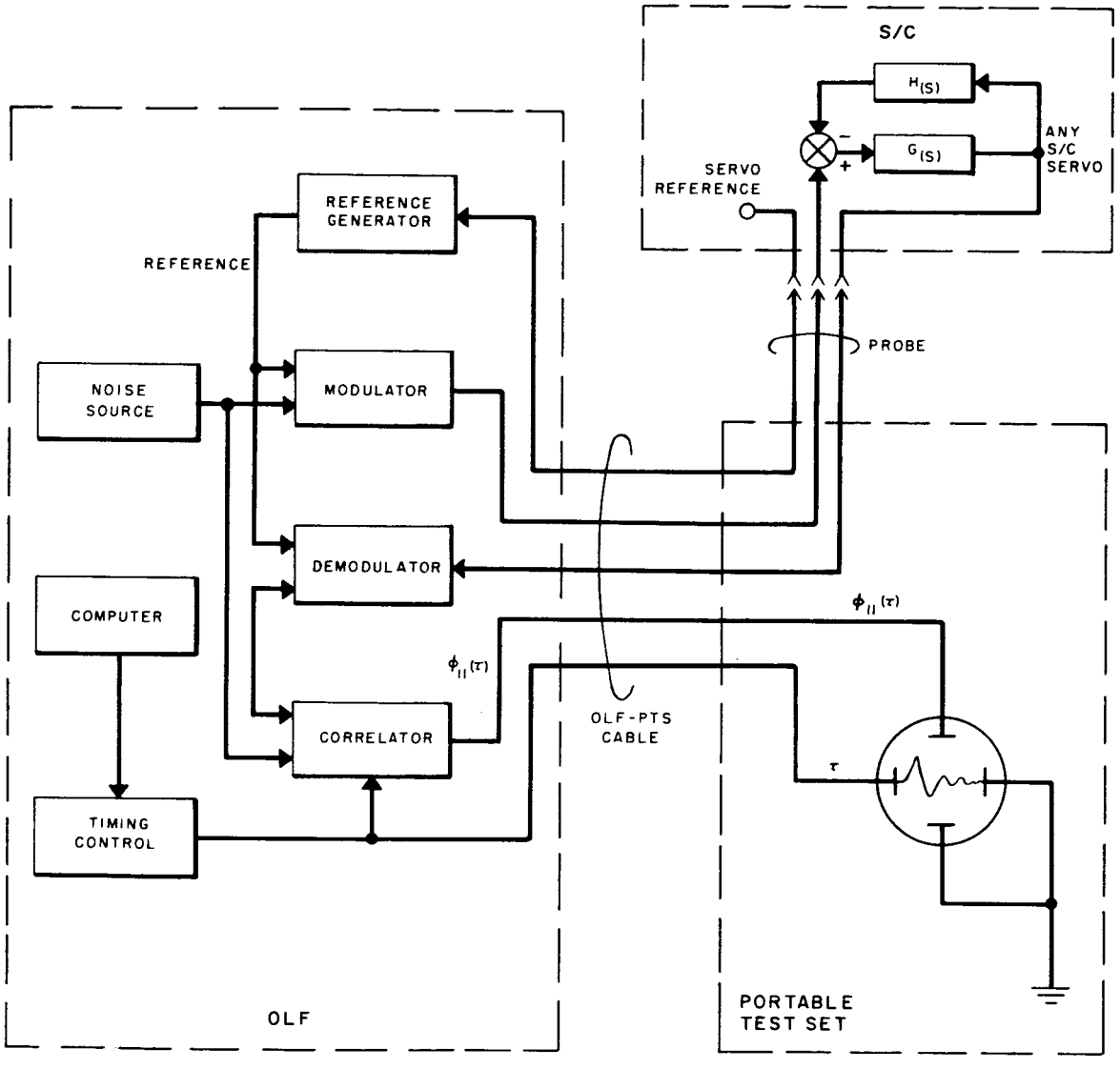


Figure 3-28. Transfer Function Measurements

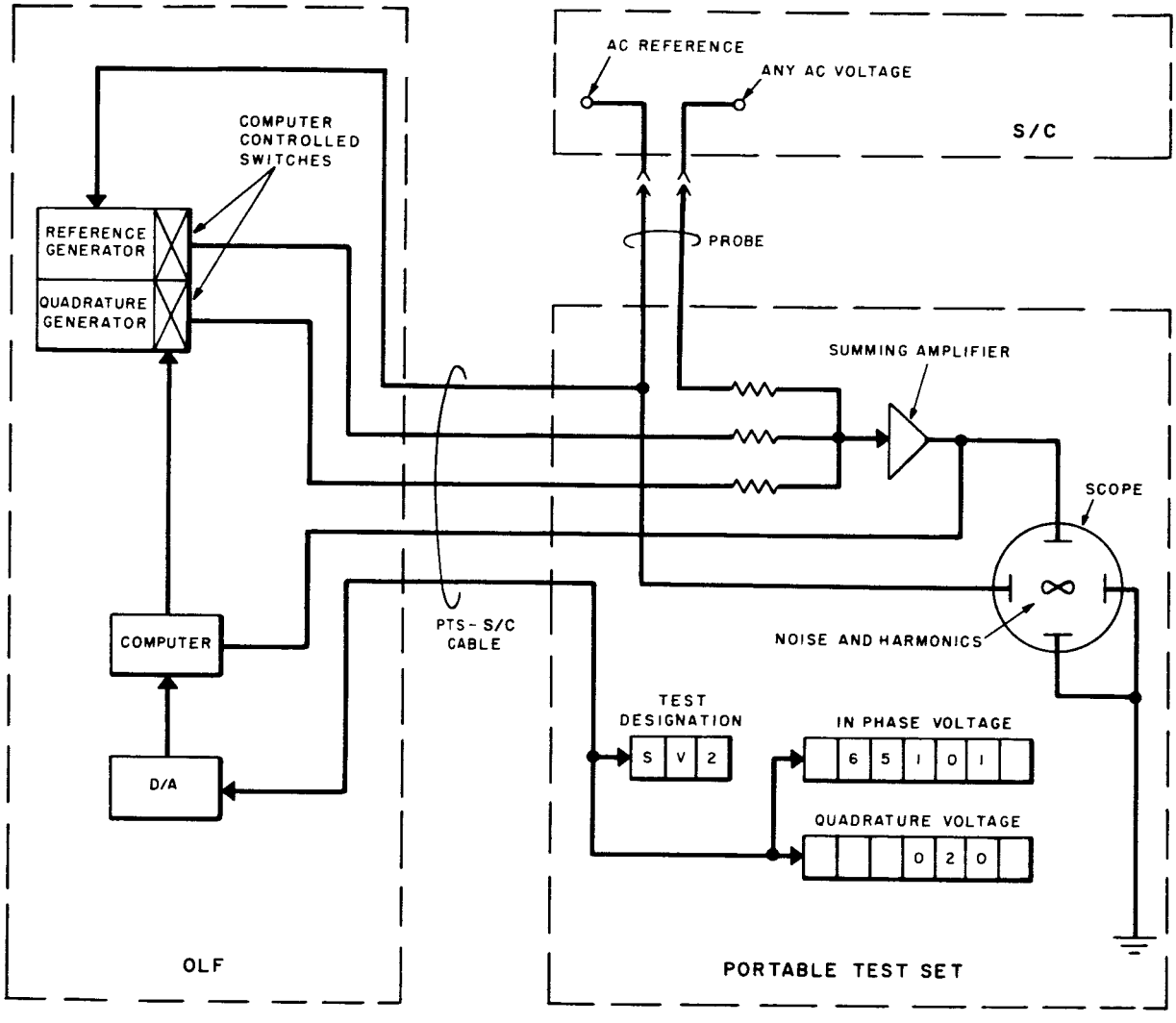


Figure 3-29. Servo Voltage Measurement

3-6. SECONDARY POWER

a. Vehicle Power Requirements

Table 3-7 is a breakout of the vehicle power requirements for the guidance and control and overall electrical power requirements.

TABLE 3-7

VEHICLES AND POWER SCHEDULE

Vehicle	Power System Schedule Year	Power Required
OLF	1968	30 kw
	1970 - 1975	60 kw
Spacecraft	1964 - 1966	1.5 kw
	1966 - 1969	2.5 kw
	1969 - 1975	3.5 kw
	3 Man OLV	
	Guidance	2.3 kw/20 min
	Misc. power	1.7 kw/120 hr
	Lunar rest	1.4 kw/x days
	Final phase	1.6 kw/1 hr
	6 Man OLV	
	Launch	3.2 kw
	Midcourse	2.8 kw
	Lunar land	3.2 kw
	Lunar rest	2.7 kw
	Lunar launch	3.2 kw
	Midcourse	2.8 kw
	Earth guidance	2.5 kw
ASTRO Tug	1968 - 1975	1.80 kw
	Guidance and Control	0.45 kw
	Lighting	0.10 kw

TABLE 3-7 (Cont)

Vehicle	Power System Schedule Year	Power Required
ASTRO Tug (Cont)	Environmental and Instruments Approximate Total Power (2 hour work period)	0.60 kw 1.2 kw
Upcoming Payload	1968 (Estimated) Guidance and Control - Instruments Attitude control	1.10 kw 0.2 kw 0.9 kw
Interplanetary Travel	1971 - 19xx	300 kw

b. Description of Recommended System for Spacecraft, Tug and Upcoming Payload

(1) Hydrogen-Oxygen Auxiliary Power System

Studies revealed that a dynamic hydrogen oxygen auxiliary power system is capable of achieving performance equal to or better than that achieved with fuel cells or solar cells. In addition, this system can provide two additional desirable functions:

- Operate on low pressure boil-off gases from the main propulsion system tanks
- Provide water for the crews.

A typical H_2 - O_2 APS concept using a cryogenic engine is shown schematically in figure 3-30. In this system, an electrical alternator and a hydraulic pump (if required) are mounted on the H_2 - O_2 engine, giving both mechanical and electrical power.

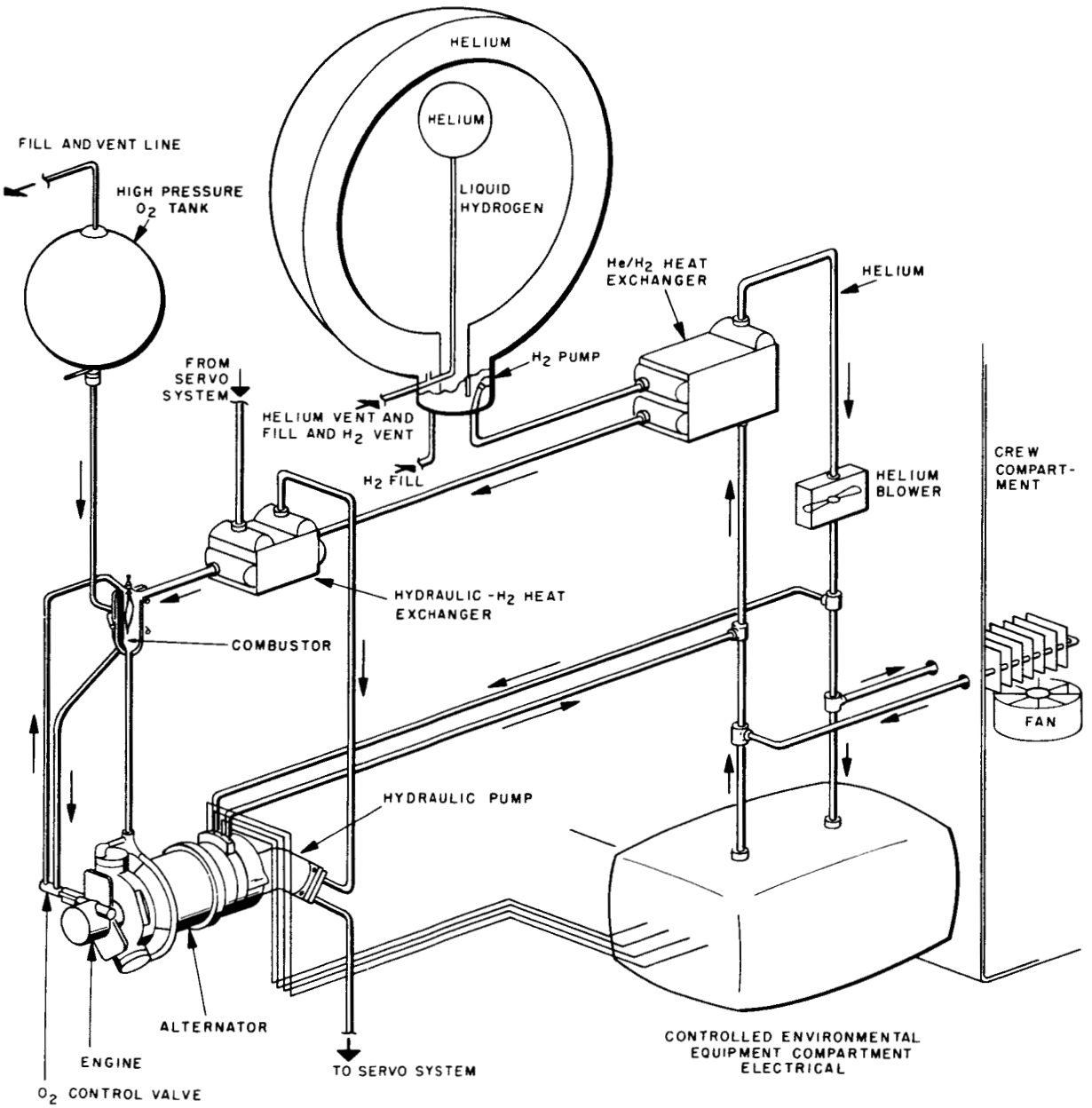


Figure 3-30. Hydrogen-Oxygen System, System Schematic

Since the hydrogen and oxygen propellants offer inherent engine system cooling, it is unnecessary to equip the engine with cooling jackets. Moreover, the engine exhaust contains steam that can be condensed for crew water. Water-payload penalties are thereby eliminated using the H_2 - O_2 propellant concept.

Table 3-8 shows the recommended systems for the four vehicles.

TABLE 3-8

AUXILIARY POWER SYSTEMS FOR THE OLO VEHICLES

Vehicle	Recommended Auxiliary Power System	Weight	Fuel Consumption
OLF	SNAP 8 Standby H_2 - O_2 System	2500 lbs* 2100 lbs	Negligible
Spacecraft	H_2 - O_2 Aps Manned Vehicle	78 lbs	
Astro Tug	H_2 - O_2 Aps Manned Vehicle	32 lbs	
Upcoming Payload	1. H_2 - O_2 Aps 2. Hot Gas Aps	32 lbs	
*Not including shielding			

(2) SNAP 8

The SNAP 8 power supply is under development by the AEC and NASA. This power plant for 30 kw and 60 kw operation

weights 1500 lbs and 2500 lbs respectively, not including the shielding required for personnel protection, and employs 7.0 kg fuel to produce 30 kw with a probable life of one year. The shielding required varies with the distance between the OLF and the SNAP 8 supply, the amount of time crew members will remain in orbit, and which sets of figures are to be believed as being the maximum radiation dose which humans can withstand and not suffer severe permanent damage. This shielding estimate varies from 10,000 lbs for spot shielding to 36,000 lbs for hemispheric shielding. Since no other secondary power sources can furnish the required power without severe operational limitations, the SNAP 8 system is the secondary power system selected for OLF operation.

SECTION IV

COST AND SCHEDULES

4-1. COMMUNICATIONS

The schedule for communications presented in figure 4-1 covers the period from the beginning of the calendar year 1962 through the calendar year 1967. For reference purposes the schedule for the intermediate and final OLF is also shown. The cost schedule summarized in figure 4-2 is projected through 1967. It should be noted that thoroughly qualified hardware must be provided by mid 1965.

With reference to figure 4-2, the lower curve totals 18.9 million dollars, the next higher curve totals 26.3 million dollars. The third curve totals 30 million dollars. The top curve is the sum of the lower three curves resulting in a total of 75.2 million dollars for the period 1962-1967.

4-2. NAVIGATION, GUIDANCE AND RENDEZVOUS

The schedule shown in figure 4-3 is a predication based upon the necessity of assuring adequate operational reliability of navigation, guidance and rendezvous equipment prior to the start of Orbital Launch Operation in support of the lunar mission. Figure 4-4 presents the cost of the proposed development up to the first manned lunar landing

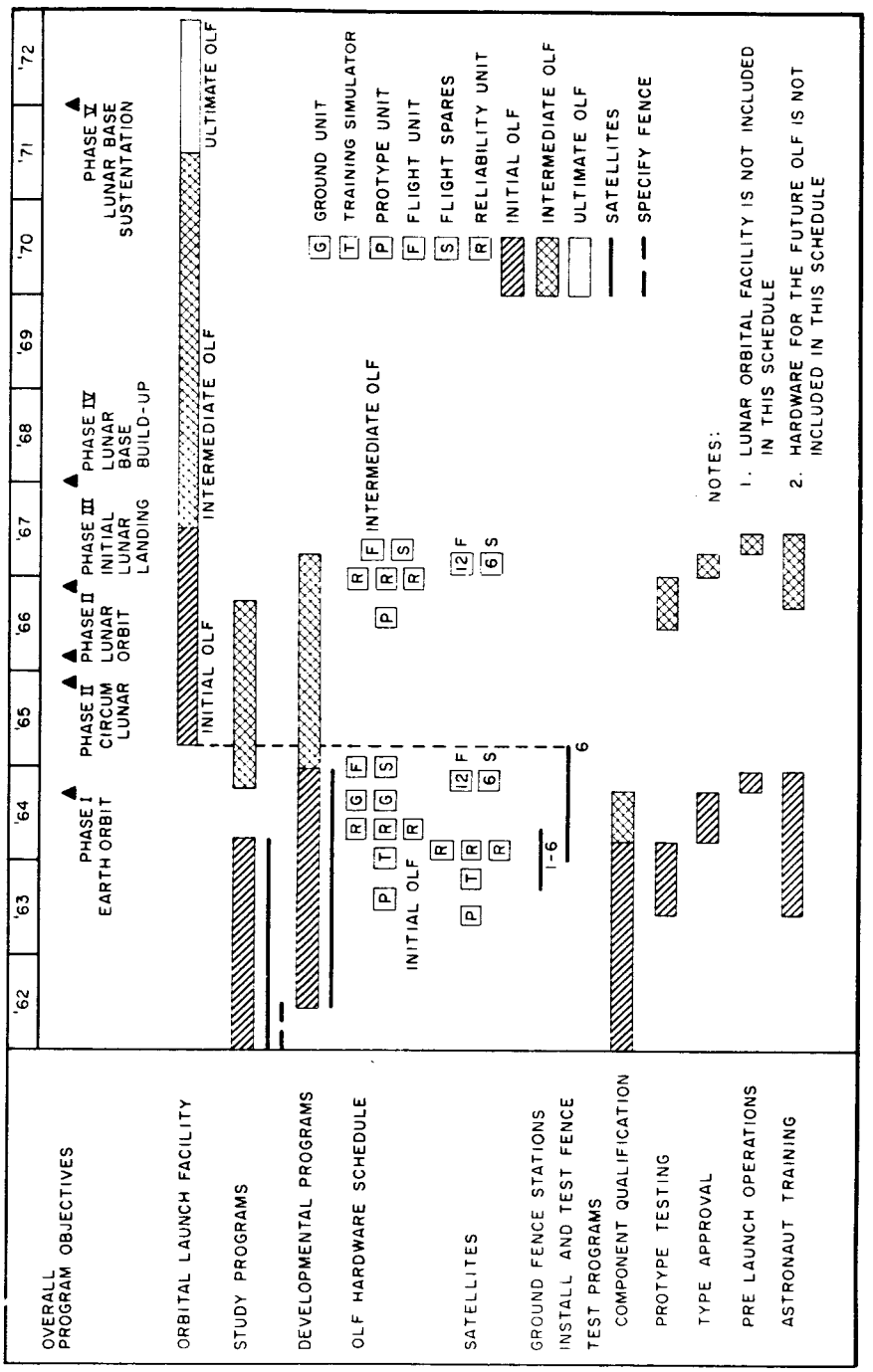


Figure 4-1. Program Schedule - Telemetry and Communications

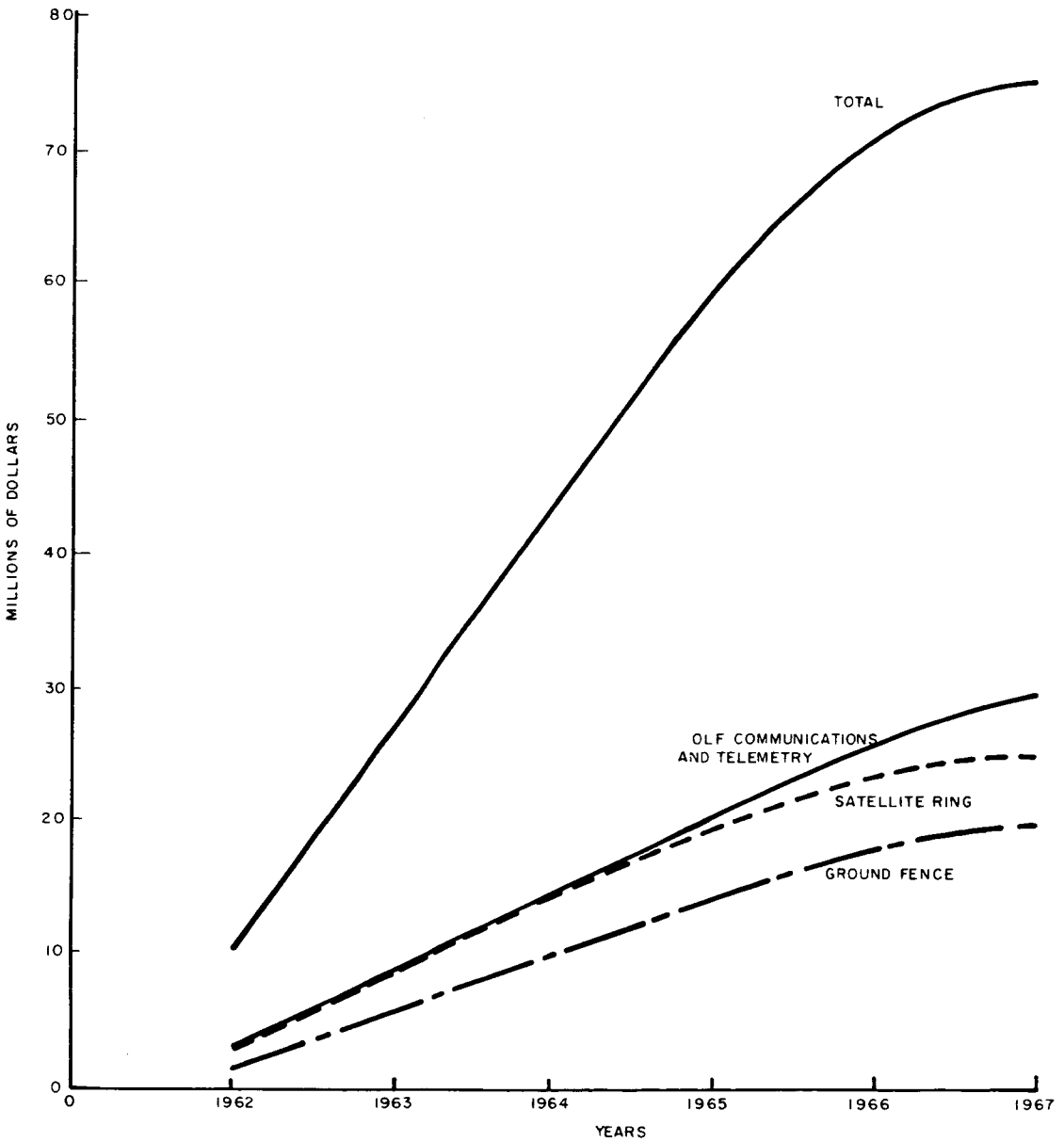


Figure 4-2. Projected Costs - Telemetry and Communications

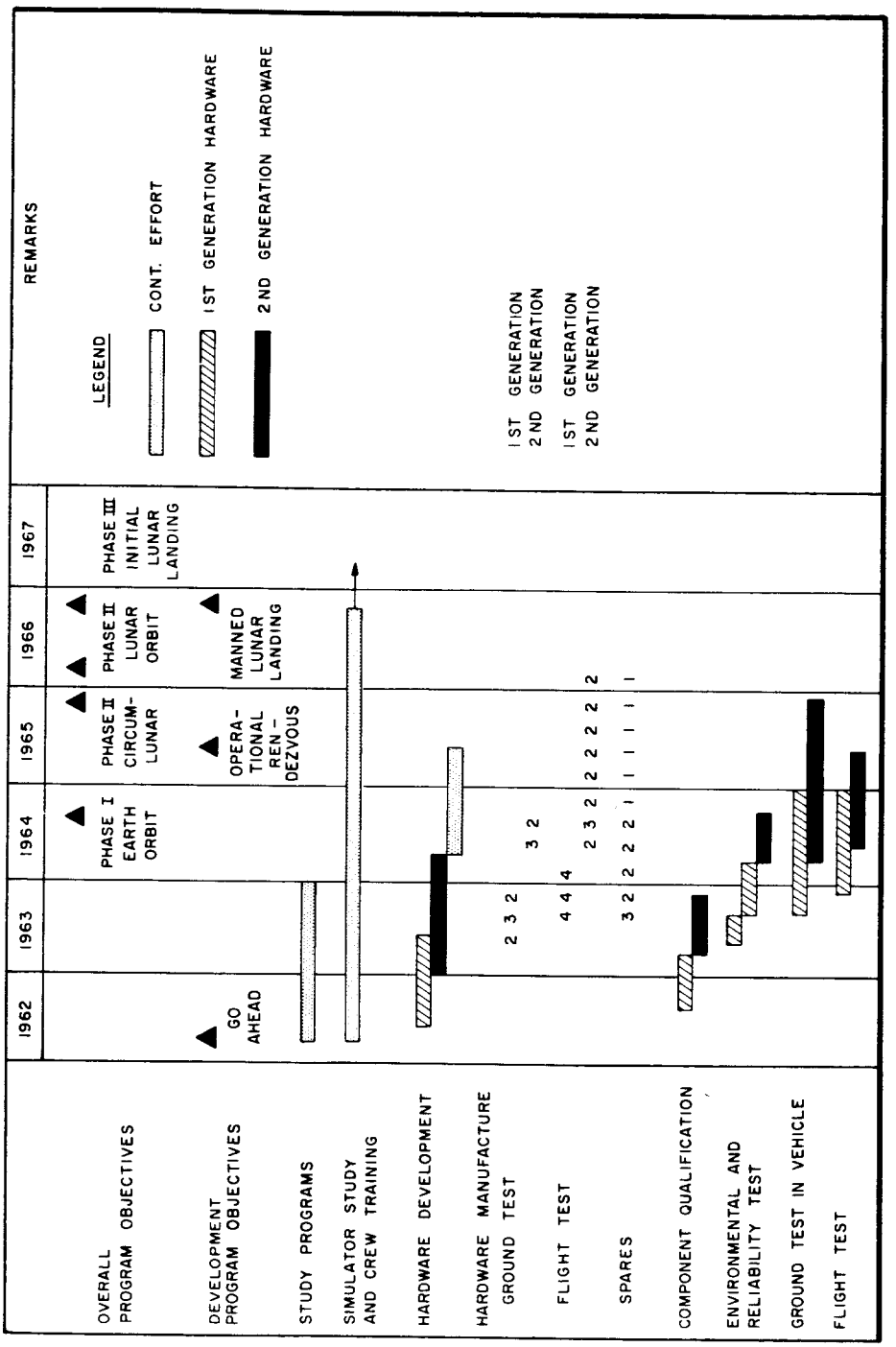


Figure 4-3. Program Schedule - Navigation and Guidance System Development

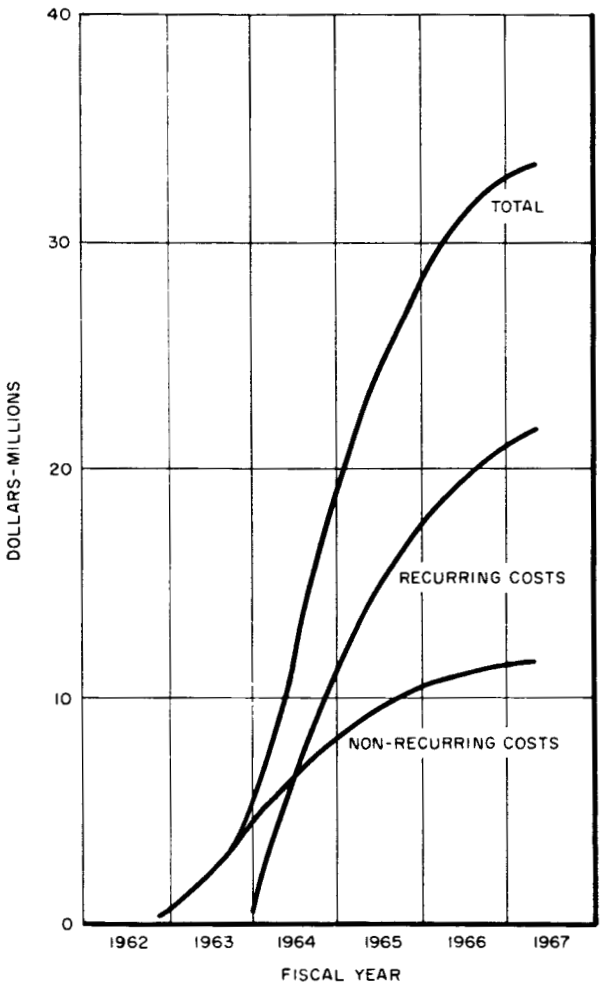


Figure 4-4. Projected Costs - Navigation and Guidance System

and covers all factors including simulator studies, test materials, hardware production and program management. The costs of launch boosters are not included. These costs are essentially independent of the presence or absence of an OLF.

4-3. ATTITUDE CONTROL

Figure 4-5 presents the overall schedule of attitude control development and production as compared with the major milestones of the OLO program.

A summary of attitude control equipment development costs is given in Figure 4-6. This summary includes development of sensors and attitude controls for use during retrieval and docking. This represents a major OLO problem area. Also included are the funds necessary to complete development of the storable bipropellant systems.

A summary of attitude control hardware procurement costs is also shown in figure 4-6. The total curve is the sum of both development and hardware.

4-4. COMPUTERS

a. Tracking Station Computer Subsystem

Table 4-1 presents a schedule of the budgetary costs for manpower and equipments required in the development of the initial land-based tracking station computer subsystem. Included in the table is an estimate of operating cost for one year, including spare parts and continuation programming/analysis.

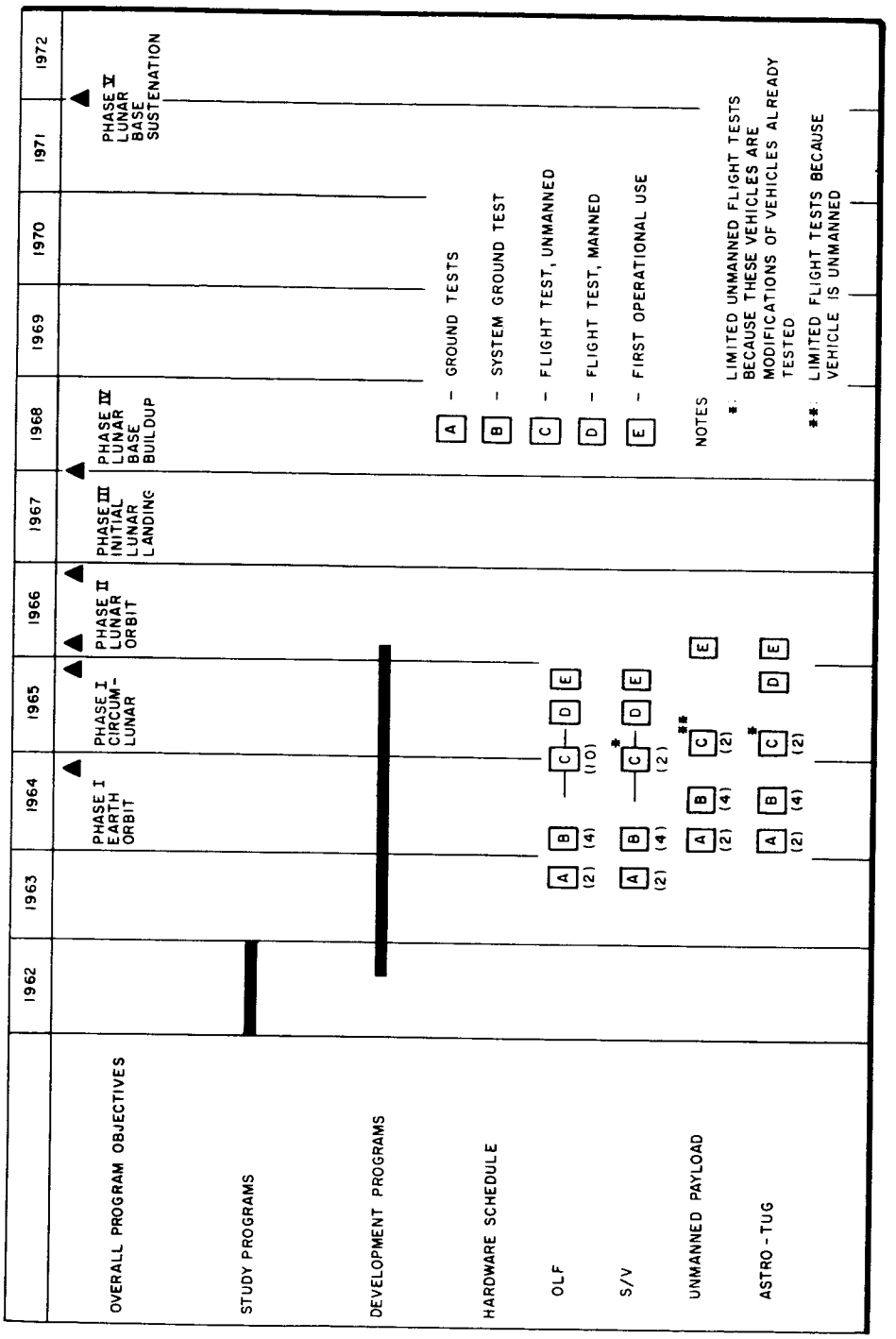


Figure 4-5. Attitude Control Development Schedule

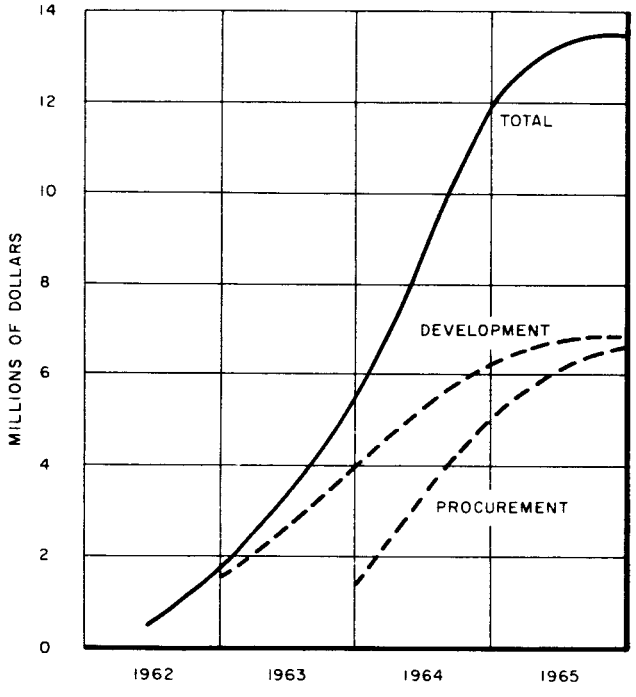


Figure 4-6. Projected Costs - Attitude Control Development

Figure 4-7 presents a development schedule for four computer subsystems to be used at land-based tracking stations. The choice of a four subsystem development starting 1 January 1962 is for illustrative purposes only. This figure summarizes the major development milestones and phases of the proposed program.

A program plan and associated cost schedule for ship-based tracking stations have not been generated during this study because of several unknowns in the areas of operational requirements, namely:

1. It is not clear at this time whether such a tracking station would utilize a multi-ship coherent Doppler tracking concept, or a single pulse-radar approach, or both.
2. More thorough operations analysis of the number, dispersion, and geographic locations of such stations is needed in order to investigate the concomitant logistics problems.
3. A political analysis may indicate that a land-mobile (via heliocopter, truck-van, airplane) tracking station concept is a feasible component for the OLO earth-based tracking system.

b. Space Borne Computer Sub-System

Development of the OLF on-board computer systems is also outlined in figure 4-7. Delivery estimates should be based upon the need-date for the entire system as conditioned by intermediate test goals and the availability of other systems. The times and dates outlined are not the earliest possible, but are balanced to give the lowest development costs.

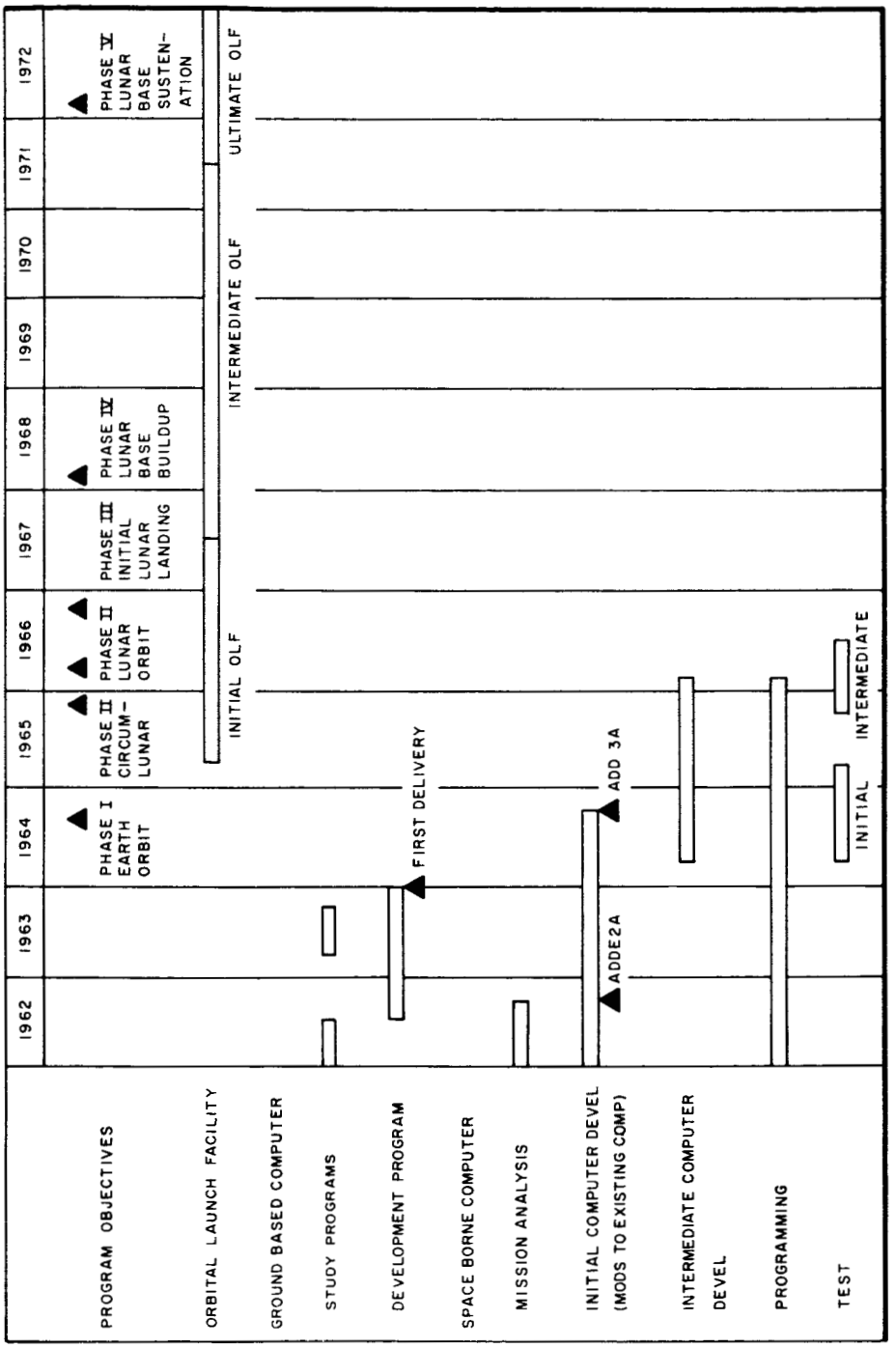


Figure 4-7. Program Schedule - Computer Developments

SUMMARY

It is deemed somewhat premature at this time to submit even the gross cost estimates for development of the space-borne computer subsystem. Additional investigation into the demands placed upon the computer by all sub-systemsof the OLO vehicles is necessary before more detailed and reliable program planning is possible.

TABLE 4-1

BUDGETARY COST SCHEDULE-GROUND TRACKING STATION COMPUTER

I.	Study Program, System Design Analysis		$ 82,000
II.	Development costs		1,734,000
III.	Operational costs		205,000
	Price of first system		$2,121,000
	Price of additional system, including spares and installation checkout	$1,231,000	
	Three additional systems @	$1,231,000	3,693,000
	Price of four systems		$5,814,000
	Estimated expenditures for the first six months, 1 June 1962 - 1 July 1962		$ 82,000
	Estimated expenditures for government fiscal year 1963		$5,422,000
	Estimated expenditures for government fiscal year 1964		$ 310,000

4-5. CHECKOUT EQUIPMENT

The schedule for checkout equipment is presented in figure 4-8. It should be noted that prototype equipment is required by the start of calendar year 1963 so as to insure

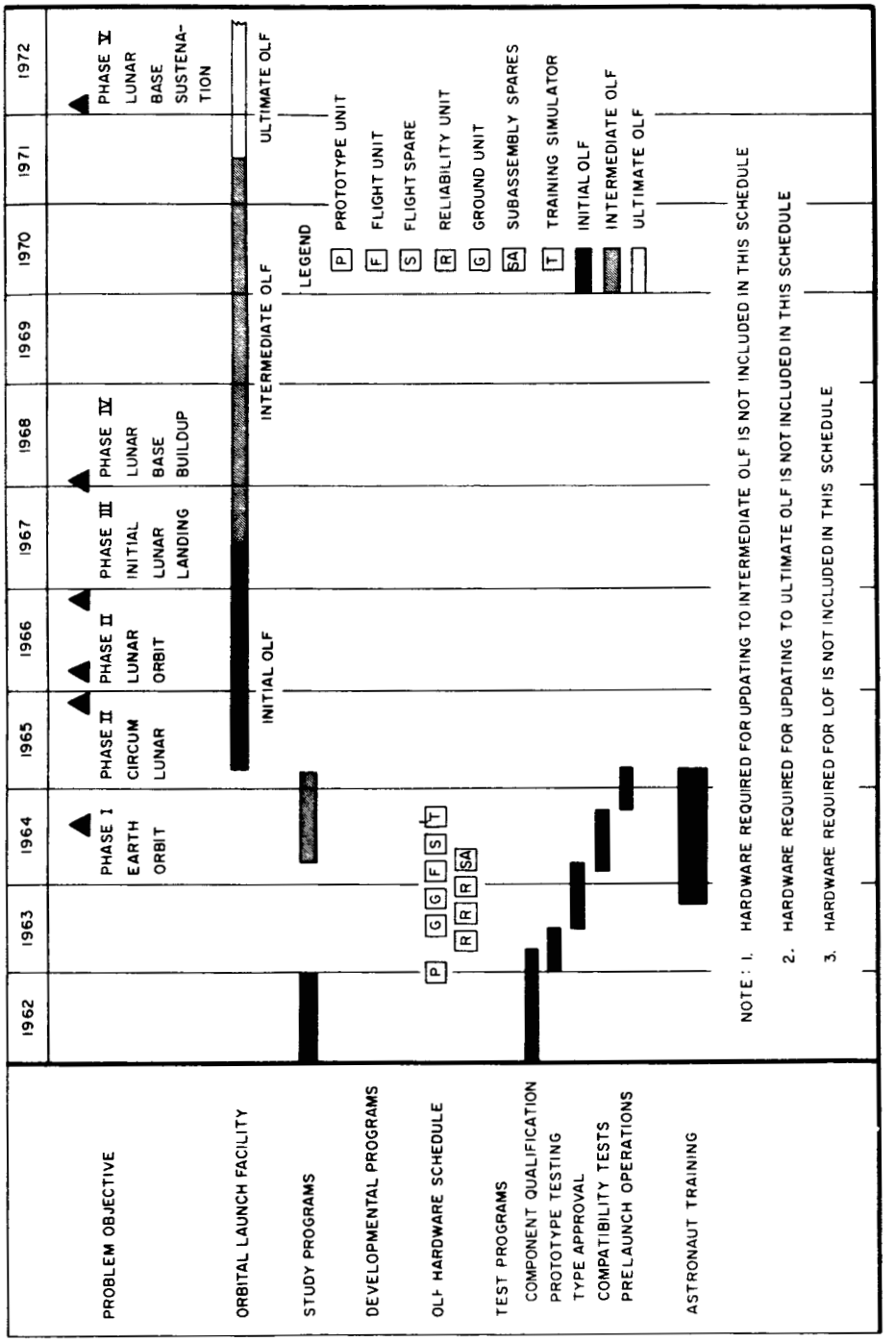


Figure 4-8. Program Schedule - Checkout and Countdown

reliable operational checkout equipment for the OLO program.

Table 4-2 presents a complete summary of costs for the checkout and countdown equipment, not including booster costs for equipment launch.

Figure 4-9 indicates the rate of expenditure and the cumulative total through the initial and intermediate OLF stages.

TABLE 4-2

SUMMARY OF COSTS FOR CHECKOUT AND COUNTDOWN

Initial costs		$17,538,200
Cost per unit		$ 1,014,875
Anticipated number of units (sides)*		
Flight	2	
Flight spare	2	
Ground	4	
System spare	1.2	
Launch pad spare	1	
Reliability	3	
Training simulator	1	
Factory checkout	1	
	15.2	
Total hardware costs		$15,426,100
Total program costs		$32,964,300

*Redundant sides are required for a complete system; that is, two sides will be supplied with a flight unit.

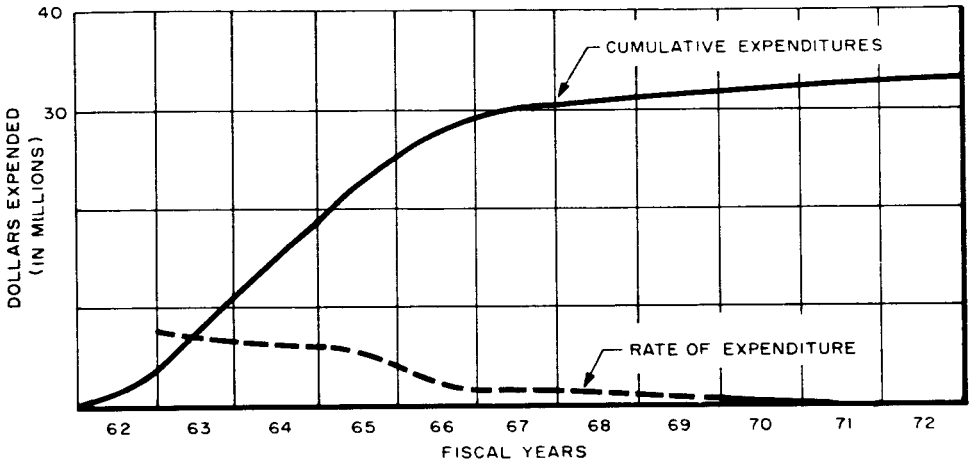


Figure 4-9. Expenditures for Checkout and Countdown

4-6. SECONDARY POWER

The power system development plan and schedules are a natural extension of work on an Air Force contract for development of an internal combustion hydrogen-oxygen engine for space power application. The present contract work allows the projected program schedule to be shortened correspondingly.

The schedule for the H_2 - O_2 secondary power systems development is shown in figure 4-10. Tables 4-3 and 4-4 give the total development costs for all systems and the per unit production costs. Figure 4-11 presents the cumulative cost buildings, not including production units.

A cost and schedule program for the NASA SNAP 8 nuclear powerplant has not been included. This is a program currently being sponsored by NASA and as such, presentation of this information would merely be repetitive.

TABLE 4-3

SUMMARY OF DEVELOPMENT COSTS FOR SECONDARY POWER
(H_2 - O_2 SYSTEMS ONLY)

Present AF Contract	$ 189,000*
Basic system development	
Study	200,000
Test	350,000
Advanced system development	
Study	350,000
Test	500,000

*Not included in total cost

TABLE 4-3 (Cont)

SUMMARY OF DEVELOPMENT COSTS FOR SECONDARY POWER
(H_2 - O_2 SYSTEMS ONLY)

Prototype system development	
Upcoming payload/astro-tug	$ 350,000
OLF	750,000
Spacecraft	950,000
Total	$3,350,000

TABLE 4-4

PRODUCTION UNIT COSTS PER UNIT

Upcoming payload/astro-tug	$ 50,000
Orbital launch facility	110,000
Spacecraft	90,000

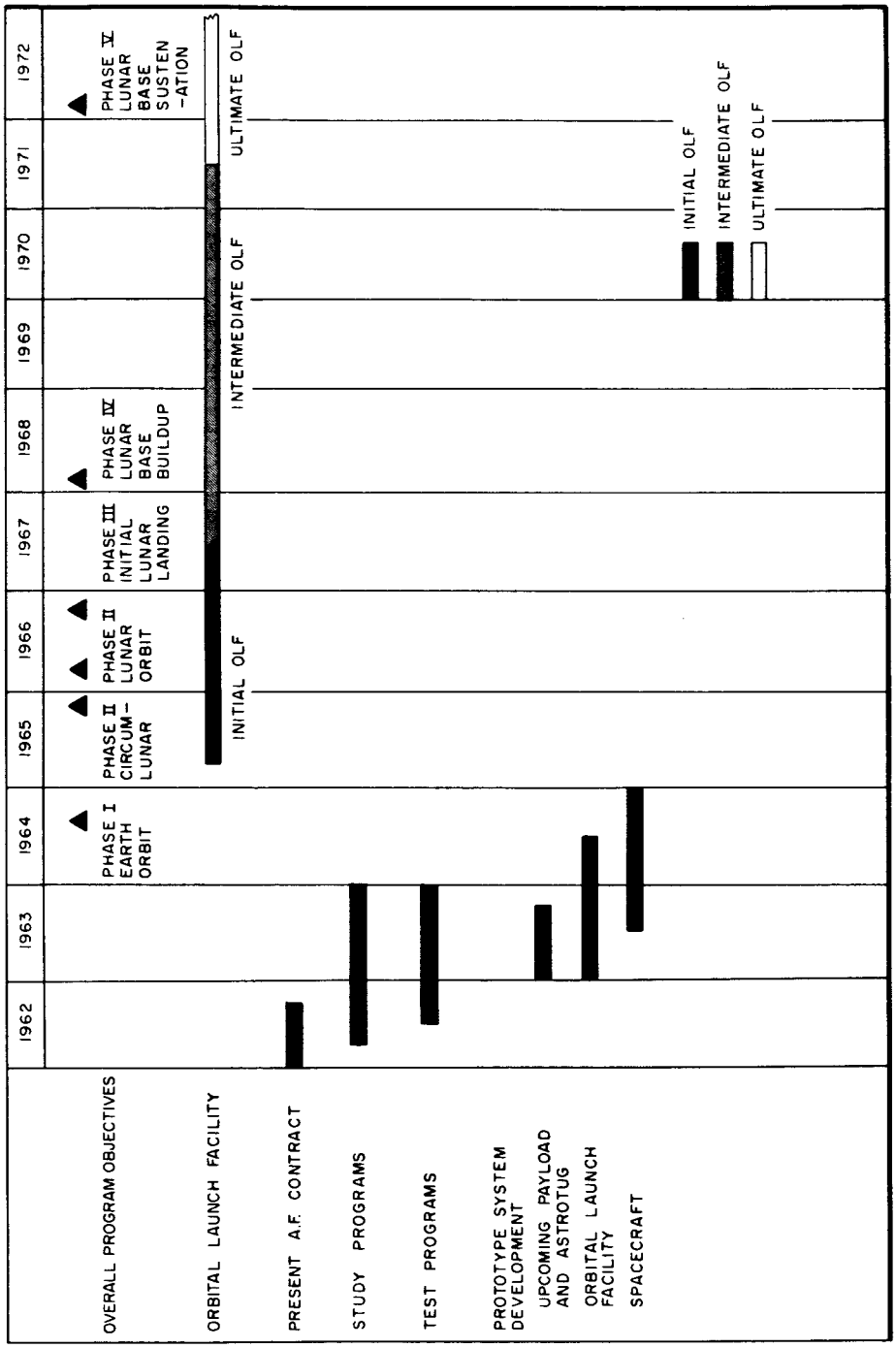


Figure 4-10. Program Schedule - Secondary Power System Development

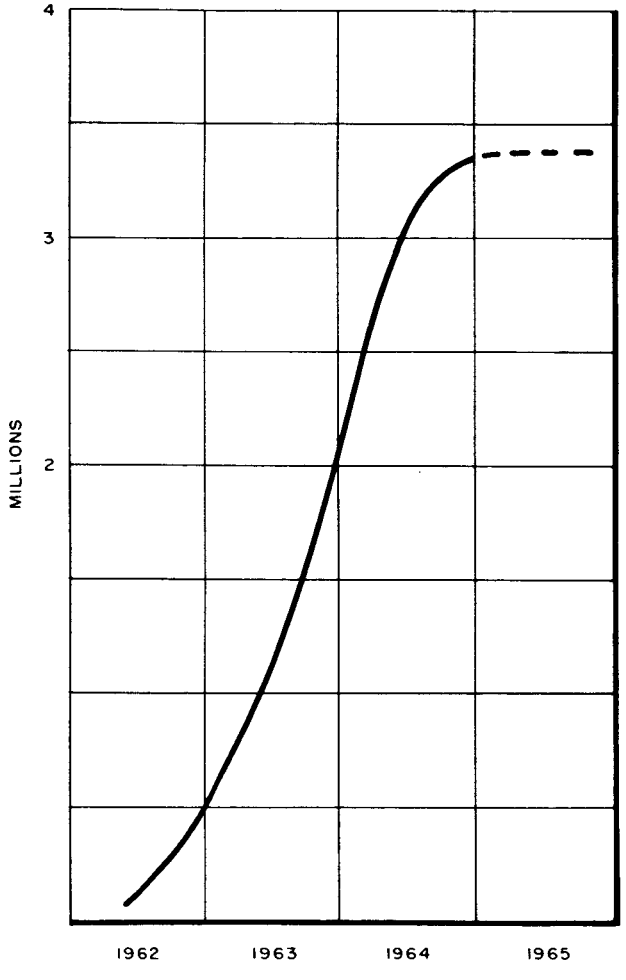


Figure 4-11. Projected Costs - Secondary Power System Development

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ORBITAL LAUNCH OPERATIONS

OTS

Xerox Price _____

Microfilm Price _____

VOLUME II

COMMUNICATIONS AND TELEMETRY

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TABLE OF CONTENTS

Paragraph		Page
	SECTION I	
	SUMMARY	1-1
1-1	Introduction	1-1
1-2	Evaluation	1-1
	SECTION II	
	INTRODUCTION	2-1
	SECTION III	
	BASIC CONSIDERATIONS	3-1
3-1	Organizational Plan	3-1
3-2	Basic Concepts	3-1
3-3	Differences Between OLO Communication and Other Communication Systems	3-5
3-4	Choice of Frequency Band	3-7
	SECTION IV	
	CONTINUOUS COMMUNICATIONS, GROUND TO OLF	4-1
4-1	Requirements for Continuous Communications	4-1
4-2	Extension of Existing Facilities	4-3

TABLE OF CONTENTS (Cont)

Paragraph		Page
4-3	Cost of Ground Extension	4-6
4-4	Use of Satellites	4-6
4-5	Comparison of 350 Statute Mile and 22,300 Statute Mile Satellite Systems	4-7
4-6	Non-Coplanar Satellite	4-9
4-7	Coplanar Satellite	4-11
	SECTION V	
	CANDIDATE OLF TO GROUND SYSTEM	5-1
5-1	Satellite Ring	5-1
5-2	Ground Fence	5-1
	SECTION VI	
	OLF R-F EQUIPMENT	6-1
6-1	R-F Equipment	6-1
6-2	Circuit to OLV	6-2
6-3	Circuits to Astrotug and Payloads	6-3
6-4	Circuits to Extra-Vehicular Man	6-3
	SECTION VII	
	MODIFICATIONS OF SATELLITE RING FOR CONTINUOUS TRACKING COMMUNICATIONS	7-1

TABLE OF CONTENTS (Cont)

Paragraph		Page
	SECTION VIII	
	OLF COMMUNICATION AND TELEMETRY TERMINAL EQUIPMENT	8-1
8-1	General	8-1
8-2	Data Handling Requirements	8-1
8-3	Functional Equipment	8-2
8-4	Operational Requirements	8-3
8-5	Equipment Mechanization	8-3
8-6	Terminal System Features	8-4
	SECTION IX	
	CONCLUSIONS	9-1
9-1	Other Systems	9-1
9-2	Candidate System	9-1
9-3	Areas Requiring Further Study	9-9
	APPENDIX A	
	BASIC NETWORK GEOMETRY	A-1
	APPENDIX B	
	TYPES OF CHANNELS	B-1
	APPENDIX C	
	NET CONTROL AND CIRCUIT MANAGEMENT CONSIDERATIONS	C-1

TABLE OF CONTENTS (Cont)

Paragraph		Page
C-1	General	C-1
C-2	Simplex Radio Communication	C-1
C-3	Duplex Radio Communication	C-2
C-4	Circuit Management and Frequency Plans	C-2
C-5	OLO Interconnection Plan	C-4
	APPENDIX D	
	THE FENCE	D-1
	APPENDIX E	
	THE SATELLITE RING	E-1
E-1	General	E-1
E-2	Frequency Plan	E-1
E-3	200-Kc System	E-3
E-4	Satellite Equipment List	E-8
	APPENDIX F	
	OLF COMMUNICATION AND TELEMETRY TERMINAL SYSTEM	F-1
F-1	General	F-1
F-2	Terminal System Data Requirements	F-1
F-3	OLF Communications Terminal System-Operating Modes and Evolution	F-4

TABLE OF CONTENTS (Cont)

Paragraph		Page
F-4	OLF Reliability	F-13
F-5	OLF Maintainability	F-15
F-6	OLF System Flexibility	F-15
F-7	Role of Man	F-16
F-8	Candidate Systems	F-18
F-9	Description of OLF Conceptual Communications Terminal System	F-24

LIST OF ILLUSTRATIONS

Figure		Following Page
2-1	Basic Requirement	2-1
4-1	Earth-Based Tracking Coverage of OLF for a 24 Hour Period	on 4-4
4-2	Maximum Line-of-Sight Path	4-5
4-3	Equatorial Ground Coverage from OLF	4-5
4-4	Pattern for Continuous Coverage	4-5
4-5	Rough Estimate of Communications Ground Net Costs	on 4-6
4-6	Types of Satellites	4-7
4-7	Comparison of Characteristics of Three Types of Satellites	on 4-8
4-8	OLF Satellite Ring - Ground Fence Concept	4-11
4-9	North-South Fence Geometry	4-11
5-1	Typical Space Relay Satellite	5-2
5-2	OLF Equipment	5-2
6-1	Path of OLV Relative to Path of OLF for Altitudes Below 6,000 Statute Miles	6-2
8-1	PCM System	8-3
9-1	Comparison of Three Basic Concepts for Continuous Communication	on 9-2
9-2	Candidate System	9-3
9-3	Table of System Equipment	on 9-4

LIST OF ILLUSTRATIONS (Cont)

Figure		Following Page
9-4	Program Schedule-Telemetry and Communications	9-8
9-5	Projected Costs	9-8
A-1	Geometric Concept Disc Rotating at Angular Velocity ω_1	A-1
A-2	Geometric Concept Segmented Disc	A-1
A-3	Geometric Concept Interchanged Brush and Fixed Connections	A-1
A-4	Geometric Concept Addition of User C	A-1
A-5	Geometric Concept Addition of User C at Different Angular Velocity	A-1
C-1	Simplex Type of Radio Communications	C-1
C-2	Duplex Type of Radio Communications	C-2
C-3	Net Control Station - Method A	C-2
C-4	Net Control Station - Method B	C-3
C-5	Alternate Interconnection Plan	C-3
C-6	Typical Station B of Alternate Interconnection Plan	C-3
C-7	Plan for OLO Communication Link	C-4
C-8	Sequence of Possible Communication Links	C-4
D-1	Fence Relay Station	D-1
D-2	UHF Section of Fence Station	D-2
D-3	Half of the HF Section of the Fence Relay Station	D-4

LIST OF ILLUSTRATIONS (Cont)

Figure		Following Page
D-4	Fence Costs for 48 KC Bandwidth	on D-5
E-1	Satellite Ring	E-1
E-2	Ring Frequency Plan	E-1
E-3	Ground Station	E-3
F-1	Estimate of Telemetry Measurement Requirements	on F-4
F-2	Summary of Telemetry Requirements By Operation	on F-11
F-3	Measurement Handling Operation	on F-12
F-4	OLF Measurements Requirements	F-13
F-5	Communications Terminal System	on F-14
F-6	OLF Telemetry System Programming	F-17
F-7	Comparison of Candidate Systems	on F-19
F-8	Telemetry Modulation Comparison	on F-20
F-9	Projected Electrical Characteristics of OLF Telemetry System	on F-25
F-10	Physical Characteristics-Basic Telemetry System	on F-26
F-11	Physical Characteristics-Additional Telemetry Equipment	on F-27
F-12	Alternate Hybrid System	F-27
F-13	Decoder System	F-28
F-14	Programmable Multiplexer Section	F-30

LIST OF ILLUSTRATIONS (Cont)

Figure		Following Page
F-15	Basic Switch	F-30
F-16	Programmed Multiplexer Driver	F-31
F-17	Waveforms of Driver Unit	F-31

SECTION I

SUMMARY

1-1. INTRODUCTION

The Sperry Rand Corporation is supporting Marshall's Orbital Launch Operations Study (Vought Astronautics Contract NAS 8-853) by investigating the major Astronic requirements for the program. The subject of this volume is Communications and Telemetry, which covers the work accomplished by the Sperry Rand Systems Group assisted by the Sperry Utah Company, Salt Lake City, Utah and the Sperry Development Center, Sunnyvale, California.

1-2. EVALUATION

The principal conclusions of this work are presented in the following paragraphs.

a. The communications and telemetry aspects of the OLO program are within the time scale for state-of-the-art development as conceived by Sperry Rand Corporation. No major technological breakthroughs are necessary to successfully establish OLO operations.

b. Certain follow-on study work is absolutely necessary to firmly establish plans for successful OLO operations. In the area of communications and telemetry these are as follows:

(1) The requirement for continuous communications must be established in depth and fully documented since present indications are that communication continuity is a major requirement for successful OLO operations.

(2) The continuous communications system utilizes satellites to great advantage. Sperry Rand Corporation proposes to develop the specifications and schedules for the necessary satellites if required.

(3) Specifications and schedule for the complete communications system should be developed by June 1962 to meet 1965 circumlunar flight schedule.

(4) Realistic reliability studies are needed for the spaceborne communications arrangement. These should include breadboarding and testing of key elements.

Since the work is well within the capabilities of the SUCO and SDC Divisions of the Sperry Rand Corporation, it is requested that assignment of this work to Sperry Rand for the next funding period be considered. Budgetary cost information for this work is presented in Section IX.

c. Other recommended follow-on work is presented. These items, while not placed in the absolutely necessary category, are still considered necessary from the standpoint of well balanced planning and proper initiation of work programs necessary to meet the objectives of the national OLO program on schedule.

(1) The NASA plans for integration of communications, telemetry, ranging and range rate determination through the

use of the same equipment using different forms of modulation should be incorporated in the OLO operation if timely. The work that JPL, MIT and STL have been conducting indicates a strong trend in this direction but considerable study is required to determine whether or not any of the concepts will be generally implemented in the existing ground facilities soon enough.

(2) After the broad communication plan has been partially detailed, a check should be made on uniformity and interchangeability in order to minimize the logistic problems aloft.

(3) Work should be continued concerning the use of low-flying satellites as the transmission medium from outlying radar sites to the operation center.

(4) Investigate methods for reducing intentional interference from an enemy and also possible unintentional interference.

d. These programs are also recommended for Sperry Rand support of the Marshall Space Flight Center OLO Development Program. The funding requirements for each of these work items is presented in Section IX.

e. Some of the major technical conclusions which are of interest to the Marshall Space Flight Center as an aid in planning the work are listed.

(1) Continuous communications and telemetry is of paramount importance to the probability of OLO mission success and on the order of 75 kc of bandwidth can be provided by low-flying satellites in the time period required.

(2) Switchable duplex circuits should be provided from the OLO operations center and computer to various portions of the OLO complex aloft in order to provide the necessary flexibility and allow for expansion, if required.

(3) Low-flying satellites may be used to supply data input paths to the operations center from outlying radar sites.

(4) The low-flying satellites system is superior to the 22,300 mile high flying system by a launch weight ratio of about 21:1 or higher.

(5) RF equipment should be readily tunable to three or more frequencies in the same two kmc band.

SECTION II

INTRODUCTION

The purpose of this report is to present the results of investigations conducted by the Sperry Rand Systems Group in the field of communications and telemetry. The Sperry Rand Corporation is conducting an across the board analysis of the various phases of Orbital Launch Astrionics to aid the Marshall Space Flight Center in developing a firm and well documented plan for establishing the highly advantageous technique of Orbital Launch. The work is being conducted in connection with Vought Contract NAS 8-853, and is being guided by Vought work statements and coordinating documents. Every advantage is being taken of the results obtained by Vought in the earlier phases of the contract.

The work included in this volume is an investigation of the various concepts and techniques for communications and telemetry.

Figure 2-1 shows a model of the desired communications and telemetry system. High quality, reliable, and continuous communications is required between the,

a. OLF and Ground

b. OLF and Spacecraft

c. OLF and Astrotug

d. OLF and Extravehicular Man.

This communications and telemetry system must provide the following:

a. Two-way data links and voice channels

b. Data detection, processing and storage

c. Telemetry.

Various concepts and techniques are investigated and a candidate system is derived by considering the following trade-off factors,

a. Performance

b. Overall cost

c. Reliability and life

d. Maintenance and logistics

e. Weight, size, and power drain of spaceborne equipment

f. Schedule.

The resulting candidate system that is described in this study report has the following outstanding characteristics,

a. High quality, continuous duplex communications between all stations at all times.

b. Only one frequency band for the entire system, thus providing the following inherent advantages,

(1) A minimum number of antennas

(2) Simple logistics

(3) Ease of maintenance.

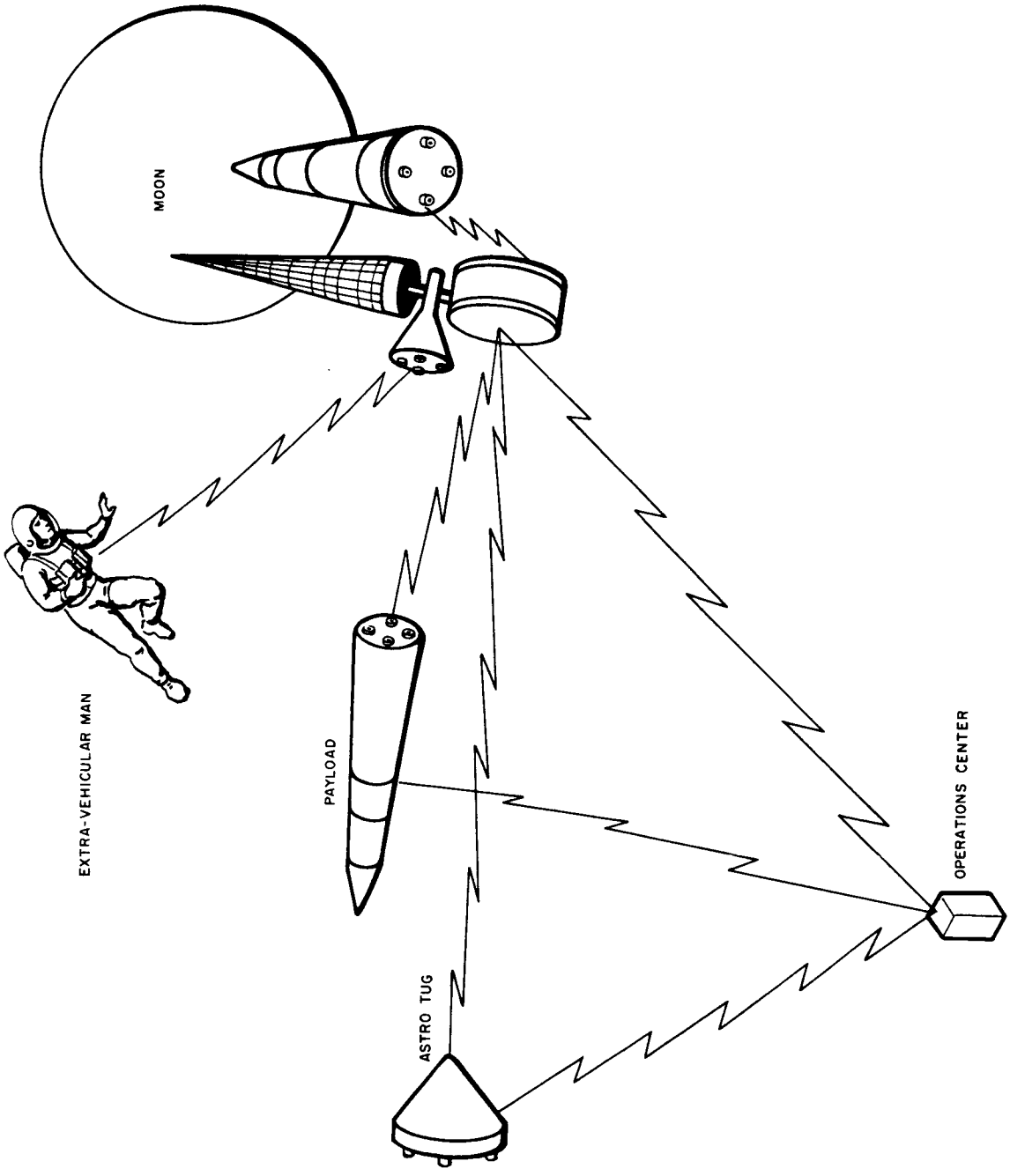


Figure 2-1. Basic Requirement

c. Every communications channel provides 50kc or more of bandwidth for multipurpose and flexible use.

This phase of the Astrionic work is being funded by Sperry Rand Corporation as a contribution to establish early the key engineering programs which are necessary to achieve success in this important endeavor. An outline of these key programs is presented in Section I of this volume. A more complete presentation may be found in Sections V, VI, VIII and IX.

SECTION III

BASIC CONSIDERATIONS

3-1. ORGANIZATIONAL PLAN

A discussion of the OLO communication problems is introduced by establishing several concepts by which a test of any conclusions is possible as the work progresses. Following this discussion of concepts, the differences and similarities between the OLO communication problems and the communication problems of other systems are outlined. This leads directly into the broad background of the candidate systems. The RF considerations will be presented first, followed by the telemetry terminal equipment. After presenting the best available facts and figures on the candidate system, comensurate with the total time and effort available to this date, the areas needing further study will be discussed.

3-2. BASIC CONCEPTS

a. Line Switched Versus Message Switched Systems

Although in concept several investigators in the same environment may arrive at similar answers to the same problem, it is usually found that the grading of the answer is scaled to the experience of the investigator. For OLO communications it is important to take advantage of the major collective experience of the industry. For example,

profit may be gained from established solutions to the problem areas encountered in our national and international telephone systems, in aircraft control and other communication systems. These and other multiple user systems have something in common with the OLO operations. A requirement for "real time" operation is a fundamental of voice type operation. However, voice operation is sometimes reduced to non real time by handling recorded messages relayed through from one station to the next on a store and forward basis. On the other hand, no matter how inefficient they may be, there are cases where real time hot circuits are a necessity. Fire alarm and police call boxes might be placed in this category. In other cases, transmission capability cannot be allowed to lie idle and must be shared with other users, for example, normal telegraph systems. The "real time" type circuit for multiple users will be referred to as a line switched system and the non real time circuit as a message switched or store and forward system. There are definite uses for each. However, there are occasions when the proponent of one type of system forgets of the existence of the other. An occasional test of the candidate system to determine which system or combination of systems best satisfies the requirements is in order. The results of the subsequent study show the superiority of the line switched system for the OLO communications and as a subsidiary consideration duplex circuits are recommended.

b. Signalling and Supervision

As indicated later, the OLO communication complex is a multiple user system. Multiple user systems require certain functions that may be overlooked if previous experience has been limited to working in two terminal, essentially single purpose systems. These added functions are signalling

and supervision. For this discussion these terms are used in the broadest sense. Signalling covers the area of addressing as well as the process of alerting the addressee of a call from another user. Supervision is related to signalling, in that, both may be considered in the category of necessary overhead in order to assure delivery or to inform the originator of progress in delivery. Supervision may be considered as information exchange between terminal points of a circuit that give the condition of the circuit and contribute only indirectly to the text or prime data being transmitted. Supervision includes such information as no-such-address, circuit-busy, circuit-out-of-commission, and message accounting numbering systems. These functions have quite a different form depending whether consideration is given the line switched system, the message switched system or other systems that might be classed as hybrids or are not well differentiated from the point of view of how the routing is achieved. Very simple systems reduce the signalling and supervisory information to a priori information. As the system becomes more flexible, much of this class of information varies from day to day, minute to minute, and must therefore be transmitted at approximately the same time as the text. There is a complete spectrum from bunched supervision once only (a priori) to fully distributed supervision with supervisory elements associated with each binary digit transmitted (error checks or correction). Another variable, on a somewhat orthogonal axis, is the manner in which the supervision shares or does not share the path facilities of the text or prime data. A priori information is really one extreme on this axis, that is, certain exchange of information concerning the operation was made without using the circuit in question; the supervisory information did not share the path facilities of the text or prime data. Some

portion of the supervisory information is always handled in this manner. An example of a system that is toward the other end of this axis is a teletypewriter system on the poorest type of transmission line. Almost all the supervision is carried on through the use of spelled out words that are time shared with the text. On the other hand, most telephone systems provided some measure of supervision simultaneous with the text although the great majority is time shared with the text. Many military communication centers provide separate circuits for the supervisory information and send address and text only on the main circuits. Attention has been given this concept since it is an area requiring detailed study before equipment specifications can be written.

c. Flexibility and Expandability

Another concept that should be used as a check on a communication system is its flexibility, including expandability. It has been shown that it is economical to plan for the future in communications and use a building block principle wherever possible. The paramount theme of this concept is to satisfy one user today and another slightly different user tomorrow without completely rebuilding. In this regard, there is little to distinquish between communications and telemetry, it is all data transfer. The capability used in communicating between machines needs little change to be able to exchange data between men. In the case of the OLO, there is a definite need for freedom in expansion since all of the requirements of 1967 cannot be foreseen today.

d. Step Forward

A step forward concept that is a bit difficult to define is worthy of consideration. That is, after investigating utilization of existing concepts, equipment and installations and finding them inadequate, thought should be given somewhat beyond the solution of the immediate problem. This is done to ensure that the step forward is great enough to be worth the effort in breaking away from the more conventional scheme. In testing, project perhaps to a limit, even if absurd, and then retrench to a reasonable position.

3-3. DIFFERENCES BETWEEN OLO COMMUNICATIONS AND OTHER COMMUNICATION SYSTEMS

a. Multi Station Complex, Continuous Communications

So much for basic philosophy, consideration is now given to the major differences and similarities that exist between the OLO communications and other communication projects. Of course there are other space communication problems that are quite similar, but most of those to this date do not include several "birds" of various natures in simultaneous and cooperative operation in orbit. Another difference between this and other space problems that can be rather severe in its consequences is that of continuous communications over an extended period of time (years). This follows the reasonable assumption that the OLF is to be permanent and not consumed by launch of the spacecraft. Both parts of this statement are differences from existing techniques. First, continuous communications implies a need for "real time" communications which has not been available previously. Second, the idea of

having men aloft continuously for years places a permanence on the communication facilities that can considerably alter the approach over that of a one-shot approach. Reference to the figure 2-1, showing the overall communication complex, immediately demonstrates that there may be several "birds" in operation simultaneously and provisions must be made for the concepts of signalling and supervision; for example, addresses are required that are not required in two-point communications.

b. Continuous Communications Vs Continuous Tracking

Truly continuous communications is a severe requirement and requires a forthright attack that may not be a mere extension of the facilities used for other programs. For the present discussion, the requirement for continuous communications is divorced from that of continuous tracking of the OLF. In the event that continuous tracking should be found to be a system requirement, the conclusions reached in this discussion would be somewhat modified. Reference will be made to the area requiring modification after the discussion excluding the possible modifications have been completed.

c. Coordination With Other Efforts

In all large operations there is some overlap between the efforts of one assigned group and that of another assigned to a different portion of the same overall project. One of the jobs of the project manager is to provide each participating group with the coordinated inputs of the total program as the project proceeds. This is a very difficult job when the state-of-the-art is being constantly improved. It is even more difficult for a limited study to include all inputs of this type

since many of the inputs are modified by the decisions of the project director. In this particular study, it is apparent that complete compatibility with the design of the spacecraft or certain of the ground facilities may not be indicated. The efforts of NASA in integrating several communication and sensing functions on one RF carrier has not been discussed. In particular, JPL, STL and MIT in the broadest sense have made considerable progress in the past few months toward this goal with specific recommendations for an Apollo "goal system". This study is in complete agreement with the concept of integration of several functions on a single RF carrier, or limited self sparing of a group of carriers. Although it is believed that there is nothing basically contradictory between this study and the integrated concepts or Apollo "goal system", there is some further coordination required before the detail specifications for the OLO communications system may be written.

3-4. CHOICE OF FREQUENCY BAND

It is well known that the atmospheric window is in the region of one to ten kmc. In this atmospheric window, smallest amount of noise is contributed from the galaxy and at the same time transmission is not cut off by atmospheric absorption. Early experiments have been accomplished on considerably lower frequencies due to equipment state-of-the-art and the immediate availability of existing equipment at the lower frequencies. In addition to the noise advantage, higher frequencies reduce the equipment weight since smaller antennas can be used to limit the solid angle of transmission to the desired value. At the present time, equipment is available in the two kmc region of the required stability at modest size and weight. There does not seem to be any dissention to the statement that within the next ten years the majority

of the space to ground communication will be within this range, going to the higher end of the range in later years. It seems reasonable then to plan on the highest frequency in this band which can be readily generated and detected today, so that the obsolescence in the next few years is kept to a minimum. Standards have been developed and frequency allocations are being made in the two kmc band and considerable equipment is becoming available at a rapid rate. The period from the present to 1964 is considered adequate to assure the reliability of the higher frequency equipment.

SECTION IV

CONTINUOUS COMMUNICATIONS, GROUND TO OLF

4-1. REQUIREMENTS FOR CONTINUOUS COMMUNICATIONS

Continuous communications requires at least one continuous channel and preferably several between the OLF and ground, the ground point being connected with the operations center with a set of channels of the same capacity as that from ground to OLF. Appendix A gives some of the details of this configuration in the form of a simplified network. This provides background details for the difficult problem of continuous ground to OLF communications. Further, it requires at least one channel either to ground or to the OLF from any other "bird" in the near range of the OLF. For truly continuous operation this means that each of these circuits must be duplex so that one station cannot lockout another by extended one-way use of reversible simplex. Appendix B gives an expansion of the terms simplex and duplex for those that may not be familiar with this terminology. It also explains the mentioned lockout danger. Simple switching is implied between the circuits connecting the operations center to the OLF and interconnection at the OLF with the other "birds" in its immediate vicinity. Some of the network control problems which may be encountered in interconnection are idicated in Appendix C. This important concept of network control has been detailed in Appendix C since it is rather lengthy and may be well known to some readers. Since it will be demonstrated that there is considerable expense involved in the OLF to ground link, a few

of the highlights to justify the need for continuous communications must be given. First, it may be argued falsely that no communications is required. Everything can be programmed as a sequence of operations clocked out of storage. This would be a true statement if everything were known to the desired degree of accuracy and reliability. It is definitely not true when human failures, equipment failures, weather changes, and other occurrences of unknown or only statistically known character must be taken into account. Then it may be said that in "rule by exception" the need to only transmit the corrections to the stored program required by failures or insufficient knowledge exists, and consequently only a very small portion of the total operation is consumed in sending corrections and discontinuous communications are satisfactory. Again this statement is probably valid so long as the variables in the program change smoothly, but may be false for step function changes to the program such as may be introduced by catastrophic failures of man or machine. The OLO is a very complicated operation involving the coordination of many, many functions and people and a failure to be able to transmit a "hold" just before launch of new crew members might require a $100,000,000 abort due to a failure in the OLF. A calculated risk is taken in assuming that any given time may be allowed for interruption of communications. In view of the length of the total operation and the total number of launches required, it becomes increasingly risky to leave holes in the communications. Another reason for communications continuity may be drawn from experiences at sea or in the air. It concerns the logging of information about the voyage. For any voyage in which the log book is salvageable from the wreckage, a self-contained log is satisfactory. However, remote logging is a necessity when a reason to believe that circumstances may obliterate the information in the

self-contained log. If the instant of obliteration is not predictable, continuous entries into the remote log are desirable. If it is believed that circumstances leading to the catastrophe might prevent similar catastrophes in the future, a rather reverent respect for the concept of an adequate log book is justified. For this purpose, the word "log" has been used in a very broad sense and includes information entered in the records by the machine complex as well as by the men aboard.[1]

After justifying continuous communications on other grounds, it is found that there are many bonus effects accruing after the basic objectives are satisfied. Operational convenience is one of the simplest to understand. Instead of reconstructing certain functions of conferences to an erratic come-and-go timetable, scheduling can be done without restriction.

4-2. EXTENSION OF EXISTING FACILITIES

In order to determine the magnitude of continuous communication type operation, the following brief analysis based on an extension of existing ground facilities is given. The chart of figure 4-1 gives the time-in-view of the OLF for ten selected locations on the earth. A series of fifteen consecutive orbits of the OLF are shown which very nearly close after 24 hours. On the first orbit, the time-in-view for Cape Canaveral is 9.1 minutes, for Canary Island it is 1 minute, etc. Near the bottom of the column are given the time-out-

- - - - - - -

1. W.D. Winters and N. Pace at the Lunar Flight Symposium Denver, December 1961. Papers presented to indicate that continuous telemetry of physiological data may become a necessity.

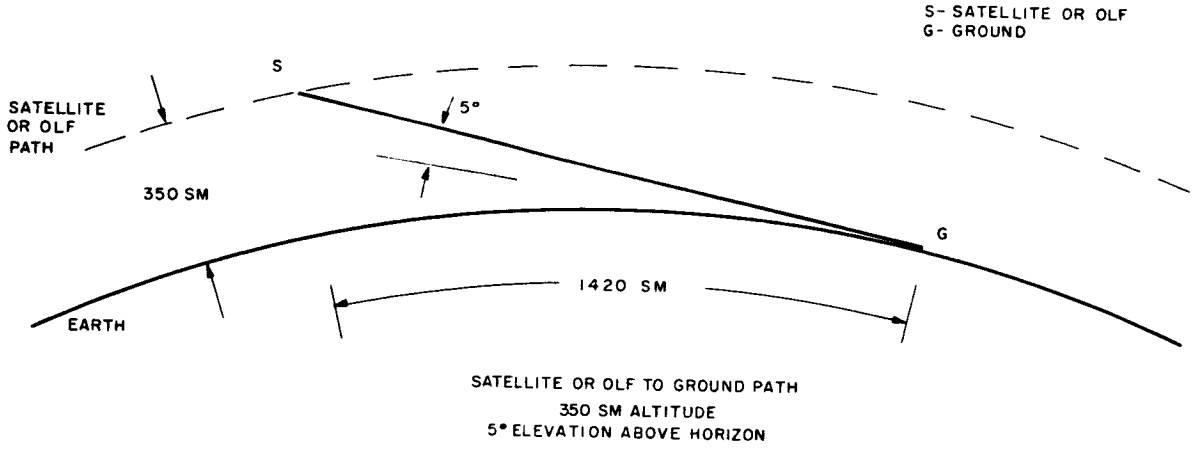
Station λ=Long.(Deg.) ϕ=Lat. (Deg.)	Orbit Period Number	Continuous Time in View (Minutes)														
		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
Cape Canaveral, Florida 80,48W	28,48N	9.1	9.1	7.3	-	-	-	-	-	-	-	-	-	-	7.8	9.1
Canary Island (near)* 15, 0W	28,48N	1.0	-	-	-	-	-	-	-	-	-	7.0	9	9.1	8.9	7.8
Ascension Island 14,41W	7,95S	-	8.7	8.0	-	-	-	6.2	8.6	5.5	-	-	-	-	-	-
Pretoria, South Africa 28,28E	25,75S	6.6	9.0	9.1	9.1	8.9	5.8	-	-	-	-	-	-	-	-	-
Mauritius Island (near)* 60E	20S	8.6	7.7	8.2	9.2	7.0	-	-	-	-	-	-	-	-	6.0	9.0
Woomera, Australia 136,79E	31,10S	6.0	-	-	-	-	-	-	-	-	-	-	7.0	8.7	9.2	8.6
Volcano Island (near)* 145E	27N	-	-	-	4.5	8.6	9.0	9.0	8.7	6.0	-	-	-	-	-	-
Hawaii 159,63W	22,16N	-	8.2	9.0	8.5	8.5	9.0	6.8	-	-	-	-	-	-	-	-
Society Island 150W	20S	-	-	-	-	-	-	-	7.3	9.0	8.3	7.6	8.5	9.0	3.8	-
Point Magu, California 119,15W	34,12N	6.3	8.3	8.3	7	-	-	-	-	-	-	-	-	-	-	-
TOTAL TIME IN VIEW		38	51	50	38	33	24	22	25	21	8	15	25	27	36	35
Continuous Time-out-of-view values (≥ 8 minutes)		11 15 11 20	11 26	14 25	36 15	44 13	46 21	37 30	14 29 30	15 28 31	88	55 26	37 11 23	35 10 23	18 12 13 12	16 11 27 8

TIV 24 HRS 30%

*Ship Stations

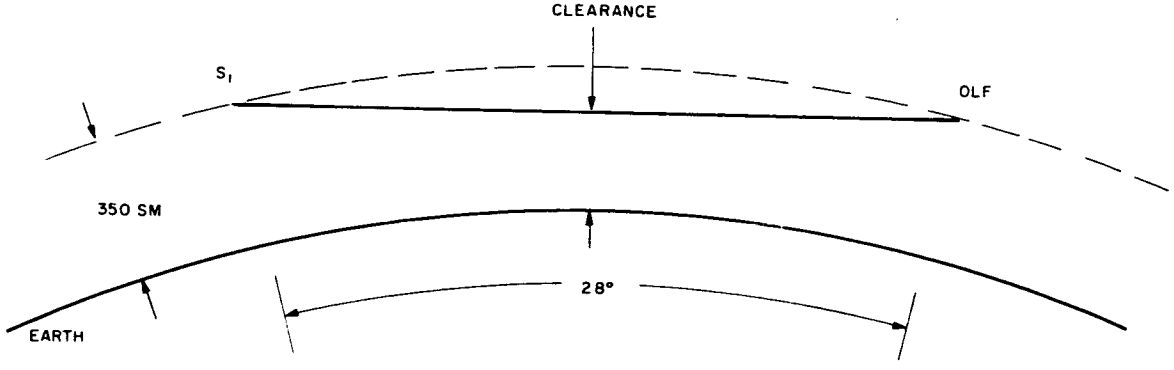
Figure 4-1

Earth-Based Tracking Coverage of OLF for a 24 Hour Period



SATELLITE OR OLF TO GROUND PATH
350 SM ALTITUDE
5° ELEVATION ABOVE HORIZON

A



OLF TO EQUATORIAL SATELLITE
PATH (OLF AT MAXIMUM LATITUDE)

S_1 - SATELLITE
OLF - ORBITAL LAUNCH
FACILITY

B

Figure 4-2. Maximum Line-of-Sight Path

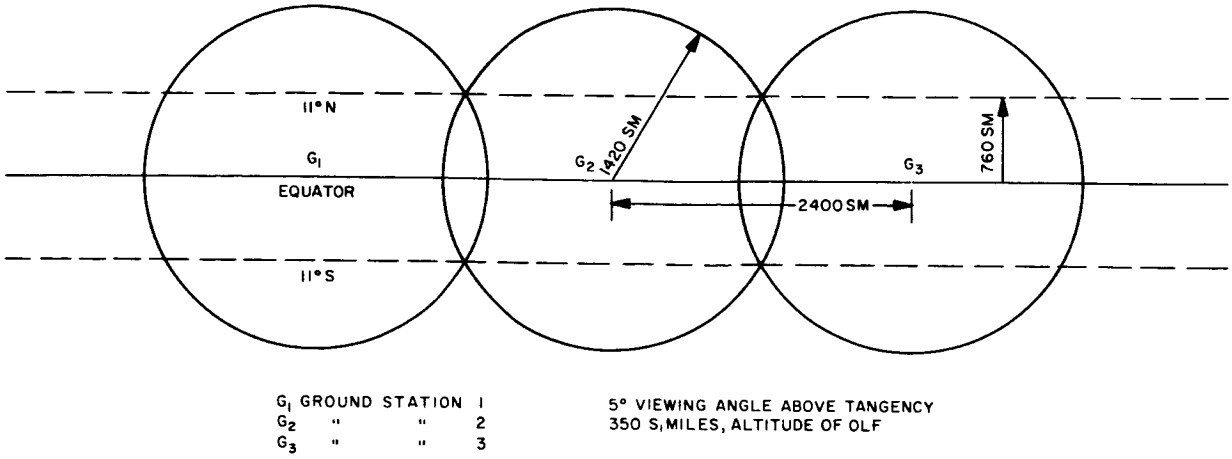


Figure 4-3. Equatorial Ground Coverage from OLF

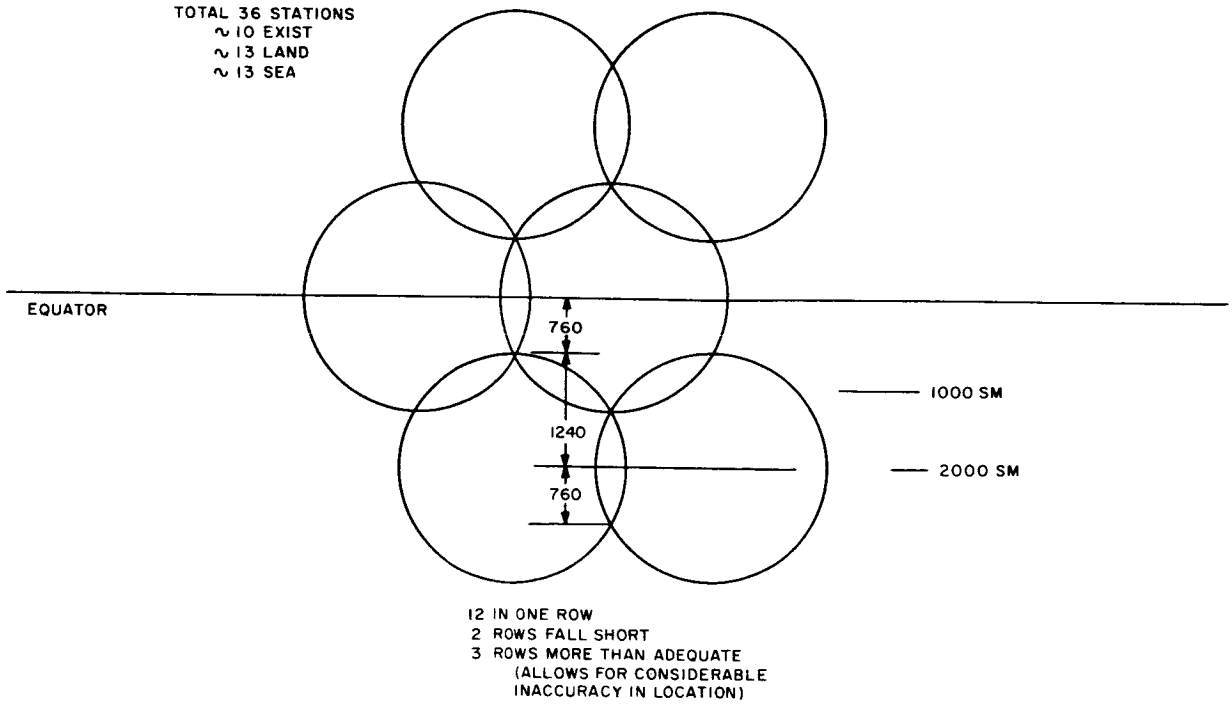


Figure 4-4. Pattern for Continuous Coverage

of-view for over 8 minutes. There are other times-out-of-view periods not shown that are less than 8 minutes. The total times-in-view for each of the fifteen orbits is shown in figure 4-1. For the locations chosen, path ten is the worst, where the time-in-view is 8.3 minutes and the time-out-of-view is eighty eight minutes. Subsequent orbits 16 through 30 do not exactly repeat but there are enough samples shown to give a representative picture of what may occur in any 24-hour period. The locations were selected with some care but even so are probably only representative, not optimum. Disregarding the distribution of in-view versus out-of-view, it has been found that during about 30 percent of the 24-hour period the OLF is in view of the 10 ground stations. From this it may be seen that many more stations are needed to affect continuous coverage. To see how many stations are required, refer to figures 4-2A, 4-3 and 4-4. Figure 4-2A shows that a 350 mile OLF can see a point 1420 miles away from the point directly under it, assuming 4/3 earth radius and minimum elevation 5 degrees above horizon. Figure 4-3 shows how individual coverages may be overlapped to form a continuous band about the equator. At 2400 statute miles between ground stations, cusps fall to $\pm$11 degrees of latitude from the line of centers. Figure 4-4 shows how two rows overlapped the same amount in both axes, as shown in figure 4-3, are inadequate for $\pm$28 degrees latitude coverage and how three rows are more than adequate. As a matter of fact, three rows allow for some inaccuracies in location. This shows that 36 stations uniformly spaced will be the maximum number required. This number gives an opportunity to estimate the costs of such a ground complex.

4-3. COST OF GROUND EXTENSION

Of these 36 stations, a very rough estimate allows for 10 existing stations, 13 on land, and 13 at sea. Considering that whatever economies there are in outfitting existing stations will be absorbed in the stations at sea, it can be seen that to an order of magnitude the cost of ten years' operation of a ground net of 4kc bandwidth is $115,000,000 and this estimate could be exceeded by a factor of 2 or more. Figure 4-5 shows this estimate. Although this estimate needs refinement, it shows that alternate schemes may be considered that cost on the order of hundreds of millions if they do a better over-all job, such as more bandwidth, fewer propagation outages, or more operating convenience.

	MILLIONS
36 STA at $1,000,000	$36
LEASED LINES (10 YRS) 36 STA 5,000,000 MIN at $5	$25
10 MEN/STA 10 YRS at $15,000/YR	$54
	$115 MILLION

Figure 4-5

Rough Estimate of Communications Ground Net Costs

4-4. USE OF SATELLITES

This leads to the consideration of the use of satellites. At first glance it appears that shared use of communication satellites would be very economical. Further study shows that this may not be practical except as a means to connect together ground stations. However, a few limiting

KIND	QTY	REMARKS
22300 SM	3	ABSOLUTE MIN QTY
350 SM	12	MIN QTY FOR COMM WITH FENCE OF 3 GND STA
350 TO 22300 SM	3<N<12	———
7000 SM	40-50	RANDOM

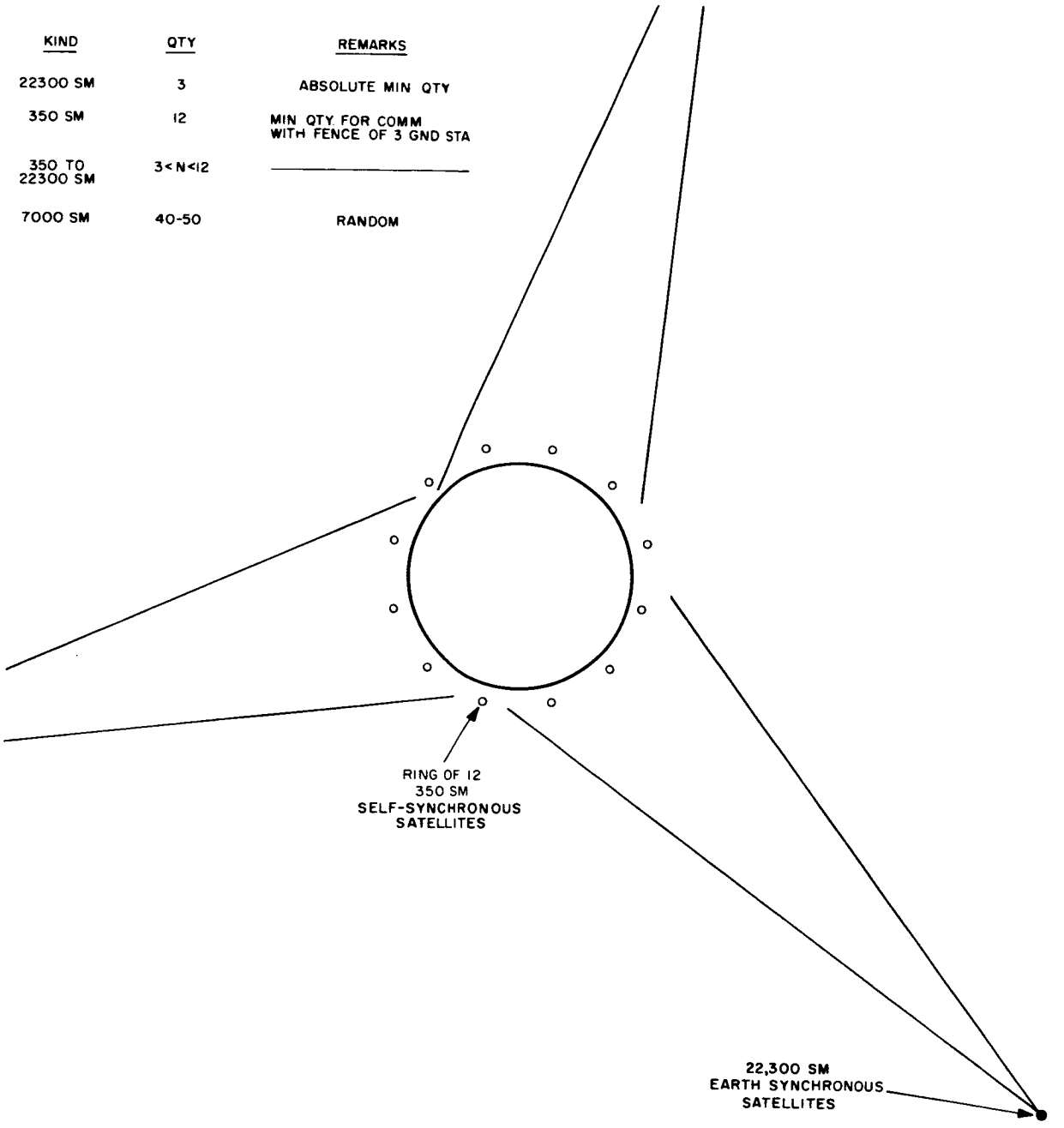


Figure 4-6. Types of Satellites

conditions may be understood from figure 4-6. The earth is well covered by the view from three high-flying satellites at 22,300 statute miles in stationary orbits. Without discussing the relative merits of high-flying satellites, momentary attention is given to other existing possibilites. A lower limit in altitude appears to be the height of the OLF orbit. As indicated in figure 4-3, such satellites would give a band 22 degrees wide around the earth. These must be mutually synchronous as must the 22,300 statute mile satellites. Two other possibilities immediately suggest themselves. Mutually synchronous orbits between 350 and 22,300 statute miles and random orbits, both polar and equatorial at various altitudes. Figure 4-7 gives a few comparative characteristics of three types of satellites. Intermediate synchronous satellites have been omitted for the sake of brevity. The line discussing random satellites has not been fully completed since a check with Bell Telephone Laboratories indicates that OLF use of random satellites would probably require shutting down a pair of ground stations that were originally designed to use the satellites. In addition, the acquisition and track column indicates that the random satellites requires considerable acquisition information to be transmitted to and stored in a computer in the "bird". Upon loss of this information, the system fails. Great difficulty is experienced in reestablishing to heal communications by search techniques. The many coordination difficulties associated with random satellite systems indicate that the emphasis should be placed on comparing low-flying satellites versus high-flying satellites.

4-5. COMPARISON OF 350 STATUTE MILE AND 22,300 STATUTE MILE SATELLITE SYSTEMS

In order to make some comparisons of these two satellite systems, firm ground rules must be established. The

Type	Qty	Pwr	Unit Mass Per Path	Nr of Terminals	Mass X Nr. Terms	Acquisition & Track From OLF	Mass Ratio	Rel. Mass Times Mass Ratio Adjusted
Synchronous 350SM	12	Low 1	1	12 + 1	13	Simple 1 Axis $\pm 15^{\circ}$ Max	1	13
Random 7000SM	~40	Med	> 1 < 121	~40	Inter-Mediate	Complex 2 Axis $\pm 90^{\circ}$	> 1 < 2.5	Complex
Synchronous	3	121	121	3 + 1	484	1 Axis $\pm 90^{\circ}$	2.5	1029

Figure 4-7

Comparison of Characteristics of Three Types of Satellites

single rule that can change the picture more than any other has already been discussed but should be re-emphasized, continuity of communications.

It makes little difference in the following discussion whether shared facilities or new parallel facilities are assumed, certain masses must be injected into orbit do do this job. Sharing is particularly difficult under present conditions where most satellite systems assume nonsymmetrical configurations with simple gear in the satellite and relatively sophisticated gear on the ground, weight and size being more or less directly related to sophistication. In communicating from one satellite to another, this nonsymmetrical system fails to bear dividends. Therefore, this analysis does not exactly parallel a ground-to-ground system. A full analysis should consider the cost of injecting the masses into the given orbits. A relative measure of this is given by the differences of the masses that must leave the ground. Assuming a ΔV of 7,000 ft/s to be required to go from a 350 statute mile orbit to 22,300 statute mile, it can be shown that with a solid fuel of $I_{sp} = 250$ the ratio of total mass to mass of payload plus structure is about 2.5.

Normalized weights will be computed with reference to the chart of figure 4-7. The least common denominator is the weight required to provide a given bandwidth at a given signal-to-noise ratio at the longest range in line 1. The maximum path length for line 1 is 2400 statute miles satellite to satellite (the 1600 miles slant range to ground does not govern). The weight of the equipment required to terminate one end of the 2400 statute mile path will be taken as unity. The 22,300 statute mile satellite has a maximum path length of its altitude plus one earth radius or 26,300 miles. Since the power required goes as the square of the distance,

the power for line 3 is $\left(\frac{26,300}{2400}\right)^2$ or 121. Since the weight may be considered to go directly as the power (for this power range and continuous power), the ratio of weights per path is 121 to 1. For line 3 there are 4 installation aloft, 3 satellites and the OLF at mass 121 against 12 satellites plus OLF at unity mass or 13. Accounting for the difference in altitudes and utilizing the mass ratio of 2.5 on the three 22,300 statute mile satellites gives the ratio 13 to 1029. This indicates that the low-flying satellite may be competitive even though many times heavier and more complex than the 22,300 statute mile satellite. The large advantage indicated is immediately reduced since each of the 350 statute mile satellites must be capable of looking in four directions. This yields (12 x 4) +1 or 49 versus 1029 or a ratio of 21. This is quite a large number and shows that further attention must be given the 350 statute mile satellites to determine what details of implementation are required.

4-6. NON-COPLANAR SATELLITE

The most attractive low-flying satellite is one at the same altitude as the OLF but in a non-coplanar equatorial orbit. A single satellite of this kind could aid the ground connection problem materially, since theoretically 12 ground stations at the equator would be in contact with it 100 percent of the time. Such a satellite could see the OLF at all times if maintained abreast of the OLF in an orbit of equal period, as shown in figure 4-2B. Such a concept may be extended to 12 satellites rather than 12 ground stations but this concept was rejected because of the difficulties in equatorial launching and maintenance.

4-7. COPLANAR SATELLITE

A derivative of the non-coplanar satellite system is a satellite in the path of the OLF. One leading and one lagging satellite spaced 30 degrees from the OLF, used as relays to each ground station, is an interesting concept. It triples the time in view over each station on the ground.

An obvious extension of this scheme is a limiting procedure. Extend these OLF relays to a ring of 12, one every 30 degrees. This reduces the number of ground stations required to a minimum of three placed in a north-south fence. Appendix D develops practical hardware for the configuration as depicted in figure 4-8. The OLF is shown at the upper right in contact with the satellite ring, numbered 1 to 12 and the ring in contact with the ground fence shown between 10 and 11. The ground station in contact with 1 is not part of the prime configuration, but gives direct contact to a given station, say the Cape, for certain intervals of time. A practical fence may include more than three stations. Figure 4-9 shows a fence using as many existing stations as possible, Antofagasta, Lima, Quito, Panama, Jamaica and the Cape.

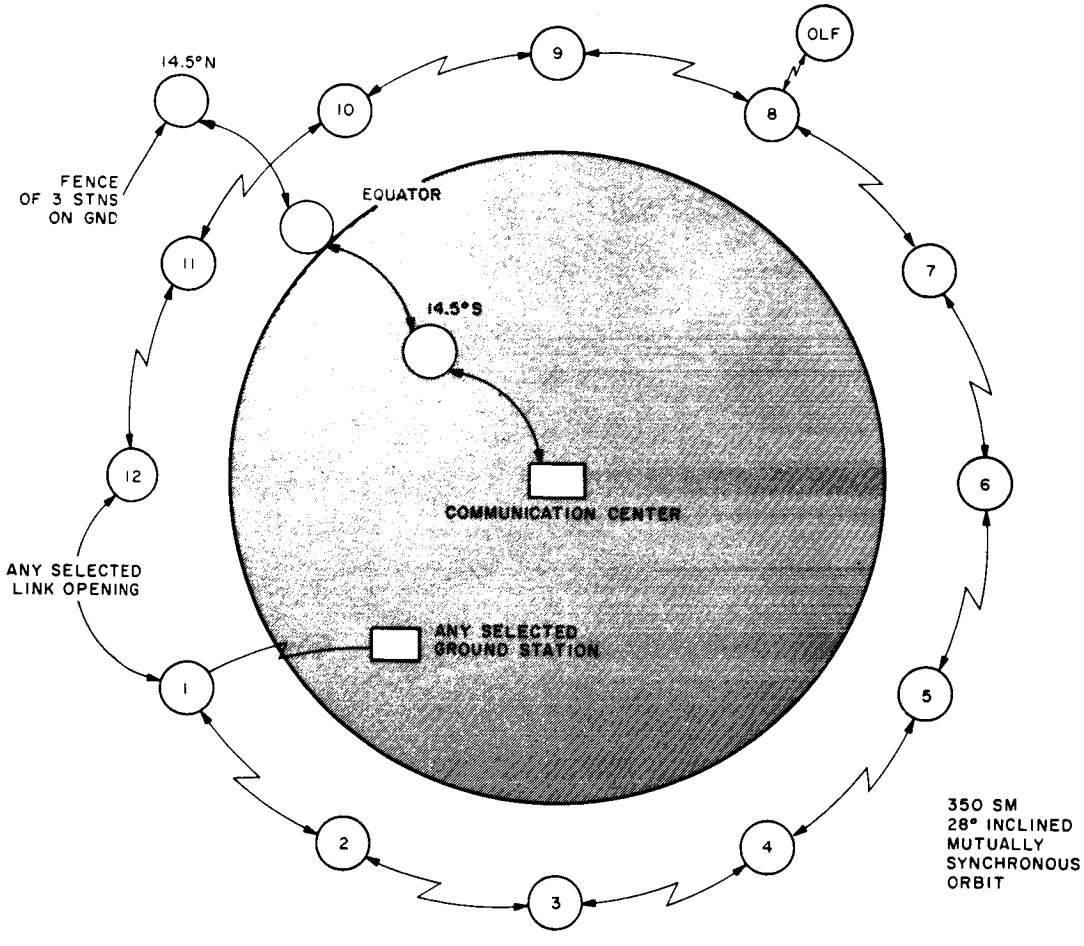


Figure 4-8. OLF Satellite Ring - Ground Fence Concept

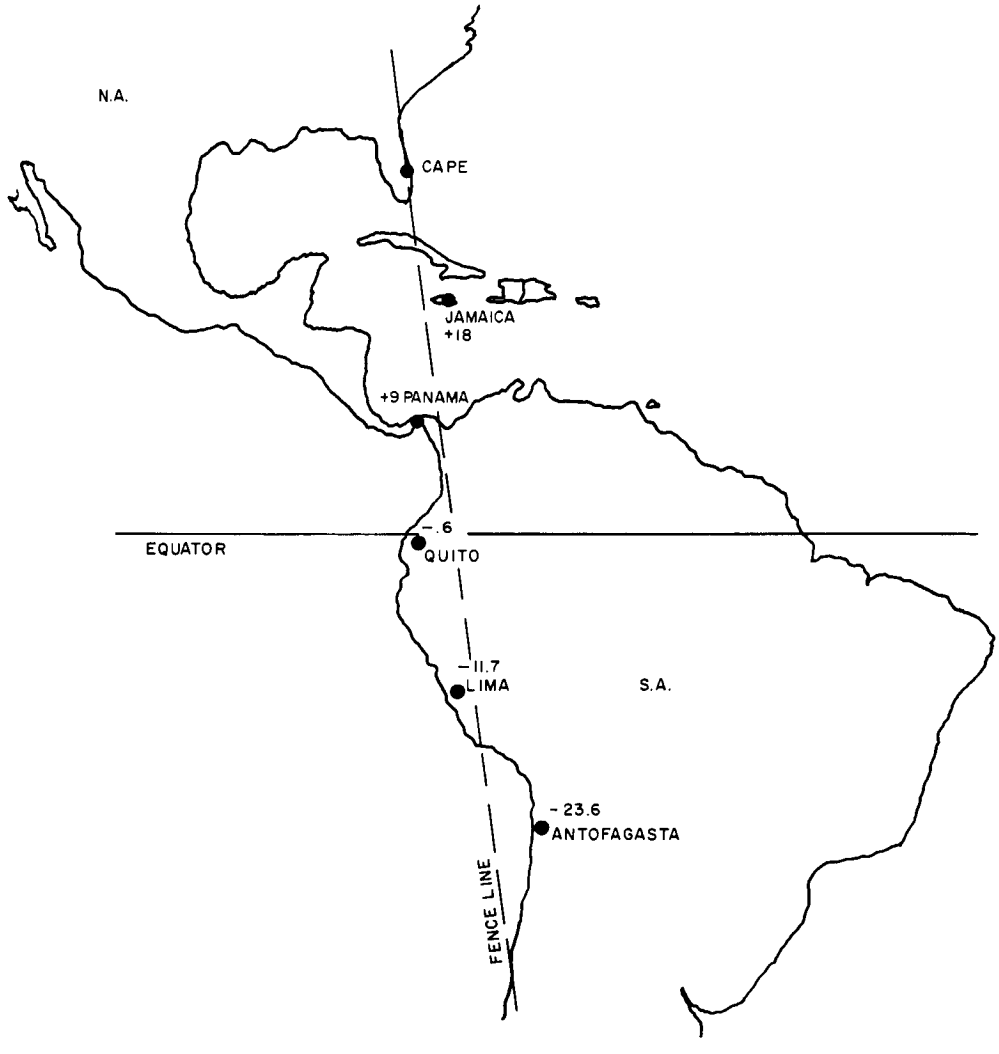


Figure 4-9. North-South Fence Geometry

SECTION V

CANDIDATE OLF TO GROUND SYSTEM

5-1. SATELLITE RING

Having established a possible means of continuous coverage through satellite relays, it is of interest to have a look at such a system in further detail. The communication system generated consists of a ring of 12 communication satellites coplanar with the OLF as shown in figure 4-8. These satellites are launched from the OLF. Each satellite is given a small boost of velocity of about 13.7 ft/sec with one satellite launched every three days. This spaces the satellites every 30 degrees around the earth and forms a ring which slowly rotates with respect to the OLF. Every 36 days each satellite comes sufficiently close to the OLF for retrieval and repair in the event of malfunction or for correction in the event of excessive deviation in position. The total weight of 12 satellites is on the order of 6,000 lbs which could be boosted with other equipment to the OLF. Directive antennas have been used on the satellites since the complication of orientation is considered well worth while in order to be able to achieve relatively wide bandwidth at a good signal-to-noise ratio.

5-2. GROUND FENCE

The ground fence is expanded to six stations each having UHF equipment looking up the ring and HF equipment

up the fence to the Cape as further described in Appendix D. The block diagram of figure 5-1 shows the equipment required in each of the 12 satellites of the ring. There are four sets of duplex radios in each satellite, one looking forward in the ring, one looking backward in the ring, one looking up to the OLF, and one looking down to the ground fence. All the equipment is in the 2300 mc band. Eight foot dishes are used for the paths around the ring, while a two foot dish is adequate up to the OLF and a slot antenna is adequate looking toward the ground. The one watt transmitters for transmission around the ring are operative all the time. The 4 watt transmitter looking to the OLF and the 2 watt transmitter looking to the ground operate on a 10 to 20 percent duty cycle. The calculations leading to the configuration of figure 5-1 are detailed in the Appendix E. The OLF equipment for connection into the satellite ring is shown at the top of figure 5-2. This shows an eight watt transmitter and a parametric receiver operating with an eight foot dish. Calculation of this circuit is also given in Appendix E. The remaining three circuits of figure 5-2 are discussed in Section VI.

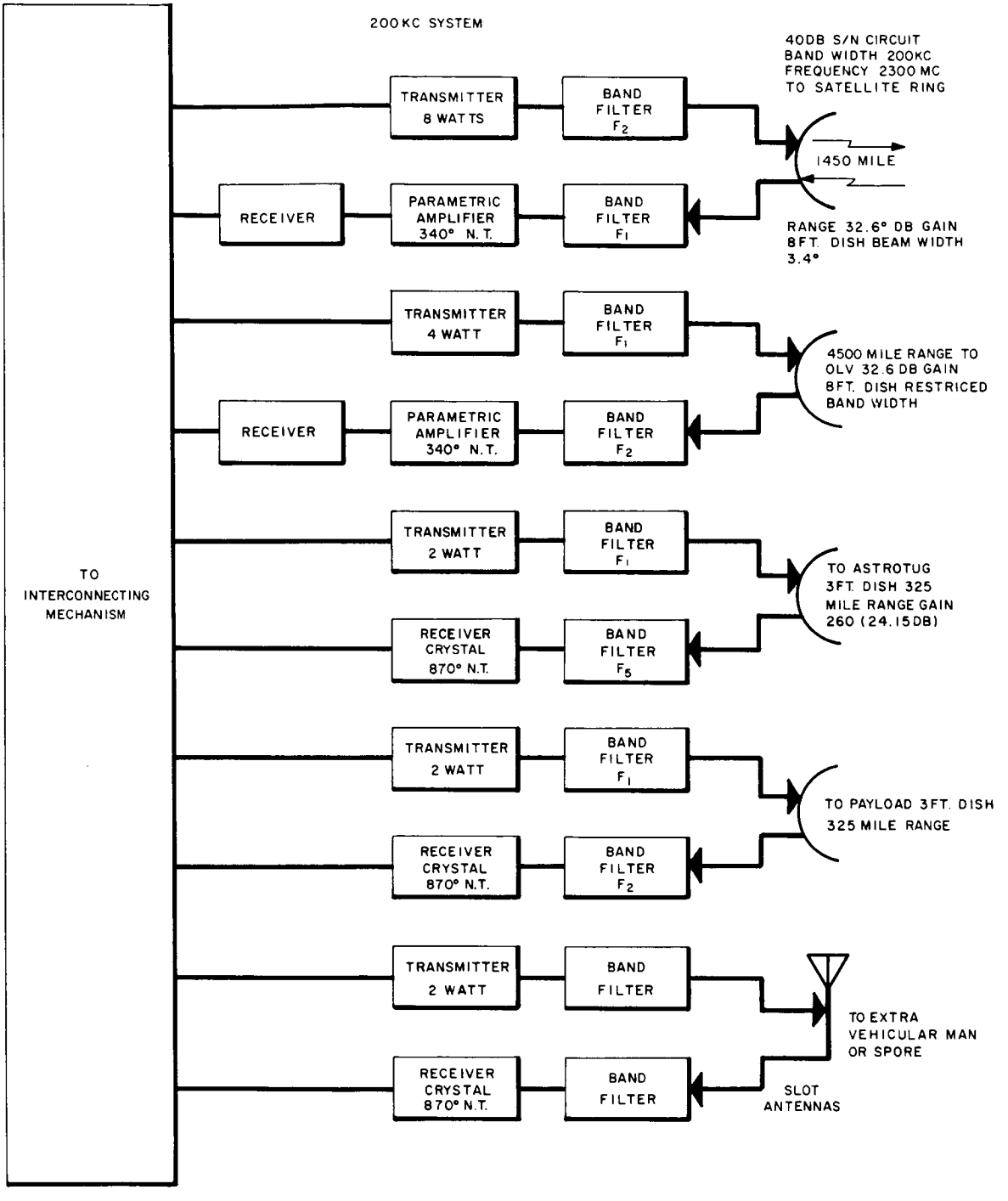


Figure 5-2. OLF Equipment

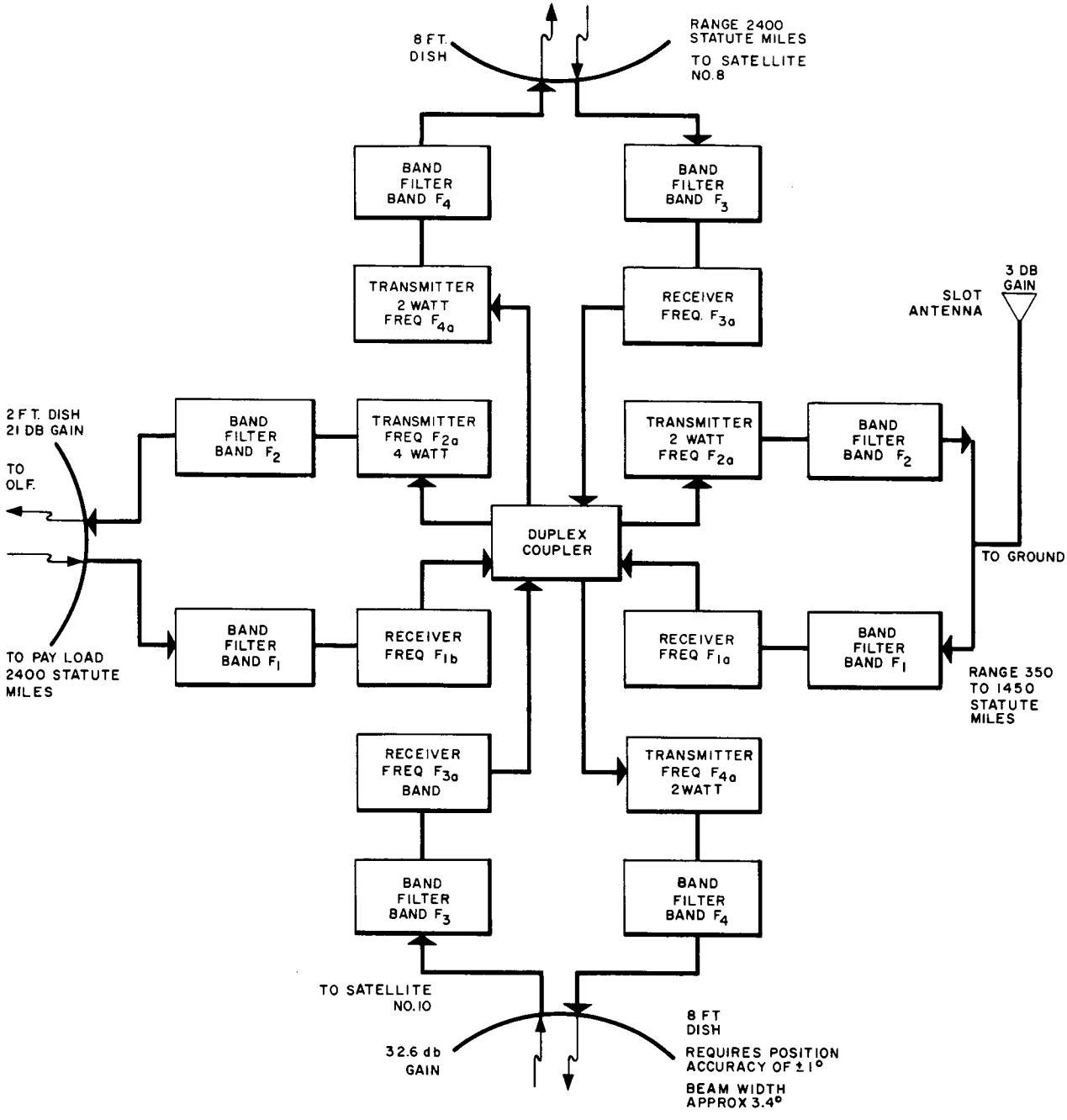


Figure 5-1. Typical Space Relay Satellite

SECTION VI

OLF R-F EQUIPMENT

6-1. R-F EQUIPMENT

The OLF equipment has been summarized in figure 5-2. Supporting calculations are given in Appendix E. All of this equipment is in the 2300 mc band and requires two 8-foot dishes and two 3-foot dishes. Although every attempt should be made to keep the powers and receiver noise temperatures the same in order to increase the flexibility and built-in redundancy (at reduced capability) one eight-watt transmitter, one four-watt transmitter and a pair of two-watt transmitters have been selected. Antenna and frequency switching should be provided, however, so that even the less sophisticated equipment could be placed on the most important circuit with somewhat reduce performance. In addition, the modules of these equipments should be the same, for direct interchange in the case of failures.

Consistent with the work statement of Section II only the satellite ring ground, and OLF ends of the circuits have been detailed. In general, the other ends of the circuits are symmetrical with those detailed in the following paragraphs. The one exception to this is that the spacecraft transmitter power is determined by lunar distance down to the DSIF station and is on the order of 25 watts rather than the 4 watt transmitter shown in the OLF.

6-2. CIRCUIT TO OLV

The circuit to the OLV is similar to that of the satellite ring except that not quite so much power is required for greater range since symmetry has been observed with the OLV, whereas in the case of the satellite ring a non-symmetrical circuit was utilized to keep the 12 satellites simpler and lighter. The range of 4,500 statute miles was selected to avoid the shadow area that might occur before the OLV would be high enough to be received by the DSIF net. The geometry of this consideration is shown in figure 6-1. This figure shows the earth, the path of the OLF at 350 statute miles altitude, the path of the OLV up to 6000 statute miles altitude, and the near triangular cross-sections of shadow volumes that are below the horizon of the DSIF stations. Letters on the two paths, a to q, show the relative times so that relative look angles may be estimated. For simplicity the DSIF stations have been shown in the plane of the paths spaced 120 degrees apart. This approximation is close enough for the purposes of this discussion. The figure shows the DSIF shadow area covering the first 2000 miles of the path. This has no particular significance since the injection point might have any position relative to the DSIF stations. It may be seen that the DSIF shadow areas could be coincident with the first 4000 miles of the OLV's path. To eliminate the possibility of this problem, the OLV may maintain communications with the OLF for the first 4500 statute miles of altitude and at this point transfer control to the DSIF net. This approach presents several advantages in that it reduces the required OLV communication payload and the required capabilities of the NASA ground complex while allowing a single station to maintain control of the OLV during initial powered flight.

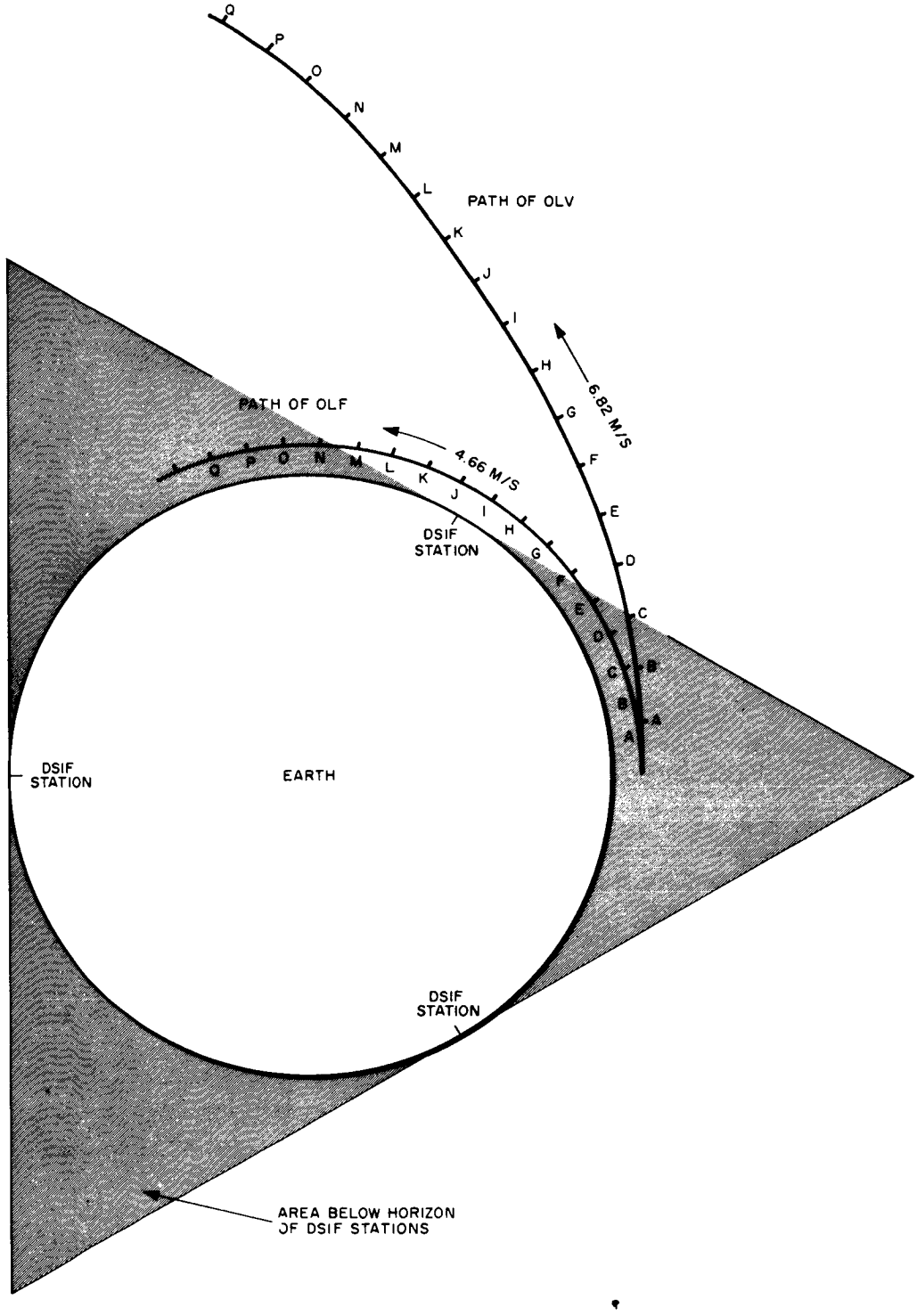


Figure 6-1. Path of OLV Relative to Path of OLF for Altitudes Below 6000 Statute Miles

6-3. CIRCUITS TO ASTROTUG AND PAYLOADS

The circuits to the Astrotug and payloads are identical and both are similar to the longer haul circuits. The range of 350 statute miles is a compromise between simplicity of equipment and early continuous communications for rendezvousing "birds" in the vicinity of the OLF. These might not be simultaneously required, particularly that to the OLV, in which case the unused circuit should be considered as a spare. Again this points out that frequency agility and input/output switching are a firm requirement.

6-4. CIRCUITS TO THE EXTRA-VEHICULAR MAN

The communications to the extra-vehicular man may be handled primarily by telephone. It is assumed that there are safety lines to each of these men much like the man in the diving suit under water. However, for emergency, or in the case where he wishes to free himself of the hinderance of such a line, radio should be used. This should not be in the low frequency bands, VLF through VHF, since very strong interfering signals may be received from the ground only 350 statute miles directly underneath. For this reason, the UHF range has been selected once again. The fact that duplex operation may be obtained more easily at the higher frequency (since sharp filters are smaller in size) and compatibility with the rest of the system makes one more interconnection capability without added equipment if time sharing is allowed when other facets of the operation are shut down. A 10-milliwatt transmitter and a 30-db receiver is more than adequate for the job. The OLF equipment is the same as that for communications with the Astrotug or

or payload. A simple multiplexer can allow simple bio-medical data to be run on this circuit continuously if required, since the duplex capability is available. Duplex is also very important for the reason stated much earlier, that of prevention of unintentional monopolization of a reversible simplex circuit.

SECTION VII

MODIFICATIONS OF SATELLITE RING FOR CONTINUOUS TRACKING COMMUNICATIONS

As was mentioned earlier, the study so far has separated the continuous communications problem from the continuous tracking problem. It is interesting to consider what use may be made of the satellite concept as proposed herein to send radar tracking data back to the central computer. Assume that a considerable number of radar stations, widely dispersed throughout the world, are made available to the project. These may be ships, or added land-based stations in order to get the necessary coverage. These stations individually have a limited use factor since, even if they are at the proper longitude, only less than one-third of the OLF passes will be visible due to the changes in latitude of the OLF. Even though the satellite ring proposed is also only in view at any selected spot on the earth for the same percentage of the time, this portion of the time brackets the pass, or several passes of the OLF. This means that the satellite ring may provide a path which does not provide continuous communications to any given ground station back to the Cape, but is available as much as an hour or more before and after the given OLF pass. This is exactly when the data is generated that must be handled in real time or near real time. Some of the available bandwidth of the ring could be diverted for this job. The information would be available for both the operations center and the OLF, if the OLF were to have

sufficient computing capability to absorb the information. The total bandwidth can be divided with rather simple base-band filters since there would never be more than one station, ground or OLF, simultaneously entering a single receiver in the ring.

SECTION VIII

OLF COMMUNICATIONS AND TELEMETRY TERMINAL EQUIPMENT

8-1. GENERAL

The OLF communications and telemetry terminal equipment is that compliment of equipment required to gather and/or display, process, and transmit and/or receive all necessary OLF information. This information will include voice, data command signal, and control data. The first step in evolving the terminal equipment is to first establish system requirements and then mechanize the equipment to fit these requirements. In this section a summary of the requirements is outlined along with the resultant system. A detailed discussion of both the requirements and equipment is given in Appendix F.

8-2. DATA HANDLING REQUIREMENTS

The OLF terminal equipment must have the following data handling capabilities.

a. Monitor and transfer to ground OLF equipment performance measurements.

b. Monitor and transfer to ground data reflecting the internal and external environments of the OLF.

c. Gather and relay to ground the performance of man and his bio-medical condition while aboard the OLF.

d. Collect, store and relay to ground the results of OLF scientific experiment.

e. Two-way transfer of checkout and countdown data between the OLF and the OLV and between the ground and the OLF.

f. Two-way relay of all OLV-Ground data.

g. Receive, transmit or relay voice between all orbiting vehicles and ground.

h. Transmit to ground single frame video images from OLF.

8-3. FUNCTIONAL EQUIPMENT

To meet the data handling requirements the OLF must incorporate the following functional equipment.

a. Data sensors

b. Data processing and format conversion equipment

c. Data storage units

d. Data amplitude and time reference generators

e. Data program routing and controller

f. RF transmission and receiving devices.

8-4. OPERATIONAL REQUIREMENTS

The communications and telemetry terminal equipment must be maintainable; have a high degree of reliability; be compatible with other OLD equipment; require minimum power, weight and volume, and require a minimum amount of attention from man.

8-5. EQUIPMENT MECHANIZATION

A functional block diagram of the CLF communication and telemetry terminal equipment is shown in figure 8-1. The system has the following characteristics.

a. Electrical

Information channels	375 inputs
Information types input	Analog or Digital
Input signal range	0 to 5 volts
	0 to 50 millivolts
Accuracy	0.1% FS (high range)
	0.4% FS (low range)
Max. sampling rate	8 KC
Max. encoding date	80 Kbits
Max. bits per word	10
Storage	Magnetic
Operating modes	Programmable
Power	417 watts

b. Mechanical

Weight	383 pounds
Volume	7.34 ft^3

8-6. TERMINAL SYSTEM FEATURES

The OLF terminal equipment is oriented and optimumized for the OLF program. Some of the special features of this system are outlined in the following paragraphs.

a. Programmability. The system will feature an extremely flexible program to schieve maximum utilization of power, bandwidth and spectrum. The system will include predetermined sampling plans which will have a discrete number of individual channel sampling rates on any frame to make the rate consistent with the signal information content. In addition, the proper degree of accuracy is obtained through the use of a high-low level encoder which reduces the transmission of redundant data. Also, programmable baseband frequency allocation will be used for the simultaneous relay of different data types which will result in the maximum utilization of a given bandwidth. Programs will be selected on the basis of signal information content and upon data priority and can be controlled either on a fixed time basis or by remote command from the ground or the OLF operator.

b. Self-Check Features. The system will periodically transmit to the ground selected signals which will show the operating condition of the system. In addition, the system will accept upon command a set of self-check signals from the ground, appropriately route these test signals through

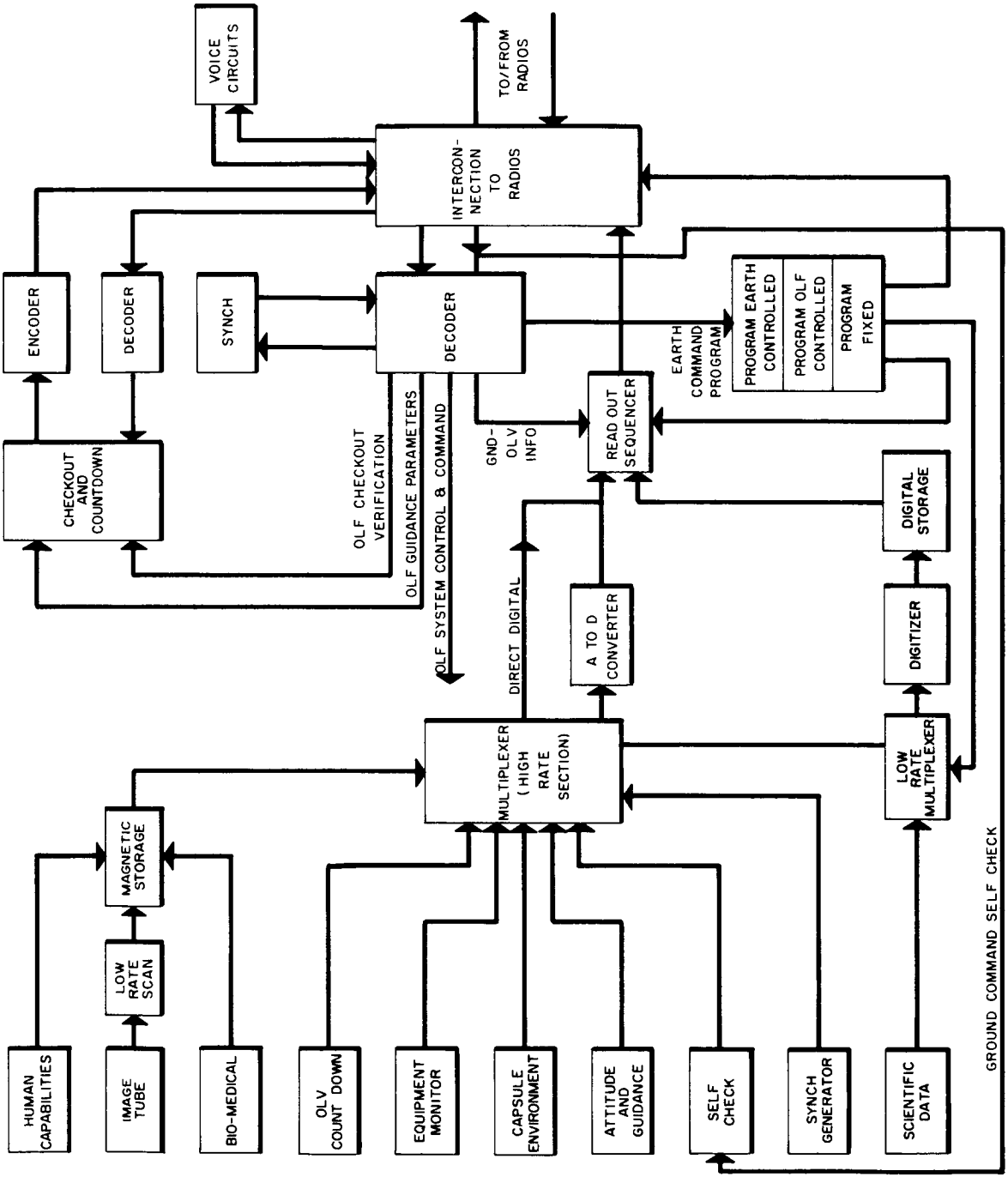


Figure 8-1. PCM System

the system and return them to the ground where they can be interpreted in terms of the operating status of each replacable subassembly.

c. Additional Features. Other program oriented features include high reliability, replaceable subassemblies which utilize micro-miniature circuits, digital and magnetic storage and flexibility for expansion or program modifications.

SECTION IX
CONCLUSIONS

9-1. OTHER SYSTEMS

Before discussing the candidate system a comparison is made of several systems, as shown in figure 9-1, in which the candidate system appears in the first column. A 36 ground station concept and an advent concept are shown in this comparison.

9-2. CANDIDATE SYSTEM

Four major aspects of the candidate system were discussed in Sections V, VI and VIII, separately. These were the satellite ring and the fence which provided continuity of communication and telemetry from the ground operation center to the OLF. Two additional parts discussed, were the radio frequency equipment in the OLF and its immediate neighbors and the telemetry terminal equipment in the OLF. These four parts are now reviewed as a complete system. Figure 9-2 illustrates the candidate system including all of these four parts. Pictorially, the telemetry terminal portion is not evident. The tabular data of figure 9-3 has been provided to summarize the major equipments included in the fence satellites and the OLF. This data forms a basis from which the hardware cost estimates were made. Study, development, test, training and spare parts have not

Figure 9-1

Comparison of Three Basic Concepts For Continuous Communication

Item	12 Satellite Repeater Concept	36 Ground Station Concept	Advent Concept (3 equally spaced satellites)
Relay Equipment - to provide continuous contact between OLF and a designated point on earth	12 satellite repeaters located around the world at approx. OLF altitude.	A net of about 36 ground stations located accurately around the world.	Use Advent Facilities-both in space and on ground
Bandwidth Limit	75 kc proposed-could be several megacycles	About 3 kc - for existing world network	Might be in megacycle range if channel is available
Cost	About $75 million - not including booster costs for satellite repeaters	Considerably more than $115 million including operating cost	Total cost unknown. High Power Transmitter and Large Antenna needed on OLF for tracking Advent Satellites - provides difficult OLF stabilization problem
Availability	2 new fence stations plus additional equipment in 4 available stations.	About 25% of required ground stations available.	Doubtful - if available on full time basis

Figure 9-1 (Cont)

Comparison of Three Basic Concepts For Continuous Communication

Item	12 Satellites Repeater Concept	36 Ground Station Concept	Advent Concept (3 equally spaced satellites)
Other Considera-tions	Would be available for other special communica-tion requirements be-tween earth and space vehicles located within the near vicinity of repeater.	Because of track-ing requirements at each station it is doubtful that stations could be unmanned.	Should be investigated more thoroughly.

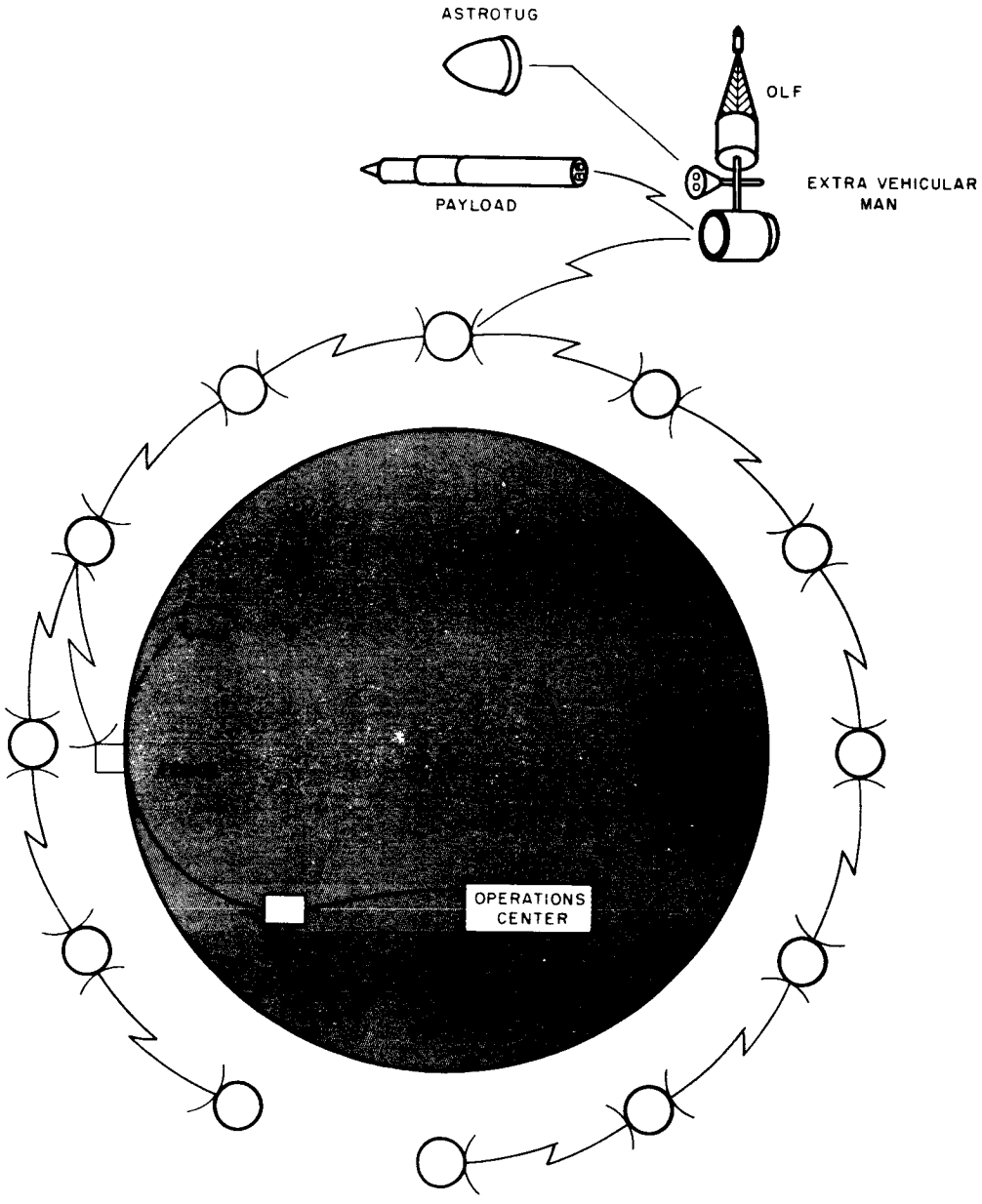


Figure 9-2. Candidate System

Figure 9-3

Table of System Equipment

Station	Item	Qty/Stn	Num of Stns	Total Qty
Fence	2300 mc diplexer	1	6	6
	Tracking 48 DB norm ant	1	6	6
	2300 mc rec, NT 200°K	1	6	6
	2300 mc trans, 10-watts	1	6	6
	Data term. - UHF	1	6	6
	2-30 mc Log. per. antenna	4	5*	20
	2-30 mc trans. filters	6	5	30
	2-30mc power amps, 10-kw	6	5	30
	2-30 mc rec., NF 15 db	6	5	30
	2-30 mc exciter	6	5	30
	data term., HF	6	5	10
	2-30 mc preamp.	2	5	10
	Building & power	1	6	6
Satellite	8 ft dish ant.	2	12	24
	Filter, 2300 mc, F_3	2	12	24
	Filter, 2300 mc, F_4	2	12	24
	2 ft dish ant.	1	12	12
	Filter, 2300 mc, F_1	2	12	12

*4 + 1/2 + 1/2 = relays + 2 ends

Figure 9-3 (Cont)

Table of System Equipment

Station	Item	Qty/Stn	Num of Stns	Total Qty
Satellite (Cont)	Filter, 2300 mc, F_2	2	12	12
	Trans, 2300 mc, 1-watt	2	12	24
	Rec., 2300 mc, NT. 870°K	4	12	48
	Trans, 2300 mc, 2-watt	1	12	12
	Trans, 2300 mc, 4-watt	1	12	12
	Base band coupler	1	12	12
	Control eqpt.	1	12	12
	Frame	1	12	12
	Power supply Solar & battery	1	12	12
OLF	8 ft dish	2	1	2
	3 ft dish	2	1	2
	Trans. 2300 mc 8-watts	1	1	1
	Trans. 2300 mc 4-watts	1	1	1
	Trans. 2300 mc 2-watts	3	1	3
	Baseband inter-connections	1	1	1
	Internal communications	1	1	1

Figure 9-3 (Cont)

Table of System Equipment

Station	Item	Qty/Stn	Num of Stns	Total Qty
OLF (Cont)	Filter, 2300 mc	8	1	8
	Rec, 2300 mc Blt. 870°K	5	1	5
	Preamp., 2300 mc NT 340°K	2	1	2
	Telephone to extra-vehicular man	3	1	3
	Telemetry Terminal	1	1	1
	Slot antenna	6	1	6
Extra Vehicular Man	Telephone	1	3	3
	Rec. 2300 mc NF 30 DB	1	3	3
	Trans. 2300 mc 10 nw	1	3	3
Payload or Astrotug (Not Priced under OLF Package)	3 ft dish	1	X	1X
	Band filter	2	X	2X
	Trans, 2300 mc 2-watt	1	X	1X
	Rec, 2300 mc NT 870°K	1	X	1X
	Telemetry Terminal, simplified	1	X	1X

Figure 9-3 (Cont)

Table of System Equipment

Station	Item	Qty/Stn	Num of Stns	Total Qty
OLV (Not Priced under OLF Package)	8 ft dish	1	1	1
	Band filter	2	1	1
	Trans, 2300 mc 25 watt	1	1	1
	Rec, 2300 mc NT 870°K	1	1	1
	Preamp., 2300 mc NT 340°K	1	1	1
	Telemetry terminal	1	1	1

been included in the listing although they form elements of the total cost. These elements as well as the hardware have been called out in the program schedule of figure 9-4. This schedule basically covers the period from the beginning of the calendar year 1962 through the calendar year 1967. Although intermediate and ultimate OLF schedule is shown for reference purposes, none of the costs have been projected beyond 1967. It is important to note that the period of time from January 1962 to the end of the first quarter of 1965 is a very short time in which to provide thoroughly qualified hardware. The success of the program depends upon having prototypes in the latter part of 1963 in order to properly qualify equipment by the end of 1964. In order to achieve prototype units in the last half of 1963 extensive study programs must be accomplished during the first half of 1962. The number of hardware units have been indicated directly on the hardware schedule.

The costs are presented in graphic form in figure 9-5. The lower curve shows projected equipment and operating costs of the ground fence for the years 1962 through 1967 totaling 20.0 million dollars. The next higher curve shows the initial and equipment costs satellite ring totaling 26.3 million dollars. This figure does not include the boost cost in that consistent information was not available. The next higher curve shows the OLF communication and telemetry system equipment totaling 30 million dollars. The top curve is the sum of the lower three curves which results in a grand total of 76.3 million dollars for the period 1962-1967.

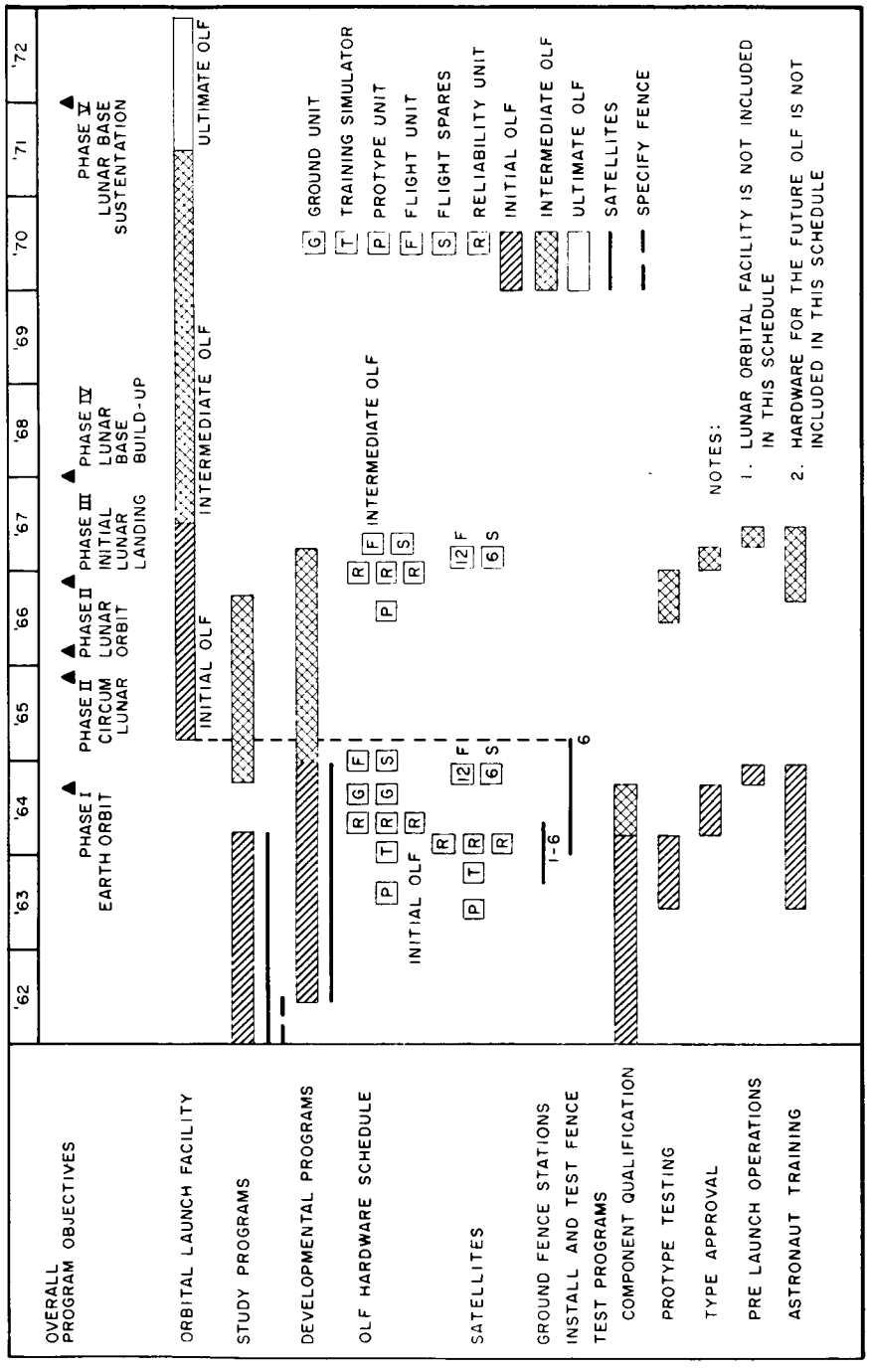


Figure 9-4. Program Schedule - Telemetry and Communications

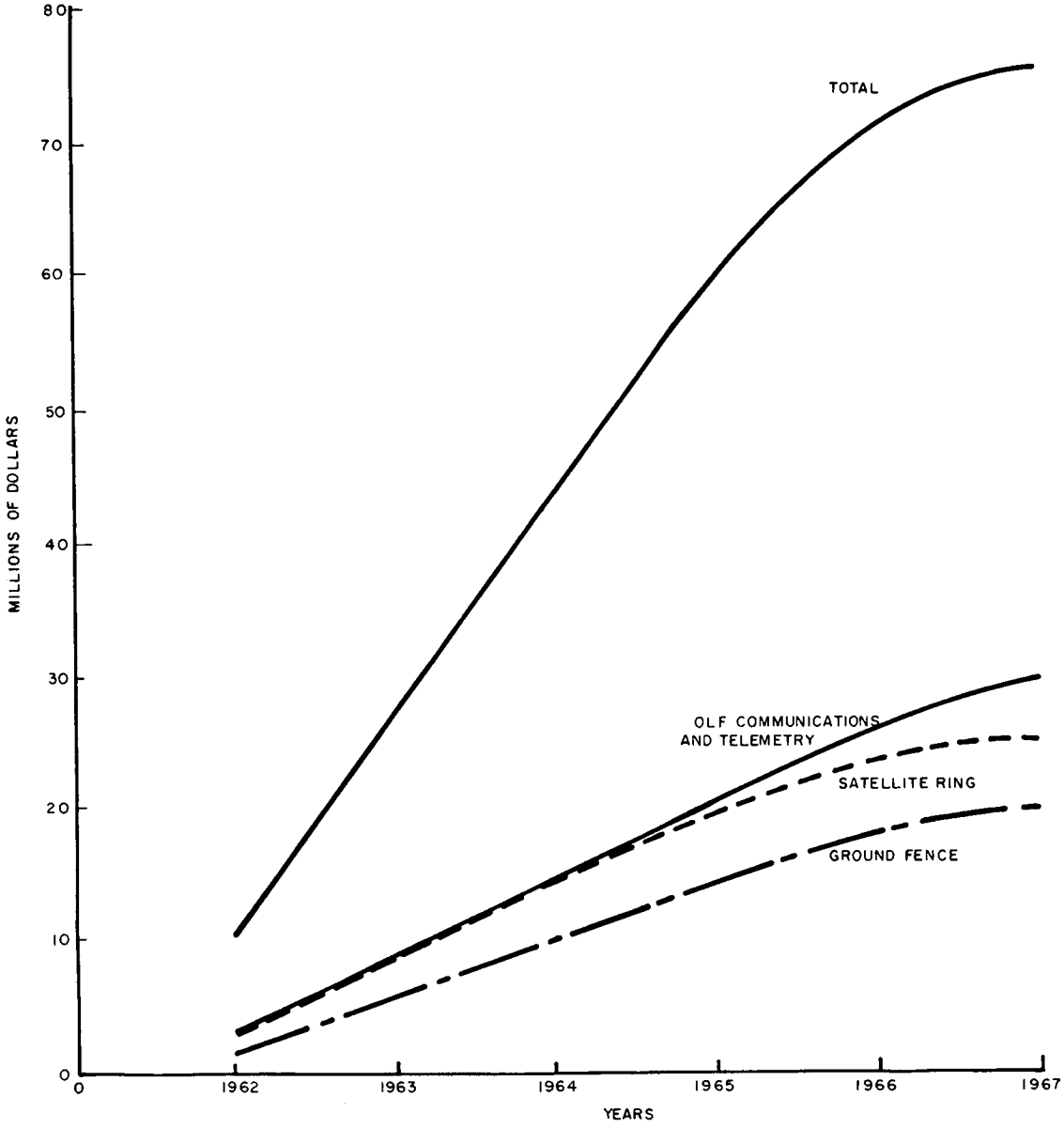


Figure 9-5. Projected Costs

9-3. AREAS REQUIRING FURTHER STUDY

In reading through this volume there are several areas which only point out a direction of investigation and it is understood that the answers to certain questions are still to be determined.

This volume has been based on the assumption that a space station, the OLF, will be constructed and exist for a considerable period of time. This space station may be associated with several programs, lunar or planetary, over a period of years. Although for certain lunar operations such a station might be reduced to merely a rendezvousing operation and be self expended, it is reasonable to assume that in the long run an enduring facility will be constructed. The question is then resolved to a revision of time scale, rather than a total cancellation, if the early rendezvousing leaves no residual permanent base. This is not considered as an area of concern for the existing or follow-on studies. On the other hand, the question as to whether or not the communications to the OLF need to continuous or can be intermittent is of primary interest to the operational success. There are many conflicting factors, some technical, some philosophical, some political that enter into this question. This is one of the areas definitely requiring further study.

A forthright approach has been taken to the problem of achieving continuous communications by the use of low-flying satellites. All of the problems in the use of low-flying satellites for this purpose have not been solved. The process of evolving a set of specifications for such satellites requires clarification of many details which have been only briefly investigated herein. Such a process of reducing the requirements to a specification must be entered into very promptly in order to meet the operations time scale.

The achieving of continuous communications was singled out and emphasized as a separate item since it is difficult to achieve by other than rather advanced concepts. At the same time that specifications are being established for satellite communications the rest of the system in the OLF and on the ground must be specified. Both of these efforts must lead to early models at least a year in advance of fly dates in order to achieve the necessary reliability.

This introduces another area of paramount importance, reliability, path redundancy and maintenance concepts. Path redundancy and maintenance concepts were discussed in Section V, paragraph 5-1 and Appendix C under net management concepts. The candidate system has appeared in one frequency band in order to satisfy some of the requirements for interchangeability and alternate routing. Further consideration must be given to the basic reliability of the assemblies as well as the way in which the size or function of the assemblies are selected. It is recommended that in the process of writing specifications for the various parts of the system that freedom be given to the preliminary specifications. Then a review should be made to establish what common elements and modules may be distilled from the preliminary design and used as synthesizing elements for the final specifications. Details of the items to be checked in such a review are given in Volume VI, Appendix D. It is further recommended that breadboards of the resultant assemblies and subassemblies be reliability tested at as early a date as possible.

No recognition has been made of possible jamming problems in the candidate system. Since this is not a military project it was not entered in the basic assumptions. However, the area cannot be totally avoided since unintentional interference could be a problem. Further study is required to

determine whether a frequency above the atmospheric window might be recommended for the satellite ring, or as a minimum what protection must be given any supervisory signals carried on the circuits.

Other areas requiring further attention will be brought forth in subsequent studies and discussions with individuals of the program. The attempt has been made to include the most important of these so early planning can arrange for filling in the gaps.

APPENDIX A

BASIC NETWORK GEOMETRY

The Orbital Launch Operations communication system has many mobile station requirements. The following over-simplification of these requirements is presented to emphasize the elements of the geometry of the most difficult of the communication problems, that of continuity. After the discussion of the five geometric concepts, each concept will be related to the OLO operation.

In figure A-1 a disc rotating at angular velocity ω_1, is shown for consideration. Located on this disc at a discrete location, is user B. External to the disc is user A, fixed with respect to the inertial reference. User A is tied to the disc with a brush and user B is tied to the periphery of the disc with a fix connection. Figure A-2 shows essentially the same concept as figure A-1 except that the periphery of the disc has been segmented, and user B is connected to each segment. Figure A-3 is similar to figure A-1 except that the brush and the fixed connection have been interchanged and the motion ω_1 is confined to user A while user B is stationary. It is true that the periphery of the disc has been disassociated from the disc and becomes a ring rotating with user A at rate ω_1.

The concept of figure A-3 may be extended to include other users at angular velocity ω_1, for instance user C, as shown in figure A-4. This concept may be further extended to include user C at a different angular velocity, ω_2, from that of user A and the ring as shown in figure A-5.

Figures A-1 and A-2 represent a ground station at user B and a mobile space borne station, the OLF, at termination A. The difference between figure A-1 and figure A-2 is the segmenting of the periphery. Figure A-2 represents the present method of a multitude of ground stations, one for each segment connected back to the central control point by radiating connections. Figure A-3 anticipates the possibility of generating a system where the segments are tied together peripherally and rotate with termination A, the OLF. Figure A-4 extends this to include additional space borne users as illustrated by user C. Figure A-5 further extends the concept to include additional space borne users at angular velocities different from that of the ring.

Figure A-1. Geometric Concept Disc Rotating at Angular Velocity ω_1

Figure A-2. Geometric Concept Segmented Disc

Figure A-3. Geometric Concept Interchanged Brush and Fixed Connections

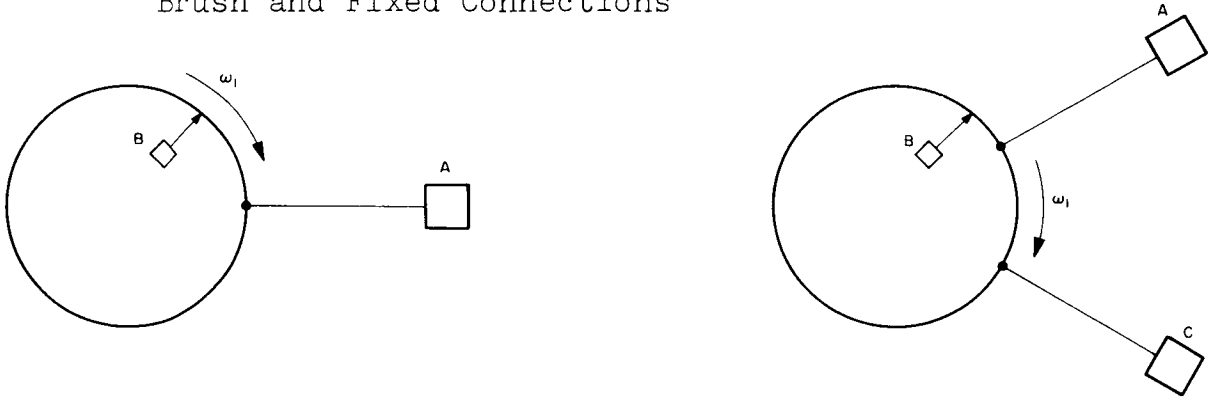


Figure A-4. Geometric Concept Addition of User C

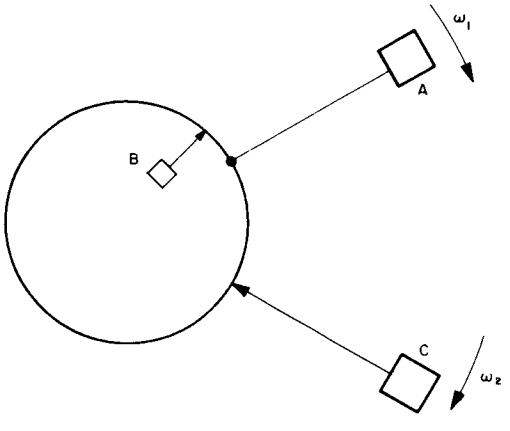


Figure A-5. Geometric Concept Addition of User C at Different Angular Velocity

APPENDIX B

TYPES OF CHANNELS

Two types of communication channels are extensively used in our communication systems today. The first type of communication channel is a simplex channel and the second channel type is a duplex channel.

The simplex type of communication allows traffic to flow only in one direction, between the source and the receptor, at any given time. Since traffic flow exists in only one direction at any given time, supervision of the traffic flow is difficult unless separate supervision channels are provided. For example, if a source is transmitting information, there is no assurance, over a simplex system, that the receptor has received and understood all of the information until the end of transmission when the direction of transmission is reversed and the receptor is allowed to acknowledge the message. The sender cannot be interrupted for any reason in the simplex system.

The duplex system of communication allows traffic to flow in both directions simultaneously between the source and the receptor. This system then provides a natural supervision system, between the ends of the communication channel, since either end may interrupt the flow of traffic at any time.

Communication networks may be designed around either of these two basic concepts. However, when more than one subchannel of information is carried on a single carrier the simplex system concept may be unmanageable. Simplex

operation of more than one subchannel requires simultaneous reversal of the circuit direction. In many cases the subchannel information does not lend itself to synchronous reversal of direction of transmission. For instance, error correction could cause an immediate reversal of direction of one subchannel which is independent of the reversal of another subchannel. In general, user requirements cause reversals at the ends of sentences or other structural parts of the information.

APPENDIX C

NET CONTROL AND CIRCUIT MANAGEMENT CONSIDERATIONS

C-1. GENERAL

When communication systems expand to more than two terminal stations the factors of interference between stations and system supervision must be considered.

With more than two terminal stations, it becomes necessary to consider the type of communication employed. With the employment of a wire system, communication is bounded and interference between stations is inherently not a problem. If radio links are employed, interference between stations may be eliminated by the use of highly directional antennas and multiple frequency operation.

Since the communication problem presented in this study requires the use of radio links, the discussion which follows will be restricted to this type of communication.

C-2. SIMPLEX RADIO COMMUNICATION

The simplex type of radio communication is discussed on the basis of three stations as shown in figure C-1. On any given frequency at a given increment of time one of the station transmitters will be the source and the other stations the receptors. During another increment of time a different station transmitter will be the source and the other stations will be the receptors. As mentioned in Appendix B,

during an increment of transmission of a given station, transmission cannot be interrupted unless supervisory channels are provided on a separate frequency.

This type of communication may assume several forms depending upon the length of the increment Δt. If Δt is long the system has the characteristics of point-to-point simplex communication. As the length of Δt becomes shorter the system may have the characteristics of point-to-point time-shared pulsed type of communication.

C-3. DUPLEX RADIO COMMUNICATION

The duplex type of radio communication is discussed on the basis of the three stations shown in figure C-2. In this system station A is assigned a transmission frequency of f_1 and a reception frequency of f_2. The stations which are to communicate with station A must be assigned a frequency of transmission of f_2 and a frequency of reception of f_1. Immediately a problem arises as station B cannot communicate with station C on these assigned frequencies since they are both transmitting information on frequency f_2 and receiving information on frequency f_1. A solution to this problem is presented with the introduction of a net control station.

C-4. CIRCUIT MANAGEMENT AND FREQUENCY PLANS

One method of net control is shown in figure C-3. In this arrangement station D has the responsibility of receiving all traffic from stations A, B, and C and transmitting all traffic to stations A, B, and C. The net control station D may also be assigned the responsibility of receiving communication from other net control stations and

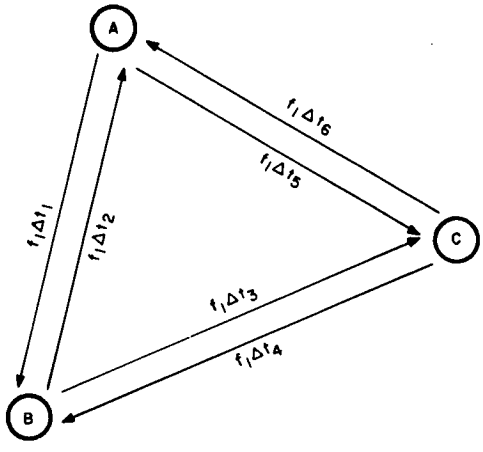


Figure C-1. Simplex Type of Radio Communications

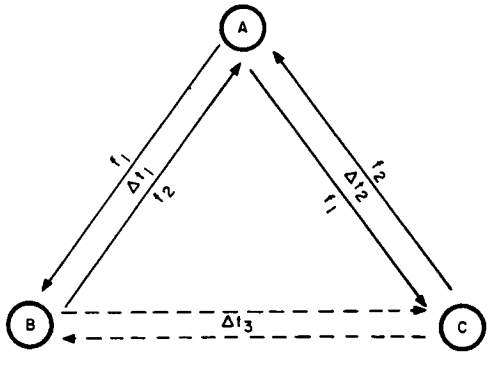


Figure C-2. Duplex Type of Radio Communications

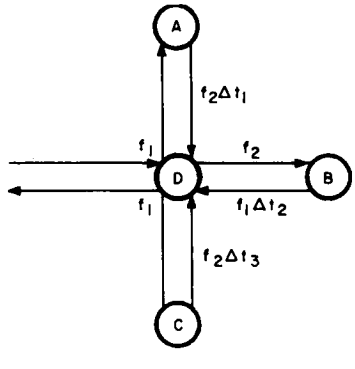


Figure C-3. Net Control Station - Method A

receiving communication from other net control stations and distributing this traffic to stations A, B, and C.

A second method of net control is that shown in figure C-4. In this system several frequencies are assigned to net control station N. The number of frequencies assigned is dependent upon the phase of operation and the traffic density. Each station A, B, C or D requests a frequency assignment from net control station N on the supervisory channel when traffic is to be transmitted. These frequencies may be assigned for any required length of time and then released to the net control station when the particular transmission is completed.

An alternate solution to the frequency overlap problem introduced in figure C-2 is shown in figure C-5. Each station has one transmitter and several receivers. The number of receivers required depends upon the number of communication links. Some type of band filtering must be employed to allow simultaneous operation of the transmitters and receivers for full duplex operation. This separation of signals may be done by electronic filtering, transmitted and received wave polarization, transmission and reception directivity or large spacing between the transmitter and receiver. The method usually employed where space is restricted is the electronic type of filtering. Stations A, B, C, and D are operated on frequencies f_{1a}, f_{1b}, f_{2a}, and f_{3a}.

The use of filters f_1, f_2, and f_3 is shown in figure C-6. The bands f_1, f_2, and f_3 are any band of non-overlapping frequencies chosen for the system. Frequency f_{1a} is any discrete frequency in band f_1; frequency f_{1b} is another discrete frequency in band f_1. Frequency f_{2a} is some

discrete frequency in band f_2. This type of frequency assignment is carried to completion for the number of stations under consideration. If the band filters are designed to fulfill the transmit or receive requirements, complete interconnection flexibility may be achieved. The differences between station B and station C as shown in figure C-5 are the interconnections between the transmitters, receivers, and band filters, and the frequencies of the transmitters and receivers.

C-5. OLO INTERCONNECTION PLAN

The OLO communication links, as shown in figure C-7, are an application of the previously mentioned considerations. Although there is continuous motion of the OLF complex with respect to the operations center, there is always a possible relay path between the operations center and any vehicle in space through the satellite ring and the OLF. Other possible paths may be established upon demand.

A sequence of possible communication paths are shown in figures C-8A through C-8F. These examples are by no means restrictive. Many other possible combinations may be suggested by the operational requirements. Figures C-8A through C-8D illustrate a series of configurations for communication to a payload during a rendezvous operation including contact with the ground, contact with the ground and the OLF, contact with the OLF and the astrotug and joining the OLF. Figures C-8E and C-8F illustrate possible communication complexes that could be used later in the program. In this case a payload is launched at some distance from the operations center. Communications may then be required with the OLF and the operations center immediately after launch. This communication link is possible, as shown in figures C-8E and C-8F, through the satellite relay ring.

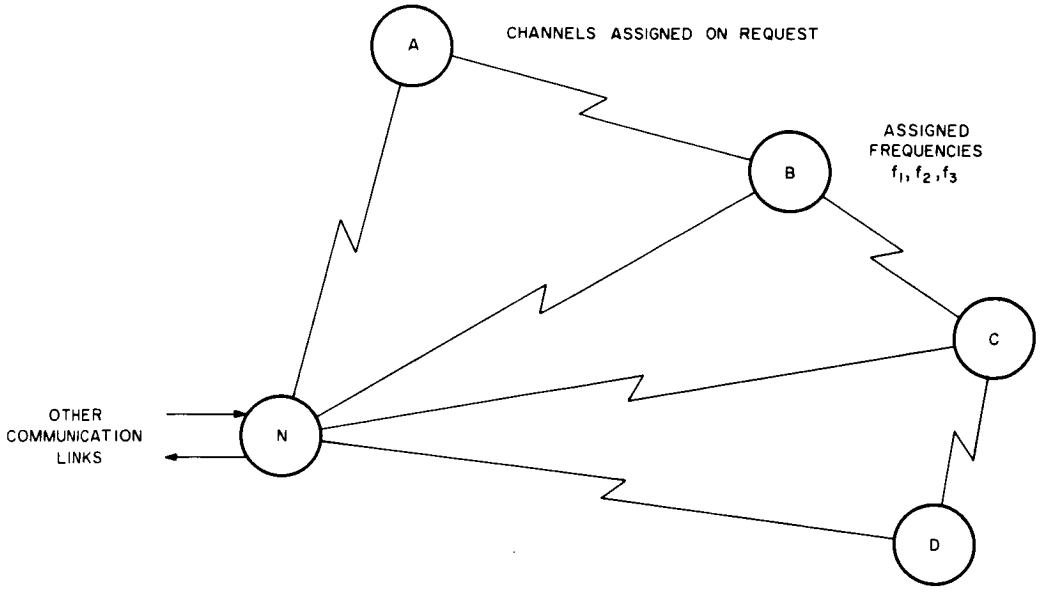


Figure C-4. Net Control Station - Method B

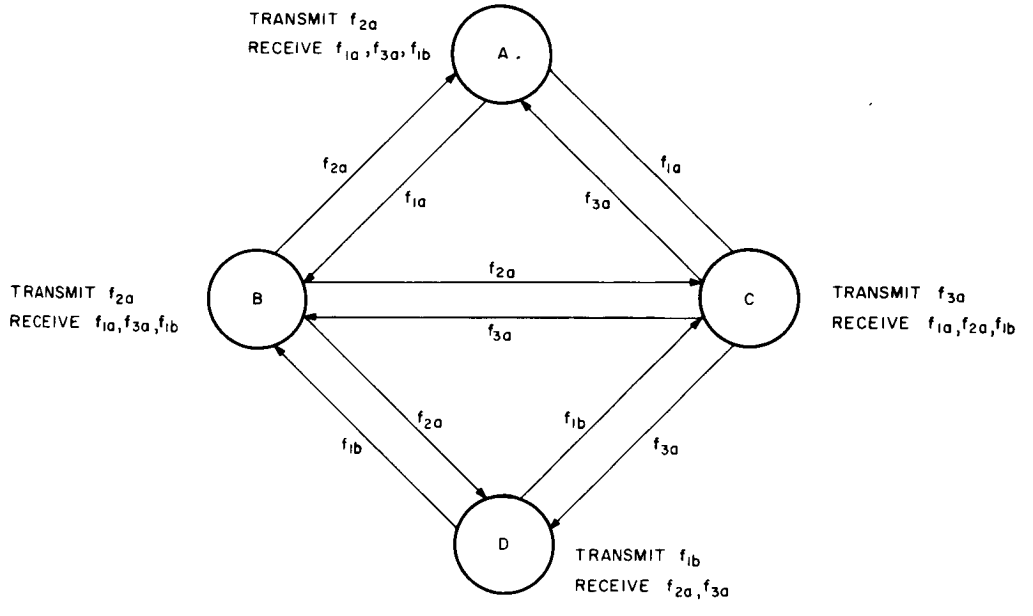


Figure C-5. Alternate Interconnect Plan

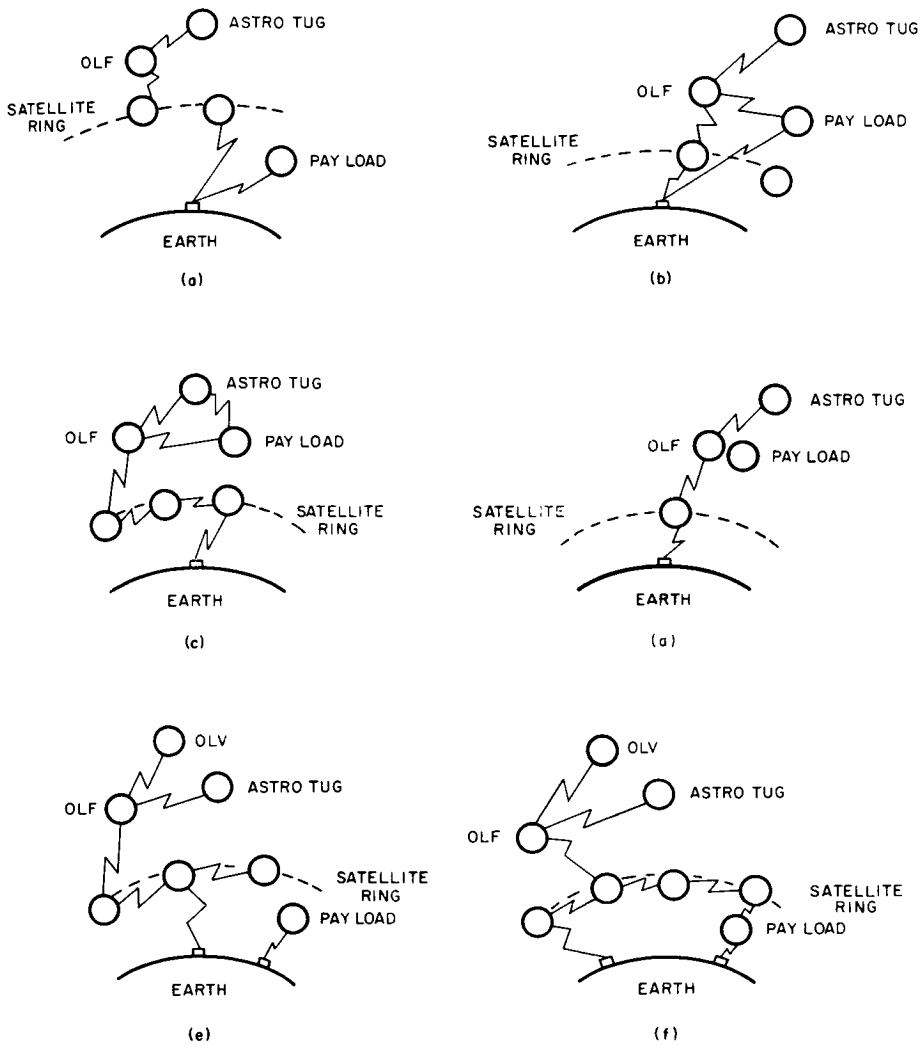


Figure C-8. Sequence of Possible Communication Links

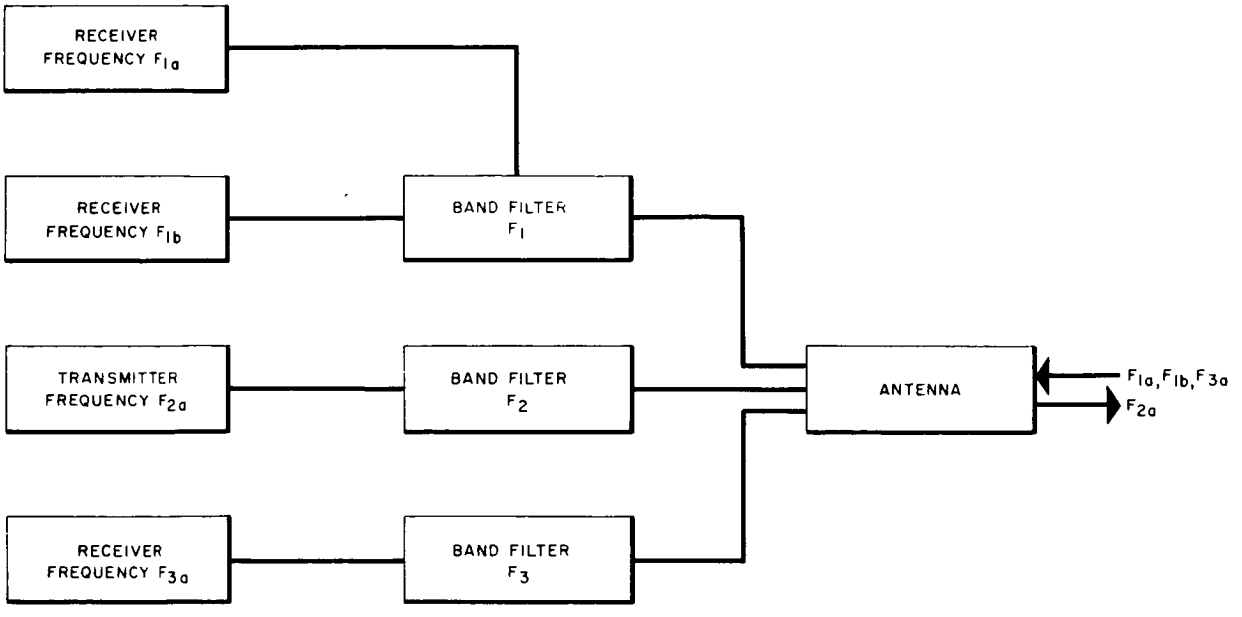


Figure C-6. Typical Station B of Alternate Interconnection Plan

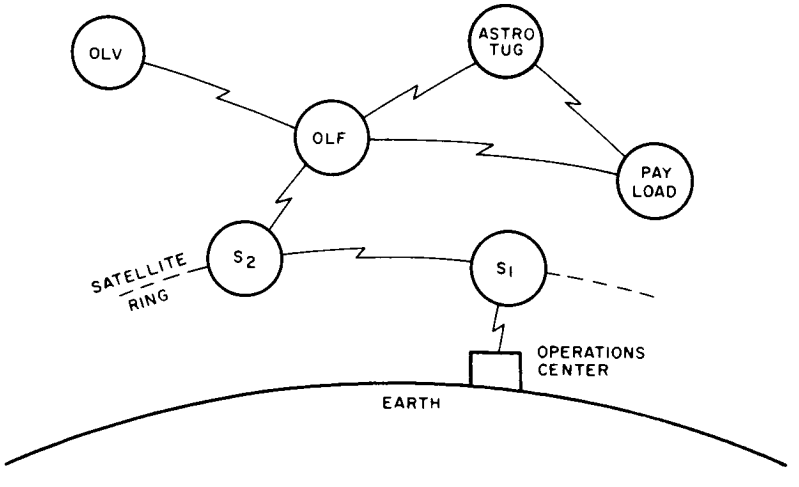


Figure C-7. Plan for OLO Communication Link

APPENDIX D

THE FENCE

In the communications system study, the fence has the assignment of receiving and transmitting to and from the satellite ring and then relaying this information to the center of operations. The first part of this assignment is limited in capability (bandwidth and signal-to-noise ratio) within all practical limits by the power or weight of the space borne equipment.

The immediate solution of the second half, that of utilizing HF radio along the fence, introduces a limitation on capability that is somewhat independent of equipment. The assignment of wide channels in the HF spectrum and the vagaries of the ionosphere as a transmission medium, have a considerable effect on what may be accomplished by HF radio. None the less, the HF solution has been pursued. Overall comparisons may then be made with other means. Cables, or ground-to-ground communication via random or 24 hour satellites such as the advent system, should be re-examined as candidates for improvement of the system. Unfortunately, time has not permitted the evaluation of these aspects of the problem and they must be relagated to the list of items for further study.

Figure D-1 is a block diagram of a typical fence relay station. Each station may be divided into three sections, the equipment to and from the satellite ring, the equipment for up-fence operation and the equipment for down-fence operation. The last two portions are identical. All

three of these sections are never used simultaneously. The dashed lines in the center of the diagram indicate the connections that are used when the station is used as an HF relay only with the UHF pick-up from the satellites ring is being done farther down fence. These sections are made up of transmitters, receivers and data terminals. Voice paths through the equipment have been omitted for simplicity. Further details of each section will be given after a short discussion of the use of this chain of station.

The path from satellite ring to Cape includes one station utilizing its UHF equipment to and from the satellite ring plus one or more stations utilizing both halves of the HF equipment up and down fence as a relay. The most northerly and most southerly station require only one-half the HF equipment. The fence stations shown in figure 4-3, of the main text, were selected to coincide as far as possible with existing stations used for other programs. The existing stations are closer together than necessary but this is an advantage since it provides some natural alternate routes so important to overall reliability. Each hop is on the order of 750 statute miles and therefore there are many combinations that might be used in returning to the Cape. Although, as far as managing the chain of stations is concerned, it would be simple to always relay through the next northerly station, however, this would not be the most desirable from a frequency assignment basis. Single hops should be utilized when propagation is good and multiple hops when propagation is bad. In this regard, it is interesting to note that north-south paths are to be preferred to east-west paths for reliability on HF circuits.

Returning to further details of the stations, figure D-2 shows the details of the UHF section. It includes a

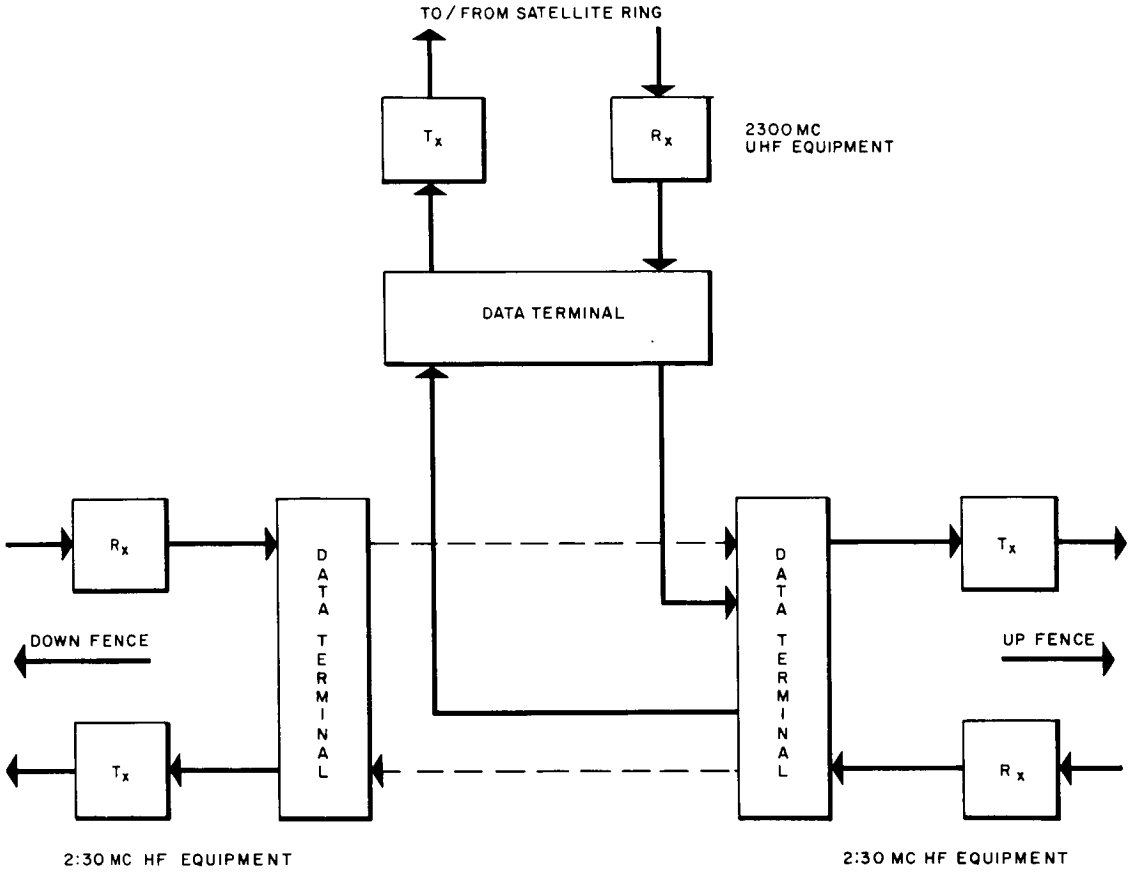


Figure D-1. Fence Relay Station

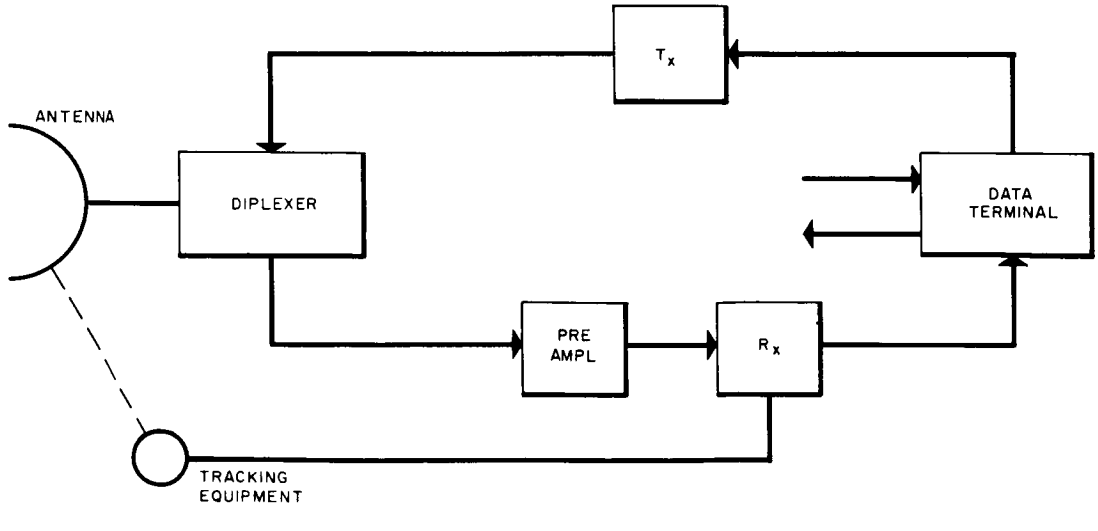


Figure D-2. UHF Section of Fence Station

tracking antenna, a duplexer, a preamplifier, a receiver, a data terminal and a transmitter. An antenna gain of 48 db is selected in order to reduce the equipment in the satellite to a minimum weight. This antenna provides a 0.5 degree beam width which requires accurate tracking. The receiver is of advanced design with a 200° k noise temperature. The transmitter is quite modest, producing only ten watts of power. A data terminal has been included since it is believed that the most flexible system is that in which the data is demodulated to binary form and converted to multi-channel binary form at a slower individual rate to meet the requirements of propagation via the HF medium. Even if the proper composite form already exists as transmitted from the spacecraft, relaying through several HF hops requires regeneration so that data terminal equipment is required at each station. In addition, there are probably some channels of data that have higher priorities than others contained in the composite signal from the UHF circuit. Demodulation is required to assure that priority portions are fowarded up the fence even under conditions of reduced capacity.

The HF section of the fence station consists of several HF systems in parallel each having a capability of 16-kc as shown in figure D-3. Separate transmitting and receiving antennas are shown. Three sets of transmitter and receiver equipment are combined on two antennas. However, the filters are almost the same price as large directive antennas and if land is no problem, separate antennas might be used for each transmitter. The transmitter system includes the antenna, the filter, the power amplifier, the exciter, and the data terminal. The antenna suggested is a vertical logarithmically periodic antenna covering 2-30 mc having a gain of 7 db over a quarter-wave monopole. A bandpass filter,

tuned to the transmitting frequency, having an attenuation of 20 db for the signal from the adjacent transmitter is required. The PA's are manually tuned 10 kw PEP 2-30 mc linear amplifiers. The exciters are manually adjustable in 1-kc steps from 2-30 mc. The data terminals are four phase state, frequency multiplexed, bit synchronous equipment giving about 1 bit per second per cycle of bandwidth (Collins-Kineplex). Figure D-3 shows 3 x 16-kc or 48-kc of bandwidth. The limitation here is not equipment but 48-kc appears to be near an upper limit of the possible frequency assignments. Of course such a system could be implemented in steps starting with a single 16-kc band or less.

Figure D-4 is a table showing fence costs, $20,000,000, for initial period of operation through 1967.

ITEM COST $	BLDG 100K	HF ANT 7 db 2-30 mc 20K	FILTER 20K	PA 10 kw 15K	EX 2-30 mc 10K	PRE AMP 2-30 mc 10K	RX 2-30 mc 10K	TERM KPX 10/B/S (BW = B/S)	2300 mc RF 500K	DATA TERM 500K
STN										
1	1	2	3	3	3	1	3	48K	1	1
2	1	4	6	6	6	2	6	96K	1	1
3	1	4	6	6	6	2	6	96K	1	1
4	1	4	6	6	6	2	6	96K	1	1
5	1	4	6	6	6	2	6	96K	1	1
6	1	2	3	3	3	1	3	48K	1	1
TOTAL	6	20	30	30	30	10	30	480K	6	6
	600K	400K	600K	450K	300K	100K	300K	4800K	3000K	3000K

Total radio equipment and buildings 13,500K

PWR 88K/STN/yr at 10¢ KWH 100 KW/STN for 3 yrs with 0.5 use factor = 792K

Personnel 75K/STN/yr at 15K/M x 5 Man for 4 yrs = 1800K

Repairs 1,300K/yr for 3 yrs = 3900K

Total Through 1967 = 20.0 million dollars

Figure D-4 - Fence Costs for 48 KC Bandwidth

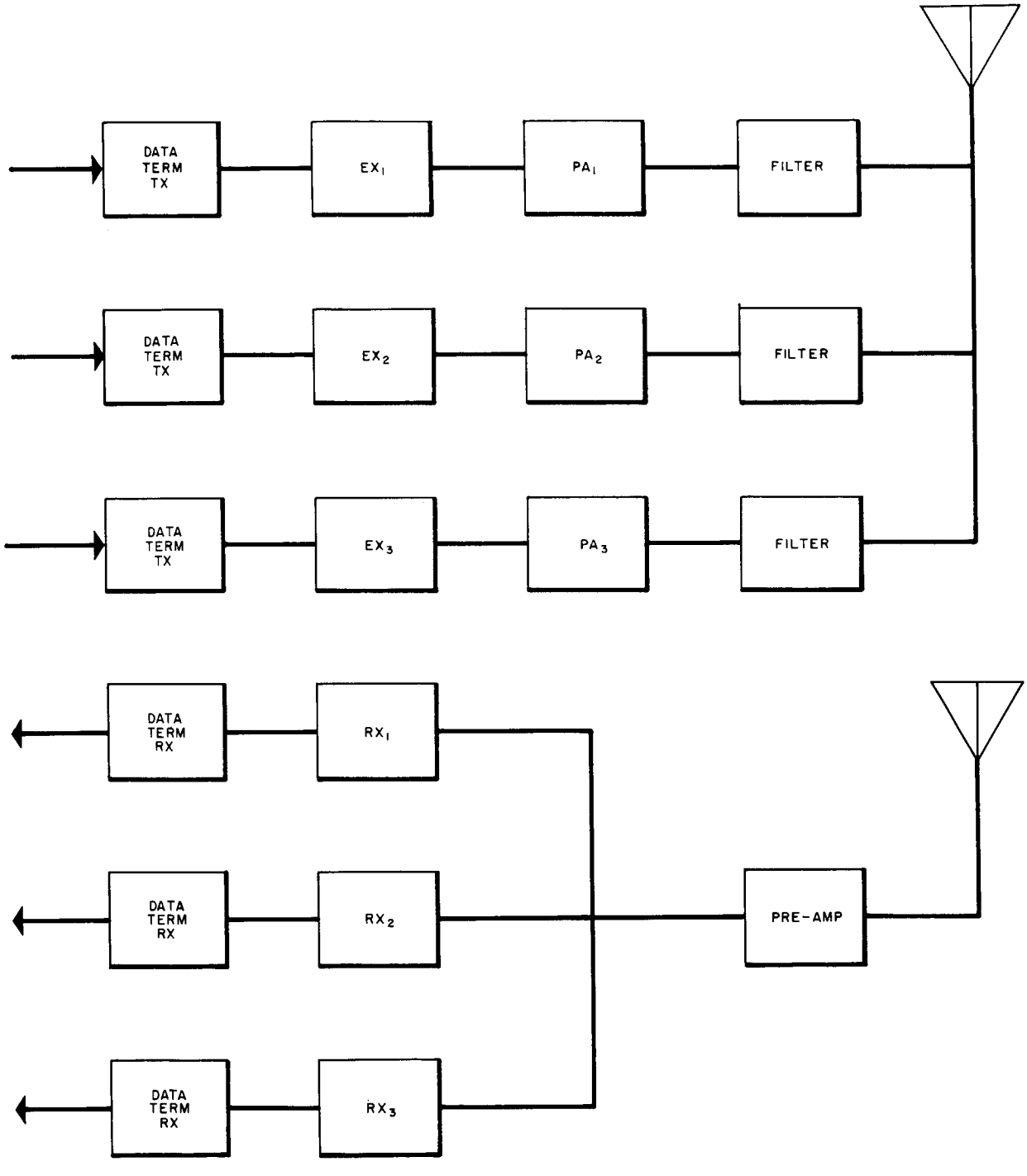


Figure D-3. Half of the HF Section of the Fence Relay Station

APPENDIX E

THE SATELLITE RING

E-1. GENERAL

A ring of satellites as shown in figure E-1 must have certain features to allow flexible operation. This ring may be compared to a microwave relay link across country. The distance between stations becomes greater since the line-of-sight range is greater. A gap between satellites 1 and 12 in the ring prevents closed loop oscillations in the system. There is no break in coverage however since satellite 1 and satellite 12 overlap each other in coverage. Multiple entrances will be provided at each satellite for communication from the ground OLF, OLV, and payloads.

E-2. FREQUENCY PLAN

Employment of the previously mentioned principles will result in the development of a frequency plan. This plan is based upon multi-band filters for the OLF, OLV, astrotug and satellite ring. One of the many possible frequency plans is shown in figure E-2. This figure indicates the number of frequency bands which would be required. The specific system under final consideration will determine the number of band-filters, transmitters, receivers etc that are necessary for the system. In figure E-2 f_1, f_2, etc indicate the number of bands required in the communication plan. These bands determine the number of band-filters required.

Calculations of candidate hardware, shown in figure 5-1, for the satellite ring are presented in the following paragraphs. These calculations are based on a 40 db signal-to-noise ratio at each station or a minimum overall signal-to-noise ratio of 30 db. This provides excellent error rates and may be on the conservative side. Any conservatism may be converted directly into added bandwidth at double the bandwidth for each 3 db of signal-to-noise. The 200-kc is I-F bandwidth and for low deviation F-M system must be divided by about 3. This is equivalent to 75 kc of base bandwidth with 30 db signal-to-noise ratio from OLF to fence.

The calculations are based on the following concepts:

a. All satelite receivers and transmitters are of an elementary variety. Receivers are crystal or traveling-wave-tube (TWT) type. Power of the transmitters are held to a low level. Two watts at 2.2 kmc is attainable with all solid state devices. Powers above this require a tube amplifier. Higher solid state powers may be obtainable at the time of unit development.

b. The size of the antenna for the satellite to OLF link is restricted by the orbit of the OLF with respect to the satellite.

c. The size of the satellite antenna is restricted by the degree of orientation accuracy of the satellite.

d. Parametric receivers are allowed in the OLF and a master receiver is allowed on the ground.

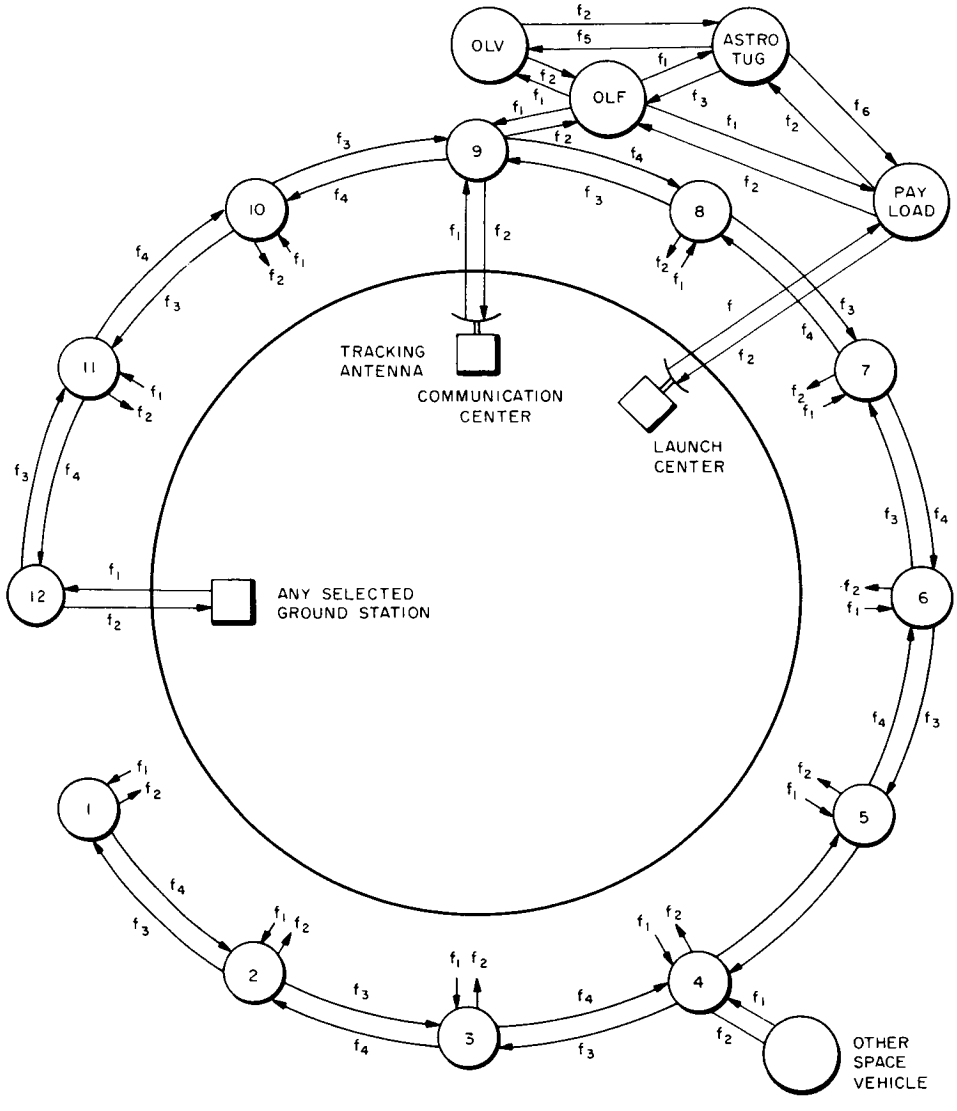
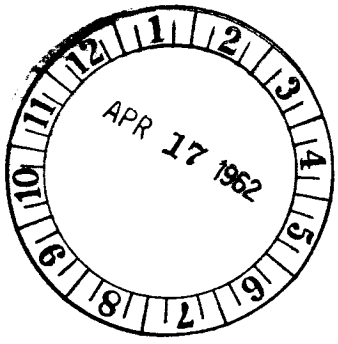


Figure E-2. Ring Frequency Plan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level. Wherever possible, this feature will be incorporated into the OLF communications system to avoid excessive weight of checkout equipment in the OLF.

(7) Power Supply Requirements. Power for the telemetry system will be supplied through a d-c to d-c converter with a multiwinding output. Each output winding will supply the necessary voltage level and isolation for a given subsystem. Balanced diode quads will be used in rectification to ensure minimum offset flux being produced by unbalanced loading. Voltage regulator sections will be provided, where needed, for precision reference sources. Power supply filtering will be used to minimize RF interference (conducted or radiated) from being fed to the primary power source.

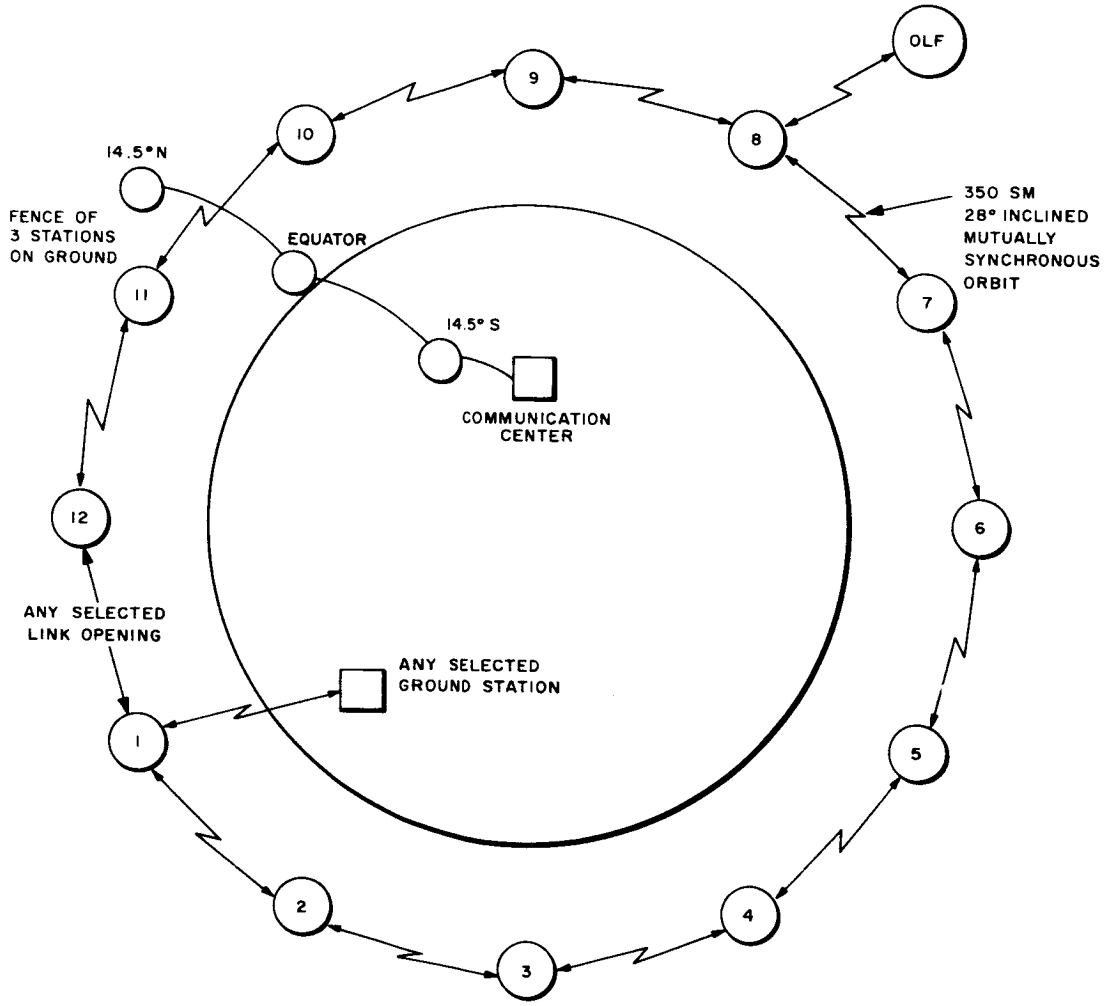


Figure E-1. Satellite Ring

e. An orientable antenna is necessary on the OLF, OLV and astrotug. (mechanical or electronic orientation).

Six sets of calculations follow for the six paths involved. Following these calculations is a summary of costs, weights and power requirements for the communication satellites.

E-3. 200-KC SYSTEM

a. Ground Station Range (350 to 1450 statute miles). Refer to figure E-3.

SATELLITE TRANSMIT EARTH RECEIVE

S/N 40 db = 10^4

Bandwidth 2×10^5

Receiver noise temperature 200°k

λ = 5 inches (2300 mc)

Gt = Gain of transmitting antenna 3 db (gain = 2)

Gr = Gain of receiving antenna

Pt = Power of transmitter in watts (2 watts)

$$Gr = \frac{S/N \; B \; 16\pi^2 \; R^2 \; k \; tr}{\lambda^2 \; Gt \; Pt}$$

$$Gr = \frac{10^4 \times 2 \times 10^5 \times 16\pi^2 \; (1450)^2 \times 5280)^2 \times 1.38 \times 10^{-23} \times 200}{(5/12)^2 \times 2 \times 2}$$

$$Gr = 7.35 \times 10^4 / \text{ or } \quad Gr = 48.66 \text{ db}$$

Antenna Size

$$A = \frac{Gr\ \lambda^2}{6.3}$$

$$= \frac{7.35 \times 10^4\ (5/12)^2}{6.3}$$

$$= 2.03 \times 10^3$$

$$D^2 = \frac{4A}{\pi}$$

$$= \frac{4}{\pi} \times 2.03 \times 10^3$$

$$= 0.259 \times 10^4$$

$$D = 0.51 \times 10^2$$

D = 51 ft DISH

b. Ground Station Range (350 to 1450 statute miles). Refer to figure E-3.

EARTH TRANSMIT SATELLITE RECEIVE

S/N 40 DB = 10^4

Bandwidth 2×10^5 cycles

Receiver noise temperature 870°K (crystal receiver or twt)

λ = 5 inches (2300 MC)

Gt = Transmitter antenna gain 7.35×10^4

Gr = Gain of receiving antenna (3db) Gr = 2

Pt = Power of transmitter in watts

$$Pt = \frac{B\ S/N \times 16\pi^2\ R^2\ k\ Tr}{Gr\ Gt\ \lambda^2}$$

$$= \frac{2 \times 10^5 \times 10^4 \times 16\pi^2 \times (1450)^2\ (5280)^2 \times 1.38 \times 10^{-23} \times 870}{2 \times 7.35 \times 10^4 \times (5/12)^2}$$

= 6.23 watts

Pt assigned 10 watts

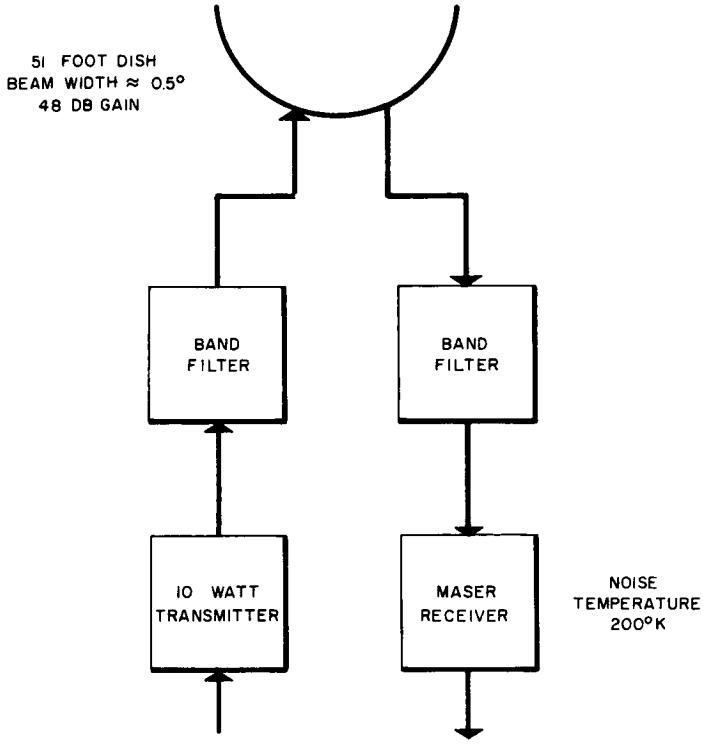


Figure E-3. Ground Station

c. OLF - Satellite (range 2400 statute miles). OLF has tracking antenna.

OLF TRANSMIT SATELLITE RECEIVE

S/N 40 db = 10^4

Bandwidth 2 x 10^5 cycles

Receiver noise temperature 870°K

λ = 5 inches (2300 MC)

Gt = Transmitting antenna gain 8 ft dish 32.6 db 1825 gain

Gr = Receiving antenna gain = 124 (20.94 db) 2 ft dish

$$Pt = \frac{B \times S/N \times 16\pi^2 \times R^2 \times k\ Tr}{Gr\ Gt\ \lambda^2}$$

$$= \frac{2 \times 10^5 \times 10^4 \times 16\pi^2 \times (2400)^2 \times (5280)^2 \times 1.38 \times 10^{-23} \times 870}{124 \times (5/12)^2 \times 1825}$$

= 7.95 watts

Use 8 watt transmitter on OLF

d. Satellite - OLF (Range 2400 statute miles)

SATELLITE TRANSMIT OLF RECEIVE

S/N 40 db = 10^4

Bandwidth 2 x 10^5 cycles

Range 2400 miles

B = 200KC

λ = 5 inches (2300 MC)

Gt = Gain of receiving antenna 8 ft dish (32.6 db 1825 gain)

Gr = Gain of transmitting antenna 2 ft dish (20.93 db) 124 gain

Tr = 340°K (parametric receiver)

$$Pt = \frac{B \times S/N \times 16\pi^2 \times R^2 \times k \times Tr}{Gr \times Gt \times \lambda^2}$$

$$= \frac{2 \times 10^5 \times 10^4 \times 16\pi^2 \times (2400)^2 \times (5280)^2 \times 1.38 \times 10^{-23} \times 340}{124 \times 1825 \times (5/12)^2}$$

Pt = 3.1 watts

Use 4 watt transmitter

e. Satellite to Satellite

$$Gt = \frac{6.3A}{\lambda^2} = \frac{6.3(4)^2\pi}{(5/12)^2}$$

$$= 1825$$

$$= 32.6 \text{ db}$$

$$\text{Beam} = \frac{60\lambda}{D} = \frac{60 \times 5}{8 \times 12}$$

$$= \frac{120}{72} = 1.67$$

$$= 3.340$$

S/N 40 db = 10^4

Bandwidth 2 x 10^5 cycles

Range 2400 miles

k = 1.38 x 10^{-23}

Tr = 870°K (crystal or twt receiver)

λ = 5 inches (2300 MC)

Gt = Gain of transmitting antenna 8 ft dish (32.6db) 1825 gain

Gr = Gain of receiving antenna 8 ft dish (32.6db) 1825 gain

Pt = Power of transmitter watts

$$Pt = \frac{13\ S/N\ 16\pi^2\ R^2\ k\ Tr}{Gr\ Gt\ \lambda^2}$$

$$Pt = \frac{2x10^5x10^4x1.579x10^2x1.607x10^{14}x1.38x10^{-23}x.87x10^3x1.44x10^2}{(1.825)^2\ x\ 2.5\ x\ 10^7}$$

Pt = 1.04 watts

f. OLF to Astrotug

Range in statute miles desirable = 300

$$Gr\ Gt = \frac{B\ S/N\ x\ 16\pi^2\ x\ R^2\ k\ Tr}{\lambda^2}$$

$$= \frac{2x10^5x10^4x16\pi^2(300)^2\ x\ (5280)^2x1.38x10^{-23}x870}{(5/12)^2}$$

$$= \frac{2x10^9x1.579x10^2x2.51x10^{12}x1.38x10^{-23}x87x16^3x1.44x10^2}{25}$$

$$= \frac{2x1.579x2.51x1.38x.87x1.44x10^3x4}{.25}$$

$$= 55 \times 10^3$$

$$= 5.5 \times 10^4$$

$$Gr = Gt = 2.34 \times 10^2$$

$$= 234$$

$$Gt = \frac{6.3\ D^2 \pi}{\lambda^2\ 4}$$

$$4Gt\ \lambda^2 = 6.3\ D^2 \pi$$

$$D^2 = \frac{4Gt\ \lambda^2}{6.3\pi}$$

$$D^2 = \frac{4 \times 234 \times (5/12)^2}{6.3}$$

$$D^2 = \frac{4 \times 234 \times 25}{6.3 \times 144} = 8.22$$

D = 2.87 ft dish with beam

E-4. SATELLITE EQUIPMENT LIST

	Wt (lbs)	Watts Input
4 Receivers	32	40
2-1 watt transmitters	8	60
1-2 watt transmitter	4	35
1-4 watt transmitter	5	50
1 Duplex coupler	3	5
8 Band pass filters	8	-
2-8 ft dishes	60	-
1-2 ft dish	10	-
1 Slot antenna	5	-
Power Supply System Batteries and Solar Cell	120	
Control Equipment	20	
Electronic Control Equipment	20	
Frame	100	
TOTAL	395	190

APPENDIX F

OLF COMMUNICATIONS AND TELEMETRY TERMINAL SYSTEM

F-1. GENERAL

In previous sections the general requirements for the OLF terminal system were presented. In this appendix the system is treated in detail; types of data to be handled are discussed; channel assignments estimated; projected evolution is presented; reliability, flexibility, and maintainability requirements are stated; candidate systems are discussed; and finally the characteristics and operation of the selected system is presented in detail.

F-2. TERMINAL SYSTEM DATA REQUIREMENTS

The OLF telemeter will be required to handle diverse data. Its capabilities will include data transmission and reception between both the ground complex and the OLV. Breakout of data links is given in the following paragraphs.

a. OLF Transmitted Data. The data to be transmitted by the OLF telemeter is as follows:

(1) OLF to Ground Complex

(a) OLF telemetry data showing the condition of the OLF and its crew and the results of scientific data acquisition. Specifically, this will include monitoring of OLF guidance,

attitude controls, propulsion, temperatures, life support parameters, bio-medical information, and miscellaneous equipment performance.

(b) Single frame video for the transmission of written displays, pictures of local conditions, and other images as required.

(c) OLV telemetry information, relayed to the ground complex via the OLF during countdown and the initial portion of powered flight by the OLV. Relay distances in excess of 4,500 statute miles do not appear feasible because of transmitter power requirements; consequently, DSIF must assume control early in flight. This approach presents several advantages in that it reduces the required OLV communication payload and the required capabilities of the ground complex while allowing a single station to maintain control of the OLV during initial powered flight. However, use of the OLF as a relay station for OLV intelligence will require expansion of the data transfer capabilities of the OLF.

(2) OLF to OLV Transferred Data

(a) OLO mission data consisting of guidance and trajectory data for the OLV.

(b) Parameters required for checkout and countdown.

(c) Command signals relayed from the ground complex.

(d) Voice transferred from ground complex.

b. OLF Received Data. The data to be received by the OLF telemeter is as follows:

(1) Ground Complex to OLF

(a) OLO mission data consisting of guidance and trajectory data for the OLV during countdown.

(b) Commands selecting operating modes of the OLF telemetry equipment and performing other functions as required.

(c) Commands for relay to the OLV while the OLF is acting as a relay station.

(d) Voice communication.

(2) OLV to OLF Transferred Data. All OLV telemetry data for relay while OLF acts as OLV communications link.

c. Estimate of Telemetry Channel Assignments. Before conceptual hardware can be derived, some knowledge of telemetry measurement requirements is mandatory. Figure F-1 presents a listing of representative measurements for a mission of this type together with estimated accuracy requirements and sampling rates. Figure F-1 contains a total of 288 measurements listed under the braod categories of equipment monitor and evaluation, cabin environment, video, bio-medical, human abilities, and general scientific. About 95 percent of the measurements required only limited sampling rates. It is anticipated that OLO mission data will be presented to the telemetry system in serial binary form and that required rates will not exceed 1,000 bits per second.

F-3. OLF COMMUNICATIONS TERMINAL SYSTEM-OPERATING MODES AND EVOLUTION

The OLF communications terminal system will possess a variety of operating modes. These modes include the telemetering of OLF performance and the condition of its crew, the acquiring of scientific data, the receiving of ground commands, the transferring of OLO mission data, and the relaying of spacecraft telemetry data.

In addition, requirements will exist for data storage and transmission on command. As the OLO program progresses, a gradual evolution can be expected in the functions monitored.

FIGURE F-1

Estimate of Telemetry Measurement Requirements

A - Indicates 0.1 to 1%
B - Indicates 2 to 7%

OLF Equipment Monitor and Evaluation

Function	Quantity	Accuracy	Maximum Sampling Rate (samples/sec)
Pitch acceleration	1	A	20
Linear acceleration	3	A	20
Angular rates	3	B	20
Attitude	3	B	20
Reaction jet control	12	A	20
Reaction jet valve position	12	A	1

FIGURE F-1 (Cont)

Estimate of Telemetry Measurement Requirements

Function	Quantity	Accuracy	Maximum Sampling Rate (samples/sec)
Fuel flow (rate)	12	B	20
Fuel supply	6	B	1
Power supply voltages	6	AB	20
Power supply frequency	4	AB	20
Programmer signals	10		1
Radio altimeter	1	A	20
Position data	2	A	20
External temperature	8	B	1
Inside air temperature	4	B	1
Equipment temperature	20	B	1
Meteorite impact	1	B	20
Command receiver signals	10		20
Pitch jets	1	On-Off	20
Roll jets	1	On-Off	20
Yaw jets	1	On-Off	20
Miscellaneous	38	AB	10
General Warning (On-Off)			
Dangerous CO_2 level	1		1
Cabin decompression	1		1
Radiation warning	1		1
Electrical system failure	1		1
Environmental control system failure	1		1

FIGURE F-1 (Cont)

Estimate of Telemetry Measurement Requirements

Function	Quantity	Accuracy	Maximum Sampling Rate (samples/sec)
Attitude control failure	1		1
	Total Functions = 167		
Cabin Environment			
Oxygen concentration	1	B	1
CO_2 concentration	1	B	1
Miscellaneous gasses	6	B	1
Total gas pressure	1	B	1
Gaseous flow	2	B	10
Cabin wall temperature	2	B	1
Cabin air temperature	2	B	1
Radiation rate	4	B	5
Humidity	1	B	1
CO_2 filter status	2	B	1
O_2 reserve	1	B	1
	Total Functions = 23		
Assumed Bio-Medical			
Electrocardiogram (EKG)	1	B	320
Phonocardiogram	1	B	180
Respiration	1	B	20
Electroencephalogram (EEG)	3	B	120

FIGURE F-1 (Cont)

Estimate of Telemetry Measurement Requirements

Function	Quantity	Accuracy	Maximum Sampling Rate (samples/sec)
Galvanic skin resistance (GSR)	1	B	20
Blood pressure	2	B	2
Electro-oculograph (EOG)	1	B	20
Body temperature	1	B	20
Exhaled gases	2	B	20
	Total Functions = 14		
Human Capabilities			
Compensatory tracking	1		1
Auditory vigilance	1		1
Warning light monitoring	1		1
Arithmetic computation	1		1
Pattern discrimination	1		1
Probability monitoring	1		1
Scale position monitoring	1		1
Miscellaneous	10		1
	Total Functions = 17		

Assumed Scientific Date

Scientific experiments will consist of cosmic ray sensors, scintillation counters, magnetometers, ion chambers, lyman alpha telescopes, electric field measurements, magnetic field measurements, plasma detectors, etc. No attempt is

FIGURE F-1 (Cont)

Estimate of Telemetry Measurement Requirements

Function	Quantity	Accuracy	Maximum Sampling Rate (samples/sec)
made to define this area in detail. For system definition, the following assumption is made:			
Miscellaneous	67		1
	Total Functions = 67		
Narrow-band video - 100,000 samples per frame.			

a. Evolution of Data Requirements During Phase II. As the program progresses, the emphasis on acquired data will change as follows:

(1) OLF During Ground Checks. During ground checks, all operating modes will be checked for proper operation.

(2) OLF Orbiting, Unmanned. Cabin environment and equipment performance will be closely monitored. Scientific data acquisition will not be commenced until the crew is on-board.

(3) Manned OLF. During the early period of manned operation, equipment performance, cabin environment, and bio-medical and psychological reactions will be heavily telemetered for detailed ground evaluation. Scientific experiments will be of secondary importance. This

orientation will be required to perform field evaluation of all OLF equipment, detect environmental conditions dangerous to health, and detect dangerous medical conditions which have not been observed during previous programs but which might occur during extended orbital operations.

After the initial proving phase, psychological experiments will acquire higher priority since the over-all OLO concept requires that man be able to function effectively in the orbital environment for extended periods. During this same period, the feasibility of OLV checkout and countdown while in orbit will be proved.

b. Evolution of Data Requirements During Late Phase II. After the initial Phase II operations, equipment performance, bio-medical reaction monitoring, and cabin environment checks will be performed on a scheduled basis. The acquisition of scientific data will now assume a role of importance secondary only to actual OLO operations, and more sophisticated psychological experiments will be performed.

c. Evolution of Data Requirements During Phase III. No marked changes in approach are envisioned during this phase. Two primary considerations will be made:

(1) Keep the system sufficiently flexible to allow the incorporation of a succession of scientific measurements.

(2) Retain sufficient capabilities to allow for modifications in the OLO program.

d. Evolution of Data Requirements During Phase IV and Subsequent Phases. It is projected that a redesigned communications package will be required at the beginning of

Phase IV to keep the OLF facilities current with the evolving philosophy of OLO. A summary of telemetry requirements by operation is shown in figure F-2.

e. Data Storage and Transmission. After the OLF has been established in orbit, continuous data transmission is undesirable because of the extremely large quantities of redundant data which would be collected. Some data is ideally suited for storage with or without time correlation, while other data is of value primarily in real time. The requirements on the various types of data are discussed in the following paragraphs.

(1) Bio-medical and Psychological Data. The crew members of a facility, such as the OLF, cannot be expected to wear a bio-pack at all times. Therefore, a bio-unit will probably be provided which will acquire data on each man after it is put on. A psychological test unit of the same type will be employed. Then data could be collected at any free period throughout the day and stored for later transmission. No real-time correlation would be required.

(2) Scientific Data. Scientific data is primarily of a slowly changing nature and is gathered over long periods of time. To circumvent the continual ground processing of this data, it can be stored and transmitted later at a much higher rate to a single station. Real-time correlation will be required in this area.

(3) Video. Image storage will be required in this area to preserve desired pictures until ready for transmission.

FIGURE F-2

Summary of Telemetry Requirements
By Operation

	OLF Ground Checks	Injection to Orbit	OLF Unmanned in Orbit	Initial Manned OLF	Rendezvous and Docking	Orbital Launch Operations	Routine Operations (Phase II)	Routine Operations (Phase III)	Routine Operations (Phase IV)	Routine Operations (Phase V)
Equipment Performance and Monitoring	X		X	X	X	X	X	X	X	X
Cabin Environment Monitoring	X		X	X	X	X	X	X	X	X
Bio-Medical Monitoring	X			X			X	X	X	X
Psychological Monitoring	X			X			X	X	X	X
Receive Commands	X	X	X	X	X	X	X	X	X	X
Transfer OLO Mission Data	X					X				
Relay of OLV Telemetry	X					X				
Video Transmission (Slow Rate)	X			X		X	X	X	X	X
Scientific Data Acquisition	X				X	X	X	X	X	X
Operate Tracking Aids	X	X	X	X	X	X	X	X	X	X
Operate Voice Links	X			X	X	X	X	X	X	X
Two-Way Video (Real Time)									X	X

(4) OLF Equipment Operation and Monitoring. After the OLF is established in orbit, ground checks of the OLF equipment will be conducted at predetermined intervals. No data storage will be required, since the primary application of this type of data is in real time.

(5) OLO Mission Data. OLO mission data will be received by the OLF from the ground complex and transmitted to the OLV. Storage will be provided by the OLF computer with transfer initiated by either local or remote command. The OLF telemetry system will not possess storage capabilities for this type of data.

Based on the preceding discussion, the concept has been developed that surveillance of the OLF will be performed by ground facilities at scheduled times. Real-time OLF equipment operation or stored bio-medical, psychological, or scientific data will be transmitted by the OLF as selected by ground command. Measurement handling operation is reviewed in figure F-3.

FIGURE F-3

Measurement Handling Operation

Measurement	Real Time	Store and Dump
Bio-medical		X
Psychological		X
Equipment performance and monitoring	X	
Scientific data		X
OLO mission data	X	

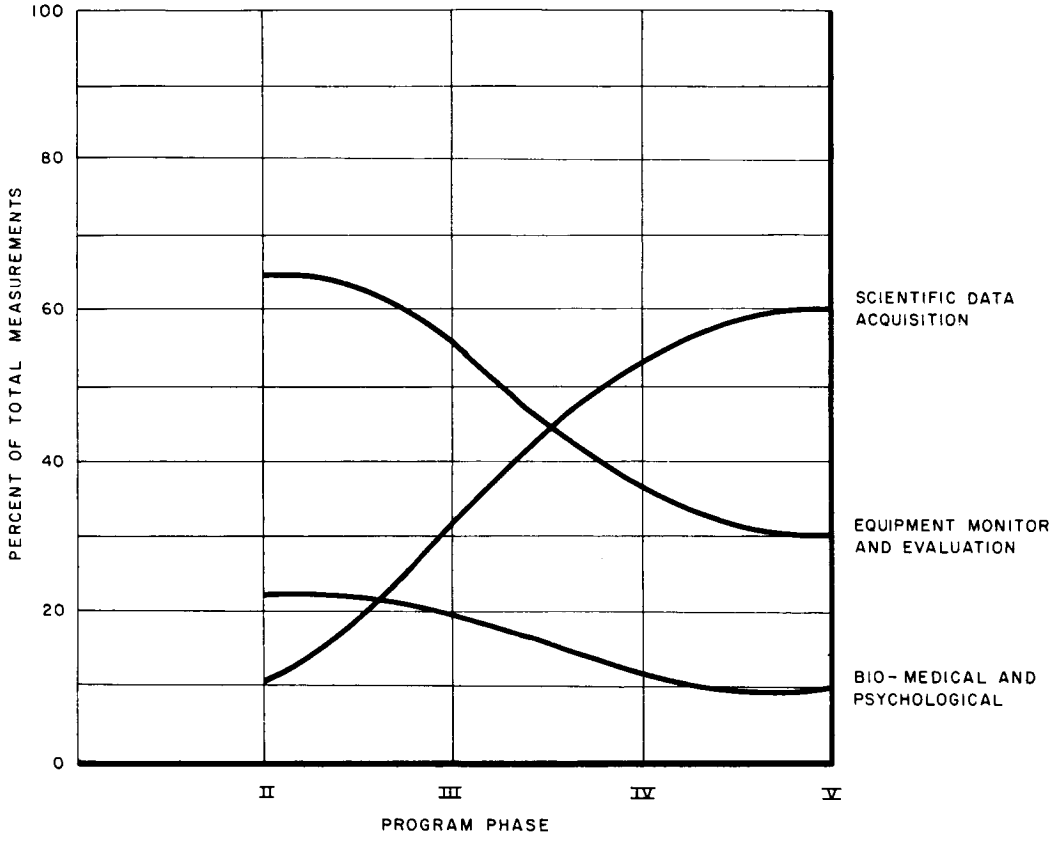
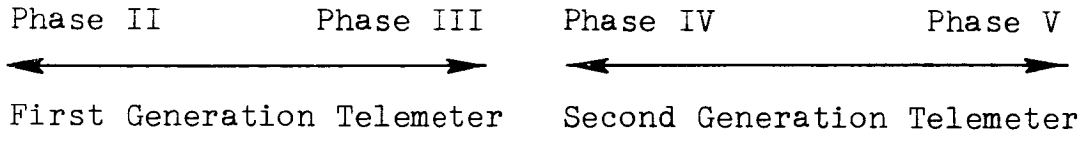


Figure F-4. OLF Measurements Requirements

f. Evolution of OLV Telemetry Measurements. It is projected that the total number of telemetry measurements will remain constant throughout the OLO program. Equipment monitoring and evaluation will decline as confidence and experience increase, but will be offset by an increase in scientific data acquisition. A graph showing the anticipated telemetry evolution is shown in figure F-4.

g. Equipment Evolution and Hardware Schedule. It is projected that two generations of communications equipment will be required in support of OLO. First generation equipment will be installed in the initial OLF. Changes in measurement requirements, refinements in orbital launch operations, a gradual shift of telemeter usage from an experimental to a research-tool, and OLF evolution will require a second generation terminal system for the intermediate OLF. The second generation equipment will be sufficiently flexible to perform adequately throughout the remainder of the present program, including use with the ultimate OLF. Hardware evolution is as follows:



F-4. OLF RELIABILITY

The reliability of the OLF terminal system will be an extremely critical factor. The inherent high cost of supplying replacement units after the OLF is in orbit and because of the OLO operational requirements, a maximum effort must be made during all preflight operations, including

design, fabrication, and checkout, to ensure that failure-free equipment is placed in orbit.

The reliability considerations for OLF communications equipment are presented in detail in volume 6, Appendix B of this report. Component selection, burn-in, and extremely conservative circuit design will be employed to insure overall reliability. In addition, designs will incorporate multiple-use circuitry at the module, subassembly, and assembly levels wherever possible. A summary of the OLF terminal equipment reliability is presented in figure F-5.

FIGURE F-5

Communications Terminal System

	Phase			
	2	3	4	5
Signal Conditioning	0.9896 0.9999	0.9913 0.9999	0.9931 0.9999	0.9949 0.9999
Decoder	0.9886 0.9999	0.9904 0.9999	0.9921 0.9999	0.9939 0.9999
Receivers	0.9886 0.9999	0.9904 0.9999	0.9921 0.9999	0.9939 0.9999
Transmitters	0.9886 0.9999	0.9904 0.9999	0.9921 0.9999	0.9939 0.9999
Storage	0.9896 0.9999	0.9904 0.9999	0.9931 0.9999	0.9949 0.9999
Antenna	0.9935 0.9999	0.9953 0.9999	0.9970 0.9999	0.9987 0.9999
Total Reliability	0.9400 0.9998	0.9500 0.9998	0.9600 0.9999	0.9700 0.9999
Critical Period (hrs.)	12	8	6	2

Phase 4
X.XX basic equipment reliability
X.XX reliability with 100 percent redundancy

F-5. OLF MAINTAINABILITY

A detailed discussion of maintainability is presented in Volume 6, Appendix D. However, some additional specific comments and a summary of the Volume 6, Appendix D material is presented here for reading continuity.

Accessibility and fault isolation are two of the major problems to be considered. The design objective for the communications terminal equipment will be fault isolation to the subassembly level through the use of self-check and ground-check techniques. For fault isolation at the module level, programming of the OLF checkout computer for subassembly testing and module isolation will be investigated. If the use of the checkout computer is feasible, module fault isolation by the computer will reduce the requirements for men as troubleshooting agents.

F-6. OLF SYSTEM FLEXIBILITY

As the OLO program progresses considerable system flexibility must be maintained because of expected long-term operation and rapid evolution of system requirements. Two basic approaches can be used to achieve this objective.

a. The telemeter can be provided with a memory unit and the program altered by local or ground command.

b. Man can act as a controlling agent and alter the program as required.

A completely automatic system for use with the OLF would be too complex and too costly. The optimum solution

appears to be a limited command program selection involving the basic telemeter only and man acting as an agent for all measurement changes.

Sufficient system capacity will be provided to allow the addition of up to 30 percent of the original number of telemetry channels. The basic telemeter will provide the transmission capacity for this number of channels. The addition of a measurement will require only the insertion of a selected module or subassembly in the telemeter and wiring in the appropriate sensor.

The principle of on-board data reduction to limit transmitter bandwidth requirements and to reduce the amount of redundant data received on the ground was also considered. Extensive use of this approach was considered too complex and costly for the OLF communications system, especially in view of the previously developed concept of scheduled OLF surveillance instead of continuous data acquisition.

A single equipment design cannot be sufficiently flexible to be used throughout the OLO program. Program hardware requirements are discussed in paragraph F-3.

F-7. ROLE OF MAN

The OLF communications terminal system will not be required to operate in an unmanned condition except during the period immediately following initial injection of the OLF into orbit. During the remainder of the OLO program, the OLF crew will be expected to provide a fundamental portion of system programming, operation, maintenance, and repair. This program will obviate the need for extremely complex equipment and will provide inherent system

flexibility. However, man's responsibility in this operation is restricted by the limited time available for these duties.

a. OLF Telemetry System Programming. Figure F-6 illustrates the concept for OLF telemetry programming. A fixed program under local or ground control determines the basic types of data being processed by the telemeter. Man will normally program only at the sub-level of basic data types (i.e., scientific data, bio-medical data, etc.) where measurements may be removed, added, or otherwise altered within the design limits of the system. This man-machine relationship appears to provide the best trade-off between complexity and flexibility for the OLF system.

b. OLF Communications Terminal System Maintenance. It is anticipated that most circuitry will be digital in nature, thereby reducing significantly the problem of marginal operation. However, some analog circuitry will be employed, particularly in the signal conditioning equipment. This equipment will require periodic adjustment by the OLF crew. The design objective will be to provide measurement points such that a single unit of general-purpose test equipment common to other OLF equipment can be utilized.

c. OLF Communications Terminal System Repair. Fault isolation and repair were discussed in paragraph F-5. Subassembly fault isolation will be performed by programmed self-checks and ground-checks, while module fault isolation will rely on the diagnostic ability of man in conjunction with checkout programming of the OLF computer.

F-8. CANDIDATE SYSTEMS

a. General

The initial requirements placed on the OLF communications terminal system were discussed in the preceding paragraph. In the selection of a system with the capability of meeting these requirements, a number of factors bearing on each requirement were considered. These factors included accuracy, reliability, checkout, repair, power requirements, weight, size, and interchangeability. It must be recognized that these relevant factors may vary in importance depending on the mission requirements of the system. The following discussion of candidate systems is based in part on the relative weighting of the system characteristics discussed.

In view of the fact that an analog system, a digital system, and an analog-digital combination provide a representative sampling of the numerous special combinations of telemeters in use, these systems were subjected to close evaluation. Specifically, a PAM/FM/FM, a PCM/FM, and a hybrid system employing PAM/FM, PCM/FM, and FM/FM were chosen for evaluation.

b. Evaluation of Candidate System. A program-orientated evaluation was performed on the candidate systems to determine OLO mission efficacy. The results of this evaluation are shown in figure F-7. The system rated best in each category is scored with a 1. If identical abilities exist for more than one system, each is scored with a 1. The implications of the scoring in figure F-7 are treated in the following paragraphs.

(1) Modulation (carrier). The FM modulated carrier was chosen because of the FM signal-to-noise improvement and

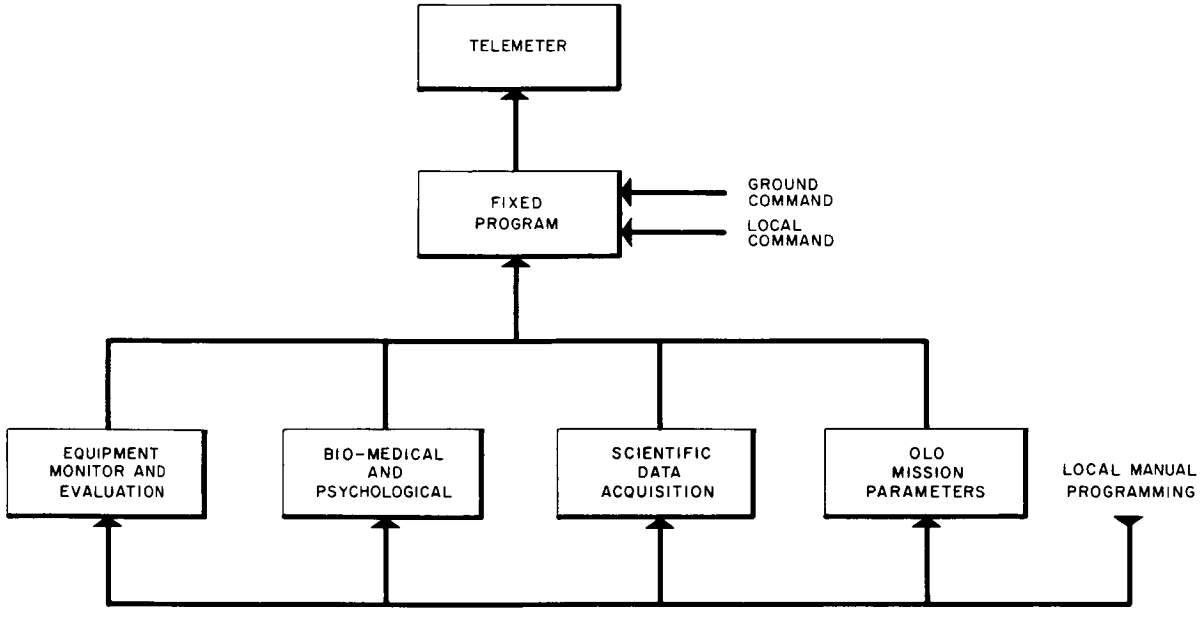


Figure F-6. OLF Telemetry System Programming

because of compatibility with projected ground facilities and other OLO communications. A comparison of PCM/FM, FM/FM and PAM/FM modulation is given in figure F-8.

FIGURE F-7

Comparison of Candidate Systems

	PCM/FM	PAM/FM/FM	HYBRID (PCM/FM/FM) (PAM/FM/FM) (FM/FM)
Optimum carrier modulation	1	1	1
Data format	1		1
Bandwidth (minimum)	1		
Transmitter power (minimum)	1		
Data storage	1		
Self-check	1	1	1
Calibration	1		
Accuracy	1		1
High frequency information		1	1
Analog information		1	1
Threshold signals		1	1
Flexibility	1		1
Reliability	1		
Maintainability	1		
Logistics	1		
Compatibility	1		1
Power	1		

FIGURE F-7 (Cont)

Comparison of Candidate Systems

	PCM/FM	PAM/FM/FM	HYBRID (PCM/FM/FM) (PAM/FM/FM) (FM/FM)
Size and weight	1		
Parallel data transmission			
Score	15	6	10

FIGURE F-8

Telemetry Modulation Comparison*

	PCM/FM	PAM/FM	FM/FM
Information Efficiency	0.24	0.05	0.03
RF Bandwidth - KC	18	85	140
RF Power Relative to PCM/FM	1	4.9	8.2

* Robert H. Dimond, "Interplanetary Telemetering", Proceedings of the IRE on Space Electronics, April 1960. Values based upon an individual channel output signal-to-noise ratio at threshold of 20 db and an information bandwidth of 1,000 cps.

(2) Data format. Since a major portion of the data will be in digital form and because of the required accuracy of certain data (e.g., guidance parameters), a system which

will accept and transmit digital data was chosen. The PCM system readily lends iteself to various code formats which provide synchronization and reduction in total RF power requirements. The PCM format further provides compatibility with digital computers, which eases the data reduction burden.

(3) Bandwidth and Transmitter Power. To achieve an optimum system with respect to minimum launch-vehicle boost requirements, a minimum communications bandwidth must be maintained for the reason that an increase in bandwidth requires an increase in operating power, which in turn prescribes an increase in the weight of the orbital equipment. A PCM/FM system will require approximately 1/8 the bandwidth of the FM/FM or the hybrid system. Through the use of orthogonal coding, PCM will require approximately 1/20 as much energy as a conventional FM/FM telemetry system in transmitting equal amounts of information. Discrete pulse position modulation requires high peak power, and sub-carrier frequency modulation requires the generation of a large number of stable frequencies.

(4) Data Storage. It is assumed that continuous data transmission will be required during initial orbit, rendezvous, and launch. During other periods it is projected that data will be transmitted only at preset periods to preselected stations. To meet these requirements, two magnetic storage units with minimum time correlation capabilities will be required, one to handle digital data and the other to record analog signals. A PCM system is superior for data storage because of the high bit rate encountered during playback.

(5) Self-Check Features. This self-check feature requires a tie-in with the ground complex. As a result, the communications checkout equipment on board the OLF can be considerably reduced if the major portion of the communications checkout burden is placed with the ground complex. This philosophy should be incorporated into the OLF communications system. Failure detection is easier in a PCM system because of the go no-go characteristics of the digital circuitry.

(6) Calibration. The required long life of the communication system makes techniques which depend on calibration for data correlation unsuitable. A PCM system, because of its digital characteristics, most nearly meets the data correlation requirement. PCM exhibits none of the errors caused by zero drift, distortion, and sensitivity change present in the FM/FM, PDM, and PAM systems.

(7) Accuracy. It is projected that the required data accuracy will fall into two ranges, one covering from 7 to 2 percent and the other from 1 to 0.1 percent. Through quantizing, either the PCM or hybrid system has this capability. The PCM system provides variable accuracies which can be programmed into the system through control of the bit rates. The FM/FM system cannot provide the 0.1 percent accuracy required by many of the anticipated guidance parameters.

(8) High Frequency Information. The complexity in programming, multiplexing and encoding sets a practical limit on the range of information rates and accuracy that can be covered by PCM. Low accuracy, high frequency data is best transmitted by FM/FM or the hybrid system.

(9) Analog Information. The PCM system has no capabilities for transmission of analog information. No requirement for analog data transmission is anticipated for the OLF.

(10) Threshold Signals. A PCM system exhibits a well defined threshold which occurs when random noise exceeds one-half of the peak-to-peak signal input to the decision circuit for any significant amount of the detection time. The other systems can provide data of degraded quality in a high noise environment. Threshold signals are not expected, however, in operations involving the OLF where adequate transmitted power will be available to provide working safety margins.

(11) Reliability. Reliability is dependent upon the techniques used in the development and manufacture of a system and upon the type of system chosen. PCM logic readily lends itself to solid state circuitry which provides better reliability. In addition, marginal operation is less frequent in digital circuits. A more detailed analysis of reliability is presented in Volume 6, Appendix B.

(12) Maintainability. By use of self-check features for detection and isolation of subassembly failures and by use of the OLF checkout equipment for failure detection and isolation of module failure, it is projected that a PCM system will require less maintenance than either of the systems which have the capability of handling analog information. Detailed maintenance and logistics concepts are given in Volume 6 Appendix D.

(13) Logistics. By the use of PCM digital building blocks identical to those used in other OLF equipment, and

by repair and, maintenance at the subassembly and module level, it is felt that the spares requirements can be reduced substantially.

(14) Compatibility. Coded data will be used in most OLO operations; therefore, a PCM system is most compatible.

(15) Size and Weight. Microminiature circuit techniques can be employed to a greater extent in the PCM system; it is estimated that a 25 percent size and weight reduction can be achieved in the PCM system over the other two systems.

(16) Parallel Data Transmission. The PCM system has no capabilities for parallel data transmission; however, no such requirement is projected for OLF telemeter.

(17) Telemetry Evaluation Summary. On the basis of the foregoing comments, the FM/FM system cannot be employed for the OLF because of its inability to provide high accuracy data. The PCM/FM and hybrid systems will both meet all necessary OLF requirements. In view of the anticipated system requirements, the PCM system will best meet the OLF requirements.

F-9. DESCRIPTION OF OLF CONCEPTUAL COMMUNICATIONS TERMINAL SYSTEM

The preceding discussion points out that a PCM/FM telemeter, supplemented with voice and tracking aid facilities is the most suitable communication system for the OLF. The conceptual design for this system is discussed and is depicted by a block diagram in figure 8-1. Electrical and physical

characteristics of the system are tabulated in figures F-9, F-10, and F-11. A hybrid system which includes the desirable features of each type of data acquisition system was selected as a second choice; a block diagram is shown in figure F-12, but no discussion is included.

a. Conceptual OLF Telemetry Terminal. The telemeter for the OLF will be a PCM/FM system optimized for OLO requirements. This system will utilize present state-of-the-art techniques. These techniques will yield efficient and flexible data acquisition programs, maximum transmission efficiency, and programmable accuracy; with the weight, volume, and power of the equipment held to a minimum. Reliability will be a prime requirements.

FIGURE F-9

Projected Electrical Characteristics of
OLF Telemetry System

Storage	Total Bits	Record Speed	Play Back Speed
Magnetic Digital	700 K 600 K	200 B/Sec 15 B/Sec	5 KB/Sec 40 KB/Sec
A to D Converter	Signal Range	Rate	Max. Accuracy / Code Format / Bits per Word
	100/1	80 KB/Sec	0.1% Binary 5 to 10
Multiplexer	Rate	Signal Range	Error
	8 KC	100 to 1	0.1%

FIGURE F-10

Physical Characteristics - Basic Telemetry System

Item	Assembly Number	Weight (Pounds)	Power (Watts)	Volume ($ft.^3$)
Signal Conditioning	1	15	50	0.18
Multiplexer	4	60	28	0.72
Sync Generator	1	15	7.2	0.18
Encoder	2	30	14.4	0.36
Programmer	1	15	10.2	0.18
Storage Unit (Electronics)	1	15	22.2	0.18
Decoder	1	15	14.4	0.18
Checkout and Countdown	1	15	20	0.18
XMTR (Ground)	0.5	7.5	5	0.09
XMTR (OLV)	0.5	7.5	22	0.09
Receiver (Ground)	0.5	7.5	20	0.09
Receiver (OLV)	0.5	7.5	10	0.09
Power Supplies	2	30	80	0.36
Storage Unit (Mechanical)		3	4	0.10
Antennas		20		
Cables and Connectors		30		
Totals (Basic Tele System)	16.0	293.0	307.4	2.98

FIGURE F-11

Physical Characteristics -
Additional Telemetry Equipment

Item	Assembly Number	Weight (Pounds)	Power (Watts)	Volume ($ft.^3$)
Sensors		40	40	2
Narro-band video (Transmit Only)	2	30	50	0.36
Control and display		20	20	2
Totals	2	90	110	4.36

The estimated total values of weight, size, and power for the OLF communication system including spares are summarized.

	Basic System	Spares
Weight	383 pounds	219 pounds
Size	7.34 $ft.^3$	3.99 $ft.^3$
Power	417 watts	75 watts

(1) Encoding. Straight binary coding will satisfy the OLF requirements, although the wide range of signal accuracies, signal levels, and signal information rates coupled with identification difficulties might make the use of redundant coding schemes mandatory. The binary format to be used will require a 14 bit word. Ten bits will be required for data identification, three bits for word identification, and one bit for

parity. Coding of binary data to an accuracy of 0.1 percent requires a minimum ten bit data word. The ten bits used to identify the data signal amplitudes provide readout to $1/10^{24}$ part, which provides an accuracy of better than 0.1 percent for the high level signals. The threshold of the comparator amplifier in the encoder (see figure F-13) will limit the accuracy obtainable from the low-level, 0 to 50 mv signal. This threshold limitation makes only the first eight bits significant or limits identification of the data signal to 1/256 part. This provides an encoding accuracy of 0.4 percent for the low level signals.

The output of the multiplexer will be introduced into the encoder assembly (see figure F-13). The sample and hold circuit will look at this signal and store the voltage on a capacitor through a compensated emitter follower amplifier. At the end of the sampling period, the charge stored on the capacitor will set the level of the output on the hold amplifier, which will be fed into one side of the comparator amplifier. A reset pulse will activate the digitizer circuit in the logic network, the output of which will provide a reference input to the comparator amplifier. The high and low-level switches will be synchronized through the programmer to allow the comparator amplifier to operate either on a low level 50-mv full scale voltage or a high level 5-volt full scale voltage. This automatic level and accuracy ranging will eliminate the need for a precision d-c voltage amplifier ahead of the analog-to-digital converter and will aid in minimizing over-all system errors. The comparator will have a regenerative type output circuit that decides which of the two inputs is larger. If the input from the sample and hold is larger than the input from the digitizer circuit, the comparator will not provide an output pulse. The next most

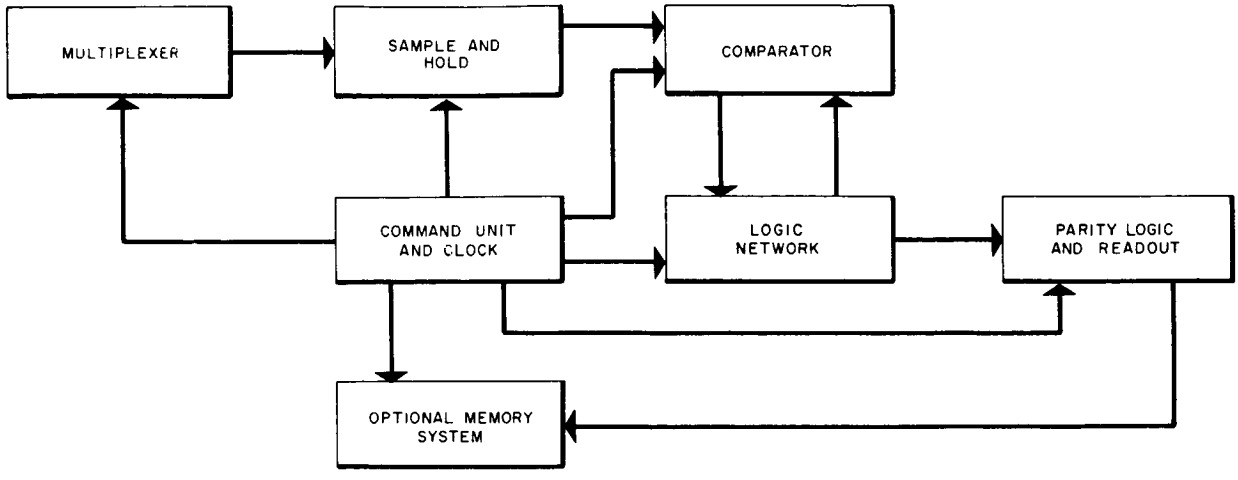


Figure F-13. Decoder System

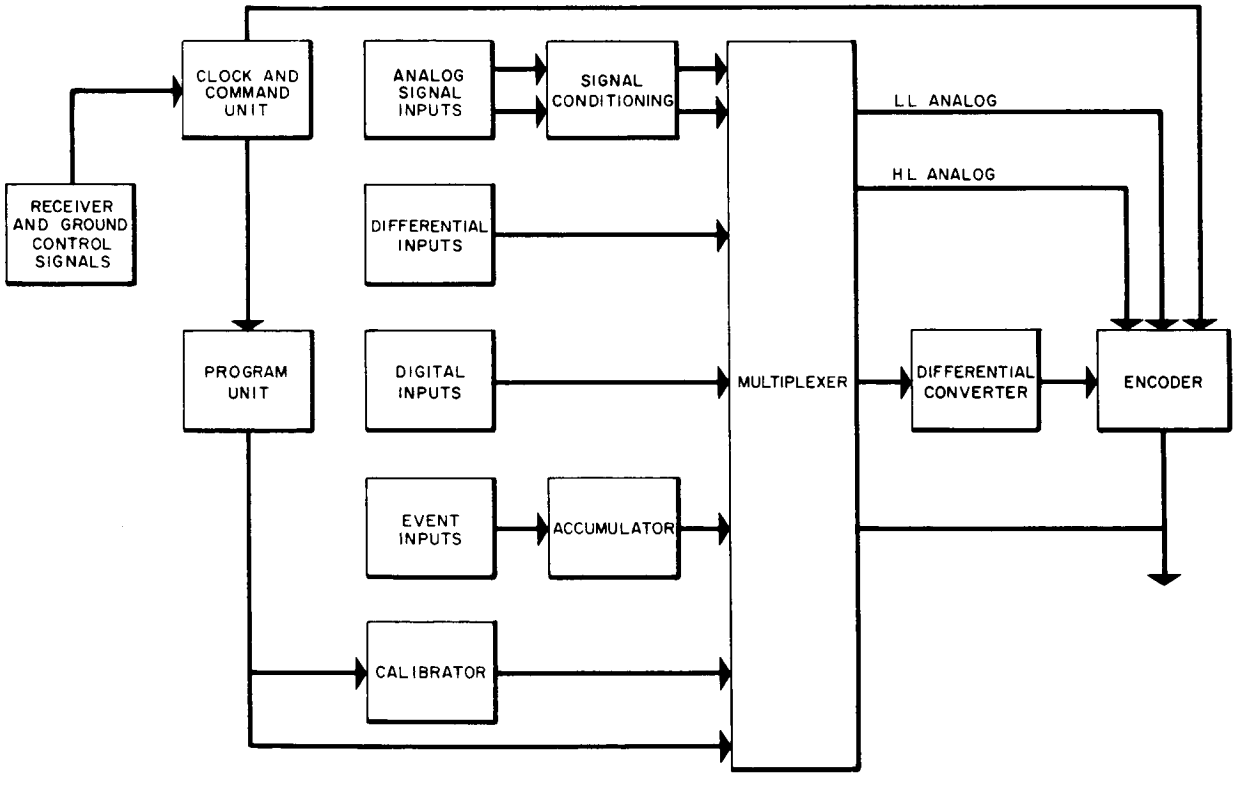


Figure F-14. Programmable Multiplexer Section

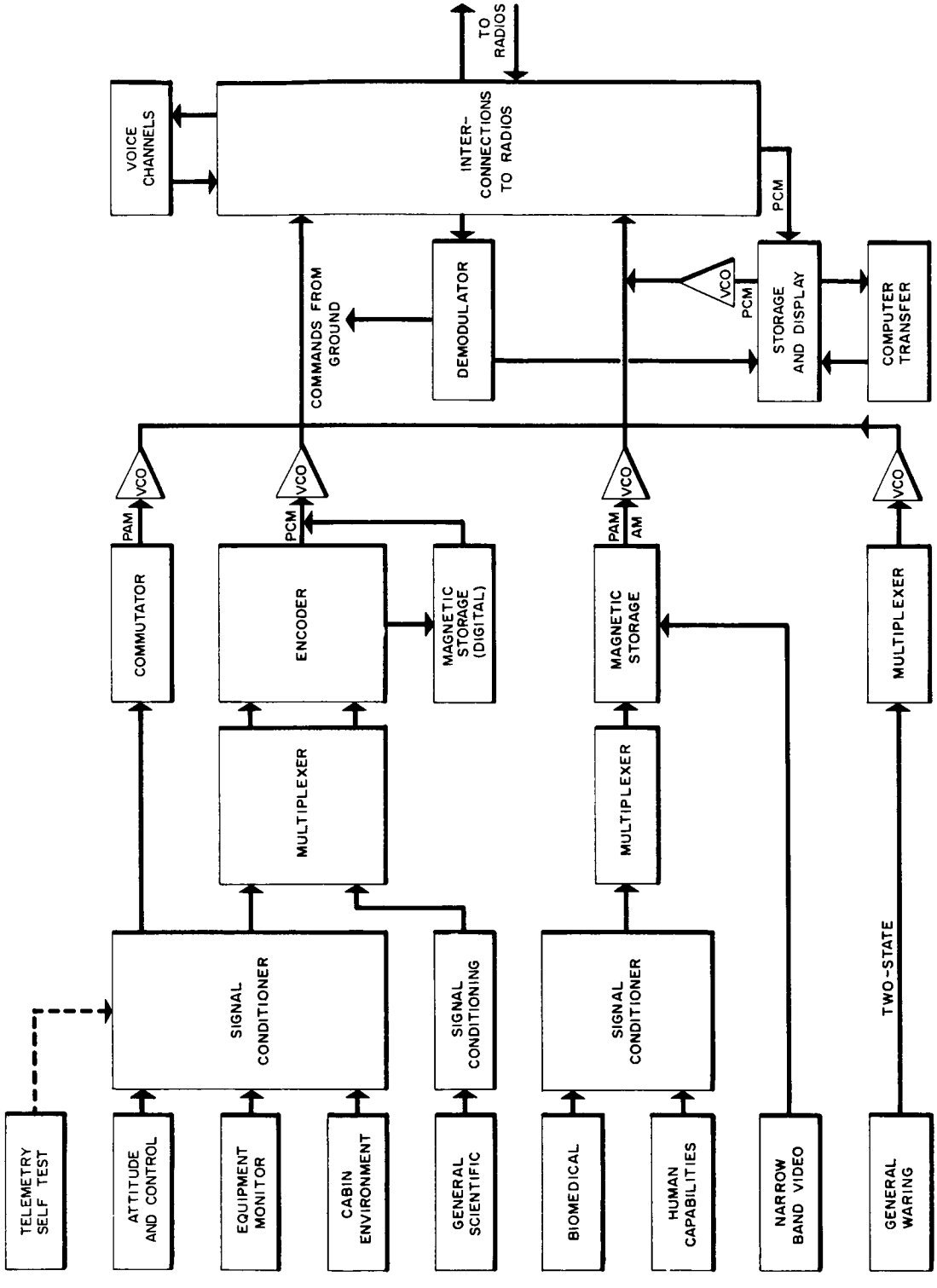


Figure F-12. Alternate Hybrid System

significant bit (1/4 full scale voltage) is then added to the most significant voltage. Should the input from the digitizer network now be larger than the input from the sample and hold, the comparator will produce a pulse output to the encoder to remove the 1/4 full scale voltage from the comparator. This process of comparing continues through the full scale of ranges: 1/8, 1/16, 1/32, and so on. Each time the comparator generates a pulse, the voltage introduced from the digitizer circuit discrete scale is removed from the comparator, and when the comparator does not generate a pulse, the discrete scale voltage is added to the input to the comparator. This is repeated for the ten bit encoding of the analog sample, and a parity bit is added in the serial pulse train from the parity logic circuit. Frame synchronization codes will be generated by a frame sync logic circuit. The frame sync will be started by a reset output from the commutators and will provide one word for frame sync after each of the data channels has been sampled.

(2) Multiplexing. It is expected that the multiplexing unit will be extremely flexible to fulfill system requirements, including those concerning a wide range of input signals, sampling rates, and operating modes. All solid-state devices will be used, and both super- and subcommutation techniques together with different rate sections will be employed.

The multiplexer will be built in sections. Each section will be programmed to operate on a predetermined sampling plan. This sampling plan may be changed on a time-base command or on an earth-to-vehicle command. The predetermined sampling plans will include a discrete number of individual channel sampling rates in any frame to make the

sampling rate compatible with the frequency content of the sample. The sampling plan will take into account data priorities and available transmission bandwidth. The multiplexer will have a low level section (0 to 50 mv) and a high level section (0 to 5V). Figure F-14 presents a block diagram of the multiplexer and the auxillary equipment associated with it.

(3) Electrical Characteristics. The multiplexer will use a basic solid-state switch consisting of two silicon surface alloy junction transistors with the collectors common. This permits the two emitters to be used as the input and output terminals. The basic switch configuration is shown in figure F-15. A square wave switching voltage E_c, will be supplied from a programmed matrix. When the switch is open, the polarity of the switching voltage E is as shown, and all junctions are reverse biased. The leakage of a back-biased collector junction is 20 times that of the emitter junction. Therefore, using the transistor with the emitter junction in series with the signal and load results in a resistance more than 20 times that of the collector junction. The magnitude of this resistance is greater than 10,000 megohms. When the switch is closed, a low impedance path will exist between the source voltage and load resistor R_L, through the saturated junctions of the two transistors. Matched transistors with minimum saturation voltages will be selected. The inverse connection will also result in a greater isolation during switch off time. The switch must also maintain voltage balance over large variations in temperature. This balance can be maintained since both transistors are made of the same material and the junctions will be located in close proximity to each other. The saturation voltage on the transistors

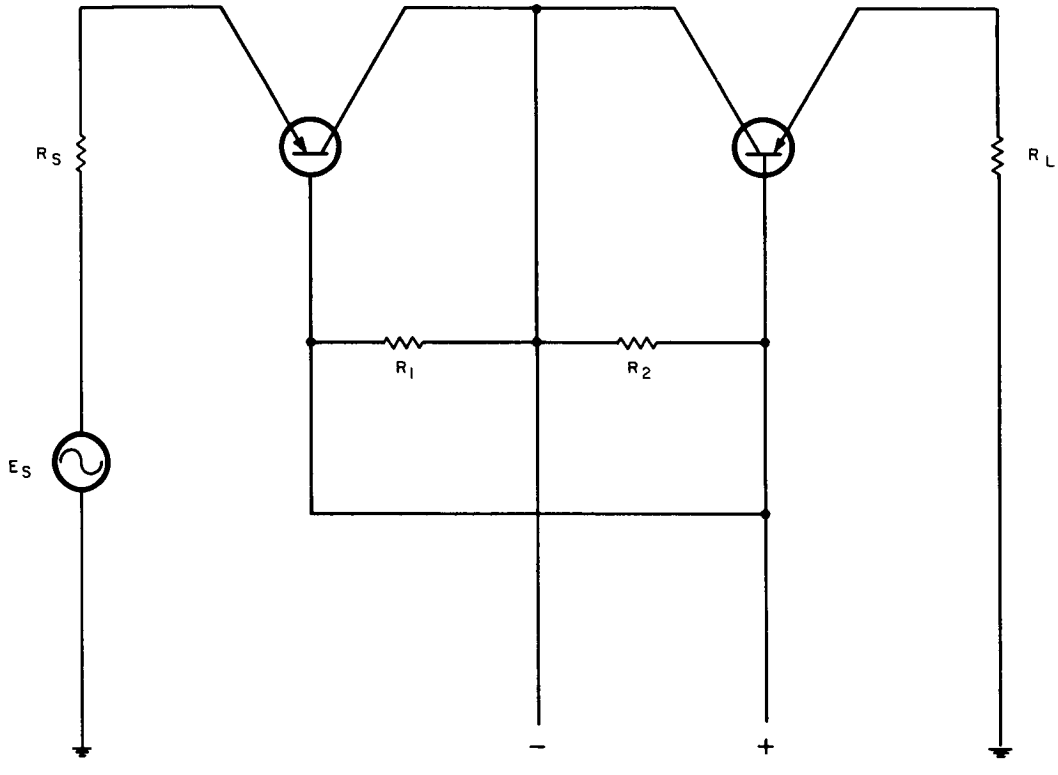


Figure F-15. Basic Switch

will be nearly equal, and the difference will be in tenths of millivolts. Consequently, for a large change in temperature the difference signal and the error signal introduced will be on the order of microvolts.

Sequencing will be provided by a programmed multiplexer driver (see figure F-16). The driver is controlled by a series of gates and a control pulse. Figure F-17 shows the pulse waveform for synchronizing and driving. Complete dc isolation must exist between the sequencing circuitry and the signal path to eliminate the chance of cross coupling. This isolation will be provided through the use of transformer coupling.

(4) Signal Conditioning. For this system a minimum amount of signal conditioning equipment is anticipated. The majority of operations in this area will be performed by automatic level and accuracy ranging in the encoder. Signal conditioning may include phase demodulators, resistive divider networks, magamps, matching networks, and that conditioning necessary to make any special sensor compatible with the telemetry measurement schedule.

Measurements with a differential output such as bridge networks will be processed through a differential converter. The converter will have a constant gain and will provide a high common mode rejection. The input to the normally a-c coupled transmitter requires special conditioning to accept a random pulse train of a non-return-to-zero format. Conditioning is necessary to regenerate pulse train signals having varying average d-c levels into pulse train signals with a constant d-c reference level. The transmitter will then see the "go" or "no go" condition of the last bit, and no dc

shift will be experienced by the transmitter when a nonsymmetrical pulse train is applied. This go no-go feature results in polarity synchronization of the transmitter with respect to the input signal from the encoder in the event of an interrupted pulse train or loss of power.

(5) Storage. The OLF will provide storage for four types of data: human capabilities and bio-medical information, picture images, scientific data and checkout and countdown information. It is anticipated that only periodical monitoring of human capabilities and bio-medical information will be required after Phase I in view of knowledge gained during Phase I and previous programs. The results from these periodical tests will be recorded on a controlled, continuous record, magnetic tape. The operator controlling the test will likewise control the record time and rate. The picture image data will also be recorded on this tape. Scientific data, for the most part, will change slowly with respect to time. This data will be converted to binary format and stored on a magnetic tape with a stepping-type transport which advances upon command. Playback for both recorders will be programmed for relaying stored information to preselected ground stations. Maximum storage time will be 24 hours. Checkout and countdown information will be transferred directly into and from the OLF checkout computer.

(6) Self-check Features. Since a malfunction in the OLF telemetering equipment could generate extraneous signals which have the same characteristics as measured information, some means must be provided to allow the ground equipment to distinguish failures. By periodically transmitting a set of properly selected self-check signals, a ground station can detect a malfunction and isolate the fault to the subassembly

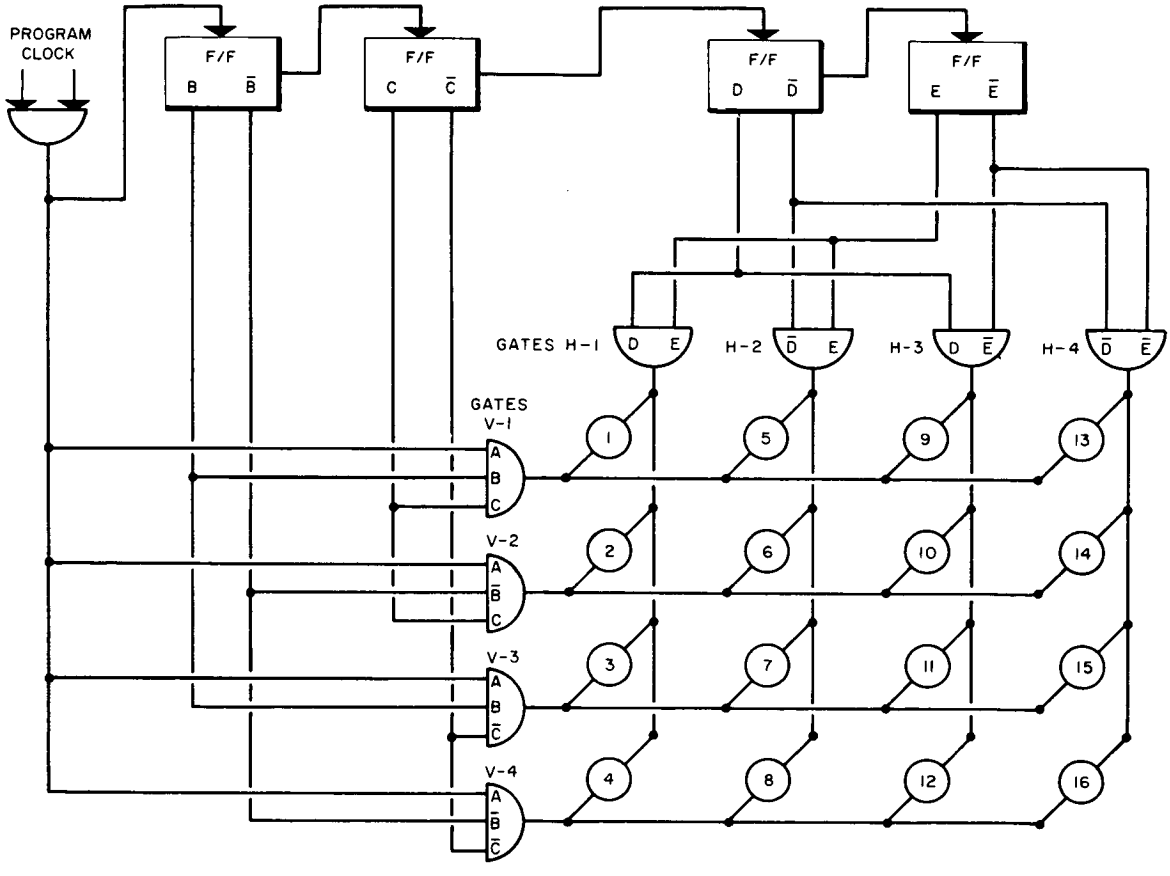


Figure F-16. Programmed Multiplexer Driver

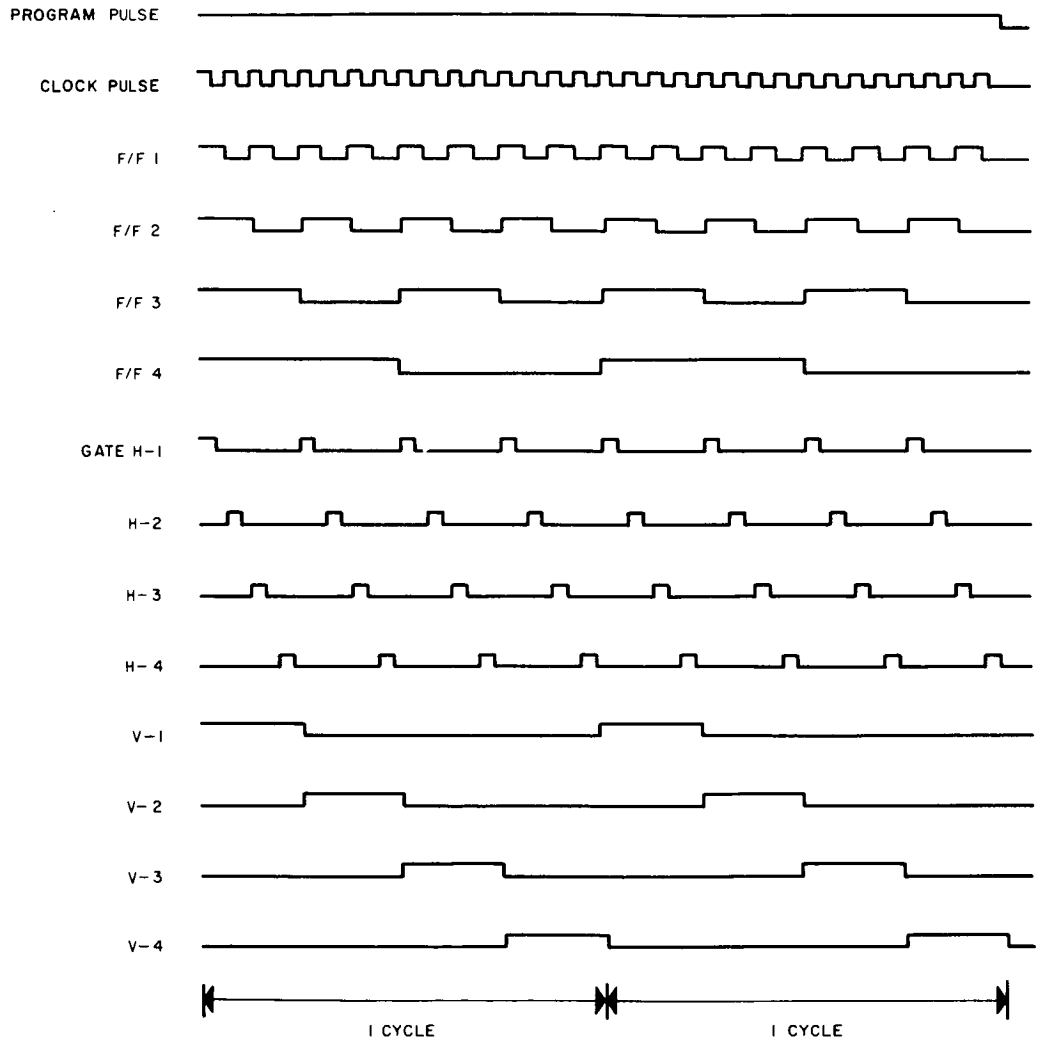


Figure F-17. Waveforms of Driver Unit

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VOLUME III

NAVIGATION, GUIDANCE AND RENDEZVOUS

PREPARED FOR
MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

PREPARED BY: SPERRY GYROSCOPE CO., GREAT NECK, N.Y.
AIR ARMAMENT DIVISION
ADVANCED SYSTEMS
HUMAN FACTORS

SPERRY RAND CORPORATION
SPERRY RAND SYSTEMS GROUP
GREAT NECK, NEW YORK

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TABLE OF CONTENTS

Section		Page
I	SUMMARY	1-1
II	OPERATIONAL CONCEPTS	2-1
	2-1. General	2-1
	2-2. Mission Objectives	2-2
	2-3. Influencing Factors	2-2
	2-4. Effect of Influence Factor on Guidance	2-5
	2-5. Ground Rules Concerning Vehicle Configurations	2-5
	2-6. Operational Variables Affecting Guidance	2-6
	2-7. Families of Operational Concepts	2-10
	2-8. Sequence of Operations	2-10
	2-9. Guidance Functions Required for Each Vehicle	2-20
III	RECOMMENDED GUIDANCE SYSTEMS	3-1
	3-1. General	3-1
	3-2. Recommended Guidance System for the Payload	3-3
	3-3. Recommended Guidance System for the OLF	3-9
	3-4. Recommended Guidance System for the Tug/Ferry Vehicle	3-15
	3-5. Spacecraft Guidance System	3-24

TABLE OF CONTENTS (Cont)

Section		Page
IV	MAN-MACHINE RELATIONS DURING RENDEZVOUS	4-1
	4-1. Introduction	4-1
	4-2. The Selection of Guidance Laws	4-2
	4-3. Allocation and Man-Machine Tasks	4-4
	4-4. Interface Display Configurations	4-8
V	PROGRAM CONSIDERATIONS AND RECOMMENDATIONS	5-1
	5-1. General	5-1
	5-2. Recommendations	5-3
	5-4. Formulate Detailed Development Plan	5-4
	5-5. Specifications for Multipurpose Radar, Manual Star Tracker on Platform, and Unmanned Payload Guidance	5-6
	5-6. Integrate Breadboards Into Operational Simulator	5-6
	5-7. Cost and Schedule Estimates	5-7
Appendix		
A	GUIDANCE ACCURACY REQUIREMENTS	A-1
	A-1. Philosophy	A-1
	A-2. General	A-2
	A-3. OLF Requirements	A-3
	A-4. Payload Requirements	A-6
	A-5. Tug Requirements	A-8
	A-6. Spacecraft Requirements	A-10

TABLE OF CONTENTS (Cont)

Appendix		Page
B	RELIABILITY REQUIREMENTS AND CONSIDERATIONS	B-1
	B-1. General	B-1
	B-2. Reliability Requirements	B-3
	B-3. Relative Reliabilities of Guidance Elements	B-12
	B-4. Estimate of Absolute Reliability	B-15
	B-5. Reliability Growth	B-18
C	GUIDANCE SYSTEM ELECTION	C-1
	C-1. General	C-1
	C-2. Guidance for the Payload	C-2
	C-3. Guidance System Selection for the OLF	C-14
	C-4. Guidance for the Tug/Ferry Vehicle	C-29
D	EQUIPMENT CONCEPTS	D-1
	D-1. General	D-1
	D-2. Reliable Multipurpose Radar for Determining Altitude, Vertical, and Line-of-Sight	D-2
	D-3. Inertial-Radar Mixing for Lunar Spacecraft Terminal Approach	D-17
	D-4. Manual Celestial Tracker	D-20
	D-5. Beacon Navigation	D-24

LIST OF ILLUSTRATIONS

Figure		After Page
2-1	General Characteristics of Space Vehicle	2-5
2-2	Saturn C Configurations	2-5
2-3	Program Plan - Ground Launch Vehicles	2-5
2-4	Booster - Payload Configuration, NO OLF Case	2-5
2-5	Booster - Payload Configuration, OLF Case	2-5
2-6	NO OLF Concept, Uses Two Rendezvous	2-5
2-7	OLF Concept Uses Four Rendezvous	2-5
3-1	Payload Guidance System, Block Diagram	3-4
3-2	Guidance System Location	3-5
3-3	Sensor Package Layout	3-6
3-4	OLF Guidance System (Primary Mode of Operation)	3-9
3-5	Primary and Secondary Redundancy for the OLF Guidance System	3-11
3-6	Tug/Ferry Guidance System (Primary Mode of Operation)	3-17
3-7	Primary and Secondary Redundancy for the Tug/Ferry Guidance System	3-20
3-8	Multiple Image Optics - One Concept	3-32
3-9	Spacecraft Guidance System, (Primary Mode of Operation)	3-33
3-10	Primary and Secondary Redundancy for the Spacecraft Guidance System	3-35
3-11	Spacecraft Guidance Reliability versus Added Weight (Redundancy) (Tradeoff)	3-36
4-1	Reliability Model for System Including Human Operator	4-5

LIST OF ILLUSTRATIONS (Cont)

Figure		After Page
4-2	Operator Loading Analysis	4-8
4-3	Rendezvous Trajectory and Vehicle Control Capability	4-10
4-4	Schematic of Projected Display	4-11
4-5	Projective Display Presentation	4-12
4-6	Projective Display with Vehicle Tracking Target	4-13
4-7	Instrument Panel Display	4-13
4-8	Moving Base Simulator with Projected Visual Field	4-14
5-1	Development Program Schedule	5-4
5-2	Cost Estimate	5-7
5-3	Project Schedule - Development Program for Navigation and Guidance System	5-7
B-1	Complexity Diagrams of the OLO Radar Family	B-13
B-2	Complexity Diagrams of the OLO Tracker Family	B-13
C-1	Payload Guidance System Reliability versus Added Weight for Redundancy (Tradeoffs)	C-13
C-2	Transmitter Average Power versus Range for Varying Integrations (Non-Cooperative Case) and Beacon On	C-21
C-3	OLF Guidance System Reliability versus Added Weight for Redundancy (Tradeoff)	C-27
C-4	Tug/Ferry Guidance System Reliability versus Added Weight for Redundancy (Tradeoff)	C-35
D-1	Conical Scan Geometry for Verticality	D-10

LIST OF ILLUSTRATIONS (Cont)

Figure		After Page
D-2A	Video Time/Amplitude Plot	D-10
D-2B	Verticality - Functional Diagram	D-10
D-3	OLF Radar, Basic Block Diagram	D-10
D-4	OLF Radar, Function Flow Diagram	D-10
D-5	Beacon Block Diagram	D-16
D-6	Optical Paths, Manual Tracker	D-22
D-7	Manual Tracker, Celestial Field of View Transmission	D-22
D-8	Field of View of Observer	D-26
D-9	Ground Beacon Geometry	D-26
D-10	Effect of Vehicle Motion	D-26
D-11	OLF Navigation Radar	D-29
D-12	Ground Beacon Ranging Geometry for Navigation	D-29
D-13	Integration of Navigation Radar with Multipurpose Radar	D-29

LIST OF TABLES

Table		Page
2-1	Operational Concepts	2-11
2-2	Guidance Functions - NO OLF Concept	2-21
2-3	Guidance Functions - OLF Concept	2-22
3-1	Systems Element Performing Required Functions	3-7
3-2	System Characteristics	3-7
3-3	Payload Component Accuracy Specifications	3-8
3-4	Primary and Redundant Modes of Operation for the OLF Guidance System following	3-9
3-5	OLF System Characteristics	3-12
3-6	OLF Component Accuracy Specifications	3-15
3-7	Primary and Redundant Modes of Operation for the Tug/Ferry Guidance System following	3-18
3-8	Tug/Ferry Component Accuracy Specifications	3-24
3-9	Primary and Redundant Modes of Operation of the Spacecraft Guidance System following	3-29
3-10	Spacecraft System Characteristics	3-39
3-11	Spacecraft Component Accuracy Specifications	3-40
4-1	System Reliability Breakdown	4-7
B-1	Guidance System Operating Time - NO OLF Concept (Earth Launch to Orbital Launch)	B-5
B-2	Guidance System Operating Time - OLF Concept With No Tug Operation (Earth Launch to Orbital Launch)	B-6
B-3	Guidance System Operating Time - OLF Concept With Maximum Tug Operation (Earth Launch to Orbital Launch	B-7
B-4	Ranking of Sensor Complexity - A Measure of Relative Reliability	B-15

LIST OF TABLES (Cont)

Table			Page
C-1	Payload System Functions		C-5
C-2	Candidate Guidance Systems for Payload	following	C-11
C-3	Candidate Systems for Orbit Determination	following	C-17
C-4	Candidate Elements for Life of Short Determination	following	C-20
C-5	Candidate Guidance Systems for OLF	following	C-23

SECTION I

SUMMARY

The Sperry Rand Corporation is investigating the major astrionic requirements for Orbital Launch Operations. The subject of this volume is Navigation, Guidance and Rendezvous, which covers the work done by the Air Armament Division of Sperry Gyroscope Company, a division of Sperry Rand Corporation.

The principal conclusions of this work are

a. The Navigation, Guidance and Rendezvous aspects of the OLO program are within the time scale for state-of-the-art development as conceived by Sperry Rand. No major technological breakthroughs are necessary to successfully establish OLO operations.

b. Certain follow-on work must be started within the next few months if operational rendezvous capability by the target date of the second quarter of 1965 is to be met. In the area of Navigation, Guidance and Rendezvous, these are as follows:

(1) Formulate a detailed development plan for the OLO vehicles' guidance system. This plan should have reliability as its primary objective, and should aim for demonstrated reliability of each critical guidance element, by successful orbital flight tests, before OLO operations begin.

(2) Prepare specifications and develop breadboards of the critical portions of the essential elements of the guidance systems, specifically, a multipurpose radar; a manual star tracker mounted on an inertial platform; and the recommended guidance system for the unmanned payloads.

(3) Integrate the breadboards of the critical guidance elements into an operational simulator, to optimize the man-machine relationships.

c. Some of the major technical conclusions which are of interest to the Marshall Space Flight Center as an aid in planning the mentioned work are as follows:

(1) The accuracy requirements of the OLO guidance systems can be met by equipment which is available or which can be designed using existing components and techniques.

(2) Reliability must be a major objective throughout the development, manufacture and test of the guidance equipment for OLO.

(3) Two specific guidance elements are required for OLO systems:

(a) A multipurpose radar for measuring altitude, vertical, angle to a rendezvous target, range and range rate; reliability is the primary goal in developing this radar.

(b) A manual star tracker mounted on an inertial platform. This is essentially an automatic star tracker in which the sensor and signal processing electronics are replaced by the astronaut's eye.

(4) The primary mode guidance system recommended for the manned vehicles in OLO consists of the two guidance elements just described, plus a digital computer and displays. Sufficient redundancy is added to attain the required reliability with full capability, plus additional back-ups with reduced capability.

(5) The same basic elements of the system will be used for all of the manned vehicles in the Orbital Launch Operation. This will allow considerable economies in development, production, training and logistics, compared to the cost if different guidance systems were used in the different manned vehicles.

(6) The OLO operational concepts are not bounded by guidance. The recommended system can accommodate any of the probable operational concepts discussed in Section II — existence or non-existence of an OLF, thrust capability of payload, etc.

(7) The guidance system recommended for unmanned payloads consists of a computer, three gyros, three accelerometers, and a horizon tracker.

SECTION II

OPERATIONAL CONCEPTS

2-1. GENERAL

This section establishes the basis for generating the requirements and functions for the prescribed guidance system for the forthcoming manned lunar missions. This basis of generating the guidance system concept consists of,

- Establishing the mission objectives.
- Postulating several operational concepts that would lead to a successful translunar injection of the manned lunar capsule.
- Selecting some of the concepts for detailed investigation.
- Identifying the required guidance functions.

The general philosophy applied in this study is to define families of operational concepts within which the "best" operation will probably be realized. To carry this study further and attempt to define this "best" operational concept would require studies and decisions from many fields, of which guidance is only one. In the discussions that follow, an attempt is made to remove many of the difficult "what if?" considerations and concentrate on the problem areas that are pertinent to guidance and are common to several concepts. Having defined the guidance functions for these basic concepts, the guidance functions and requirements are applicable to any of the related operating concepts in these families.

2-2. MISSION OBJECTIVES

The objectives of the lunar missions described in this study are as follows:

- To place man on the moon before the end of this decade.
- To establish a lunar base.
- To accomplish these objectives with a high regard for human safety.

2-3. INFLUENCING FACTORS

The factors influencing the selection of a specific operational concept for accomplishing these objectives run the gamut from gross national product considerations to those dealing with man's capabilities in space. Many of these factors are intangible; however, several others can be directly related to the configuration of the guidance system. Of these, four of the most important are:

- Probability of mission success
- Human safety
- Rocket booster development
- Demonstrated space rendezvous involving man.

a. Probability of Mission Success (P_{MS})

A high probability of mission success can only be assured by diligent testing of equipment and training of the men both on the ground and in space. In addition to the expense involved in conducting these tests, the time factor looms very important when national prestige is involved.

Assuming certain risks (not human safety), a decision may be made to attempt a manned lunar shot at a level of P_{MS} that is just barely acceptable to an earlier time. Without elaboration, P_{MS} can be considered variable and dependent on several of the intangible influence factors.

b. Human Safety

In many respects human safety can be treated as a variable similar to P_{MS}, except that the acceptable lower limit is much higher. This is accomplished by including a highly reliable abort capability so that in the event of mission failure, the astronauts have a high chance of being returned safely to earth.

c. Rocket Booster Development

This certainly is one of the pacing items in space venture that affect almost all other space technologies. The time table, P_{MS} and types of lunar missions are contingent on successful booster development of the Saturn C series and the giant Nova. Current planning indicates that only the Nova is capable of a direct manned payload launching to the moon. The Nova, unfortunately, is not slated for operational use for several years, and planners have turned to the earlier available Saturn vehicles to fulfill the mission using an Orbital Launch Operation (OLO).

This approach would require one or more rendezvous of Saturn C-4 or C-3 vehicles in a low (200-500 nm) near circular parking orbit. The concept of rendezvousing with several vehicles to fuel or assemble the space vehicle for its lunar flight is indeed complex and requires a marshalling of the country's aerospace industry and university talents.

Studies indicate that either two C-4 or four C-3 vehicles are sufficient to put the payload into the required 300 nm earth orbit for launching a manned lunar landing mission. From a guidance system point-of-view, it does not matter significantly whether four C-3 or two C-4 vehicles are used in the space vehicle buildup since the guidance functions to be performed are similar in both cases.

The delivery of the three man crew to orbit could be performed by the Saturn C-4 or C-3 vehicles provided the men have a high chance of survival. Otherwise, the Saturn C-1 vehicle can be used as the ferry since by 1965-1966 the reliability of this vehicle should be sufficiently high.

d. Demonstrated Space Rendezvous

The decision to utilize the rendezvous technique for lunar landing rests mainly with man's ability to control a space rendezvous of two large vehicles. In addition to the hazards of docking, man must perform certain tasks such as assembly and repair of the space vehicle or transfer of fuel from tank to spacecraft in a hostile and exotic environment.

Basic questions to be answered are: what degree of inspection, checkout and other preparations for space vehicle launching will be done in space; what role would the ground facilities play, and the time duration of the entire operation? Answers to these questions would define the burden on the astronaut and determine whether some type of an orbital launch facility (OLF) is required to ease the burden of the spacemen.

The OLF could provide uncramped sleeping quarters for the men, a pressurized repair, checkout and assembly

volume, sufficient tools and spare parts for the mission and numerous other tasks. These include scientific experimentation and possibly assisting the deep space net in tracking the space vehicle on its trip to or from the moon, and aiding in rescue operations.

The existence of one or more such facilities is certainly desirable. Whether or not this facility can be operational soon enough to speed the initial lunar landing program requires definition.

2-4. EFFECT OF INFLUENCE FACTORS ON GUIDANCE

The impact of these influence factors on the configuration of the guidance system is to broaden the scope of investigation to include several candidate concepts of operation. The reasoning for this is that to date the intangible factors have not been clearly defined and the booster and rendezvous technologies are just now being investigated. The only factor on which there can be no compromise is human safety.

2-5. GROUND RULES CONCERNING VEHICLE CONFIGURATIONS

Figures 2-1 through 2-7 illustrate the vehicle characteristics and operational concepts which have been assumed for this discussion. Figure 2-1 indicates the general characteristics of the space vehicle which is injected into a translunar trajectory, while figure 2-2 shows the varous Saturn C booster configurations. Figure 2-3 is the launch program plan. Figure 2-4 shows the booster and payload configuration for orbital buildup using NO OLF. Figure 2-5

shows the booster and payload configuration for orbital build-up with an OLF. Figures 2-6 and 2-7 show one possible arrangement of vehicles during orbital rendezvous in the NO OLF and OLF cases, respectively.

This study assumes that two-stage C-4 boosters will be used in the buildup of the space vehicle, and the three man crew will be delivered into orbit by a tug or ferry powered by a C-1 booster. The tug and ferry are assumed to be similar vehicles insofar as guidance is concerned. Referring to figure 2-3, this implies operation late in 1966 and thereafter. In the event that launchings are contemplated late in 1965, the two or three stage C-3 boosters may be utilized. This will require a greater number of rendezvous and docking operations, so that the reliability of each rendezvous must be greater if the probability of success of the overall mission is to remain the same.

The term "Payload" is used in this discussion to denote an unmanned package which is injected into orbit, and which may contain fuel or stores, or may be a part of a vehicle for injection into translunar orbit. A payload may be powered or unpowered. It is assumed that the boost and injection of payloads into orbit will be under the control of the ground. The OLF in orbit will take over control of the approaching payload after injection of the payload into orbit. The exact point at which the OLF takes over control will depend upon how close the upcoming payload is placed to the OLF. It is assumed that the OLF will monitor the upcoming payload and be prepared to assume control.

2-6. OPERATIONAL VARIABLES AFFECTING GUIDANCE

There are three operational variables which affect guidance,

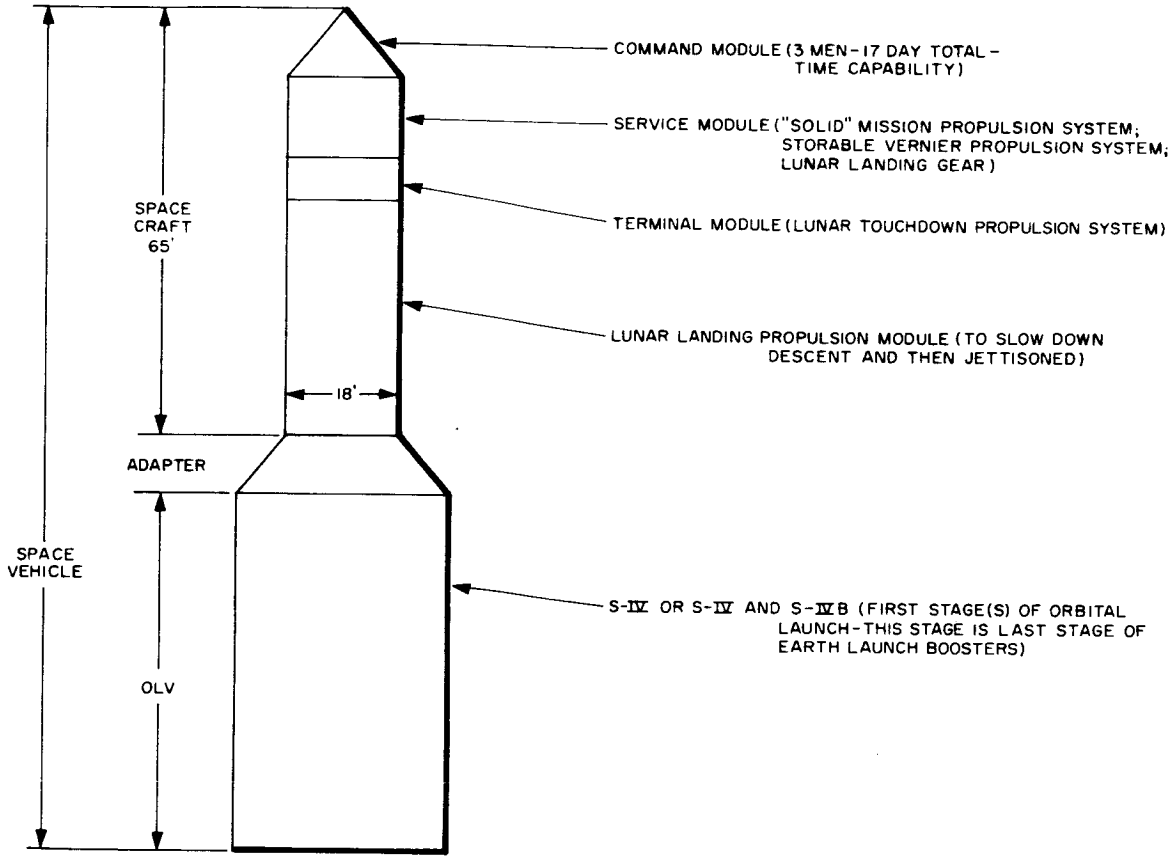


Figure 2-1. General Characteristics of Space Vehicle

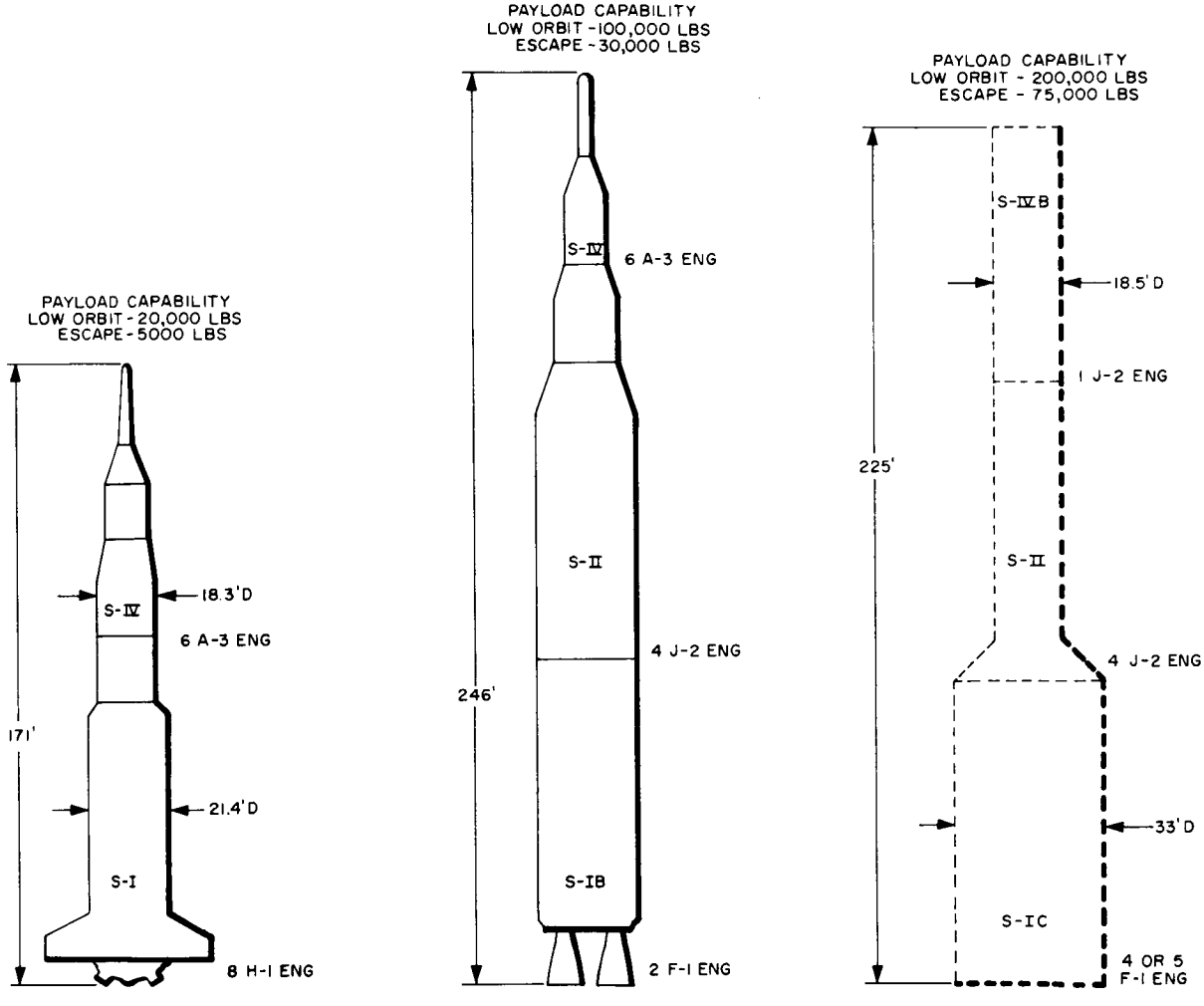


Figure 2-2. Saturn C Configurations

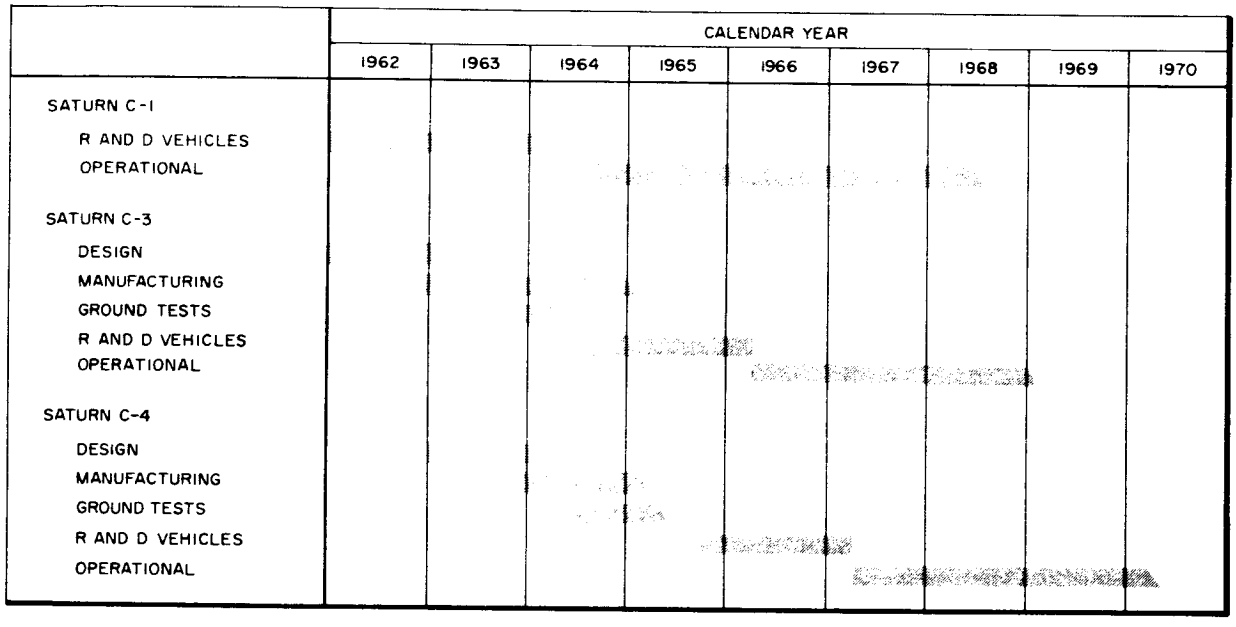
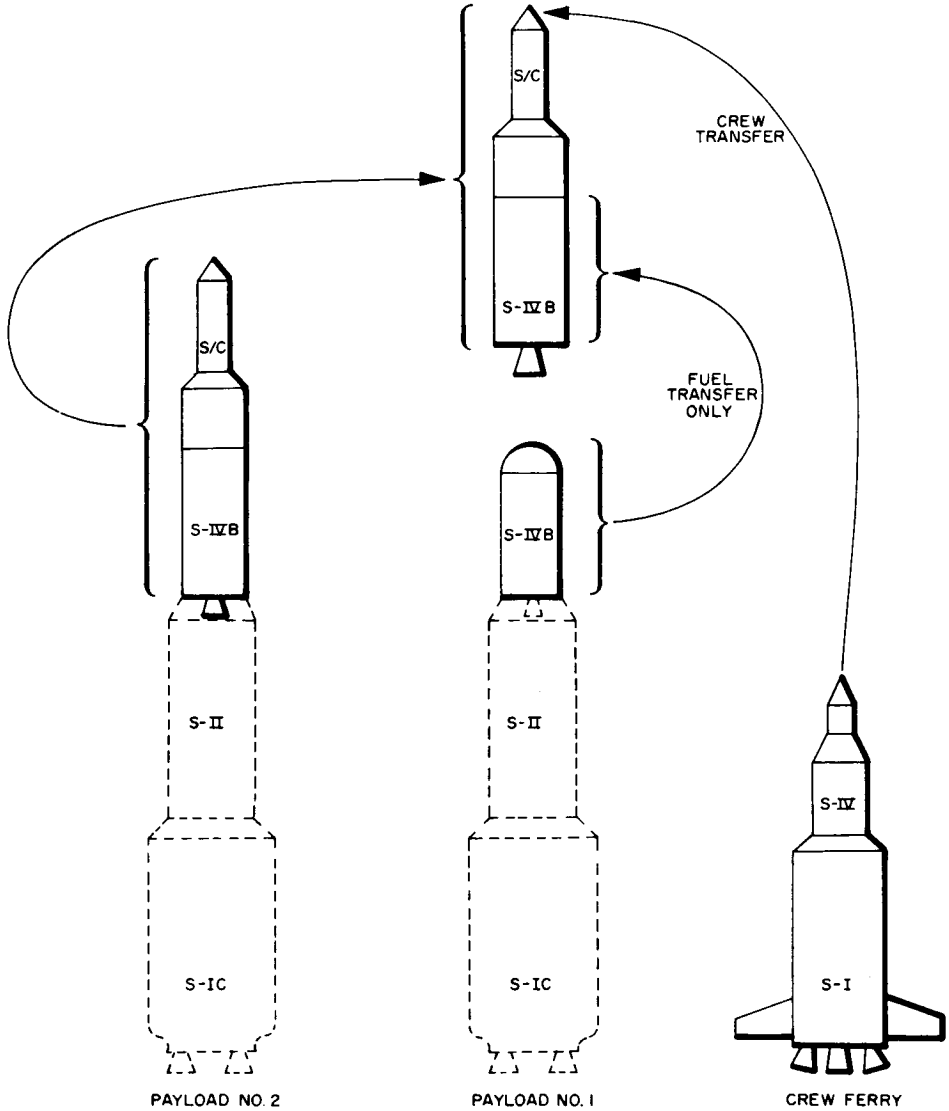


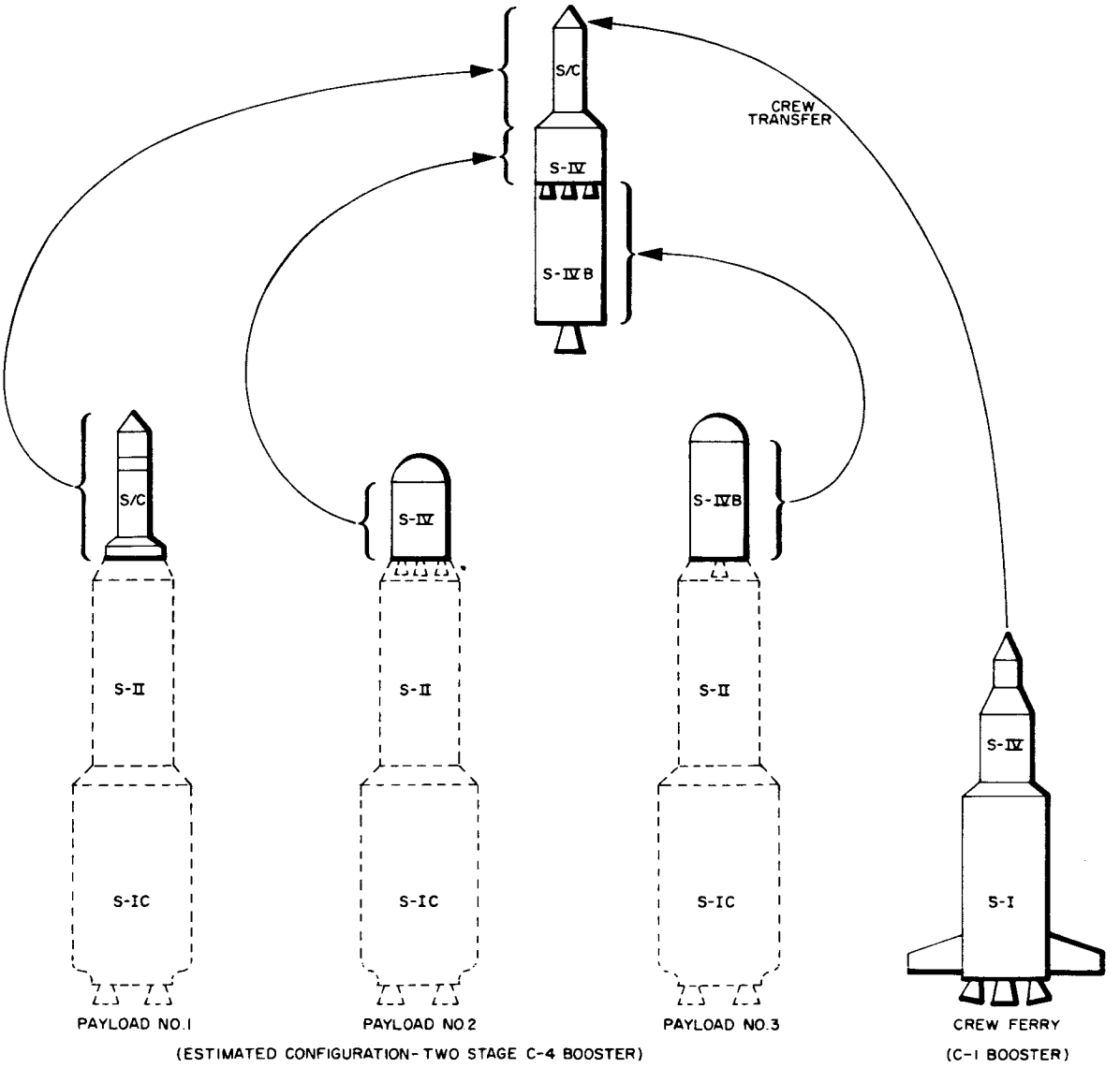
Figure 2-3. Program Plan - Ground Launch Vehicles



WEIGHT STATEMENT FOR PAYLOADS

P/L NO.1 S-IVB W/O ENGINES (61% FUELED)	142,834 LBS
P/L NO.2 SPACECRAFT, ADAPTER AND S-IVB (11% FUELED)	142,833 LBS
SPACECRAFT (ALL CRYOGENIC PROPELLANT SYSTEM)	96,040 LBS
SPACE VEHICLE AT ORBITAL LAUNCH	265,000 LBS

Figure 2-4. Booster-Payload Configuration, NO OLF Case



WEIGHT STATEMENT FOR PAYLOADS

P/L NO.1 SPACECRAFT AND ADAPTER	144,500 LBS
P/L NO.2 S-IV (70% FUELED)	81,700 LBS
P/L NO.3 S-IVB (66% FUELED)	155,805 LBS
SPACECRAFT (STOREABLE SOLID AND CRYOGENIC PROPELLANT SYSTEM)	143,300 LBS
SPACE VEHICLE AT ORBITAL LAUNCH	382,005 LBS

Figure 2-5. Booster-Payload Configuration, OLF Case

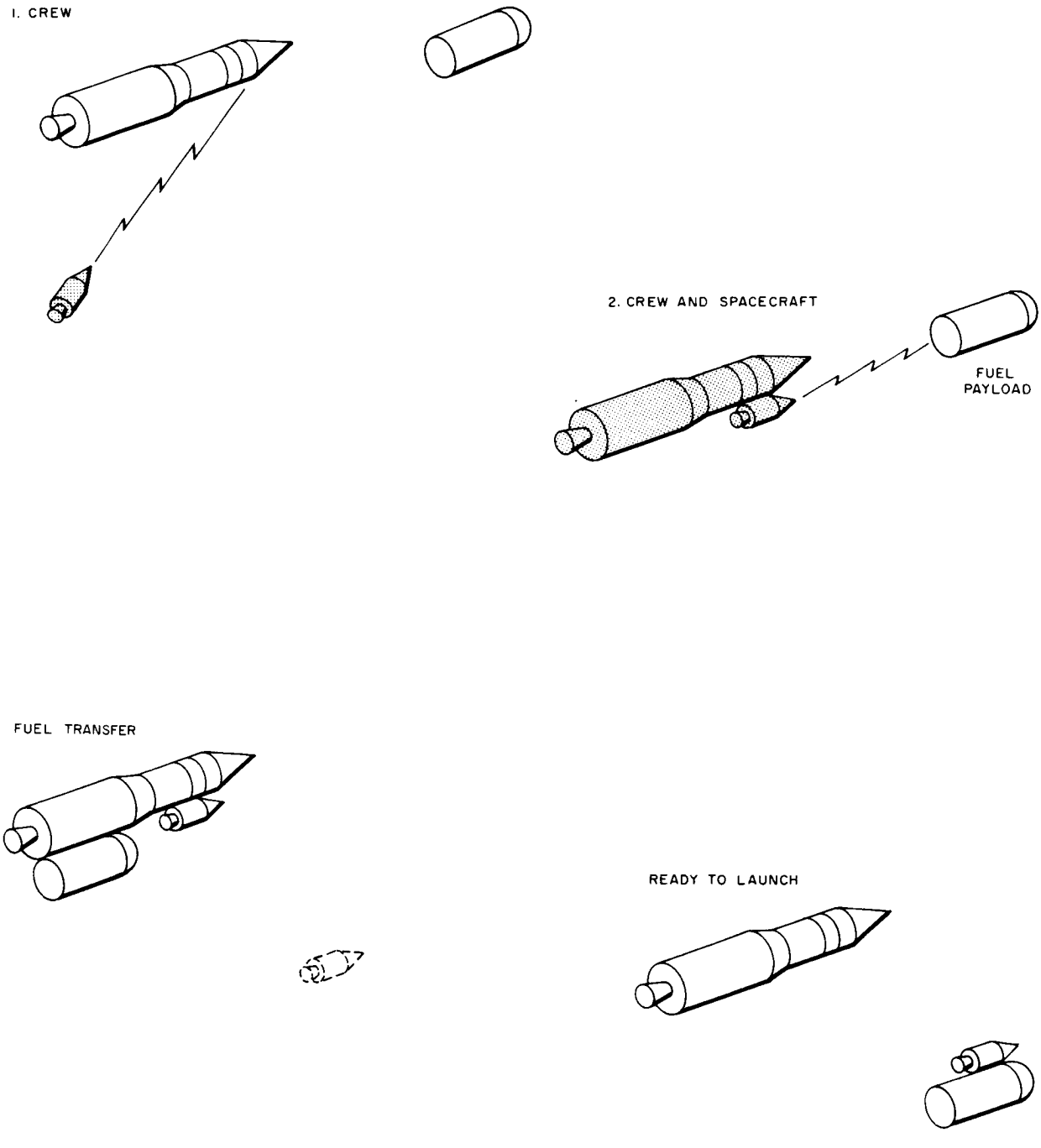


Figure 2-6. NO OLF Concept, Uses Two Rendezvous

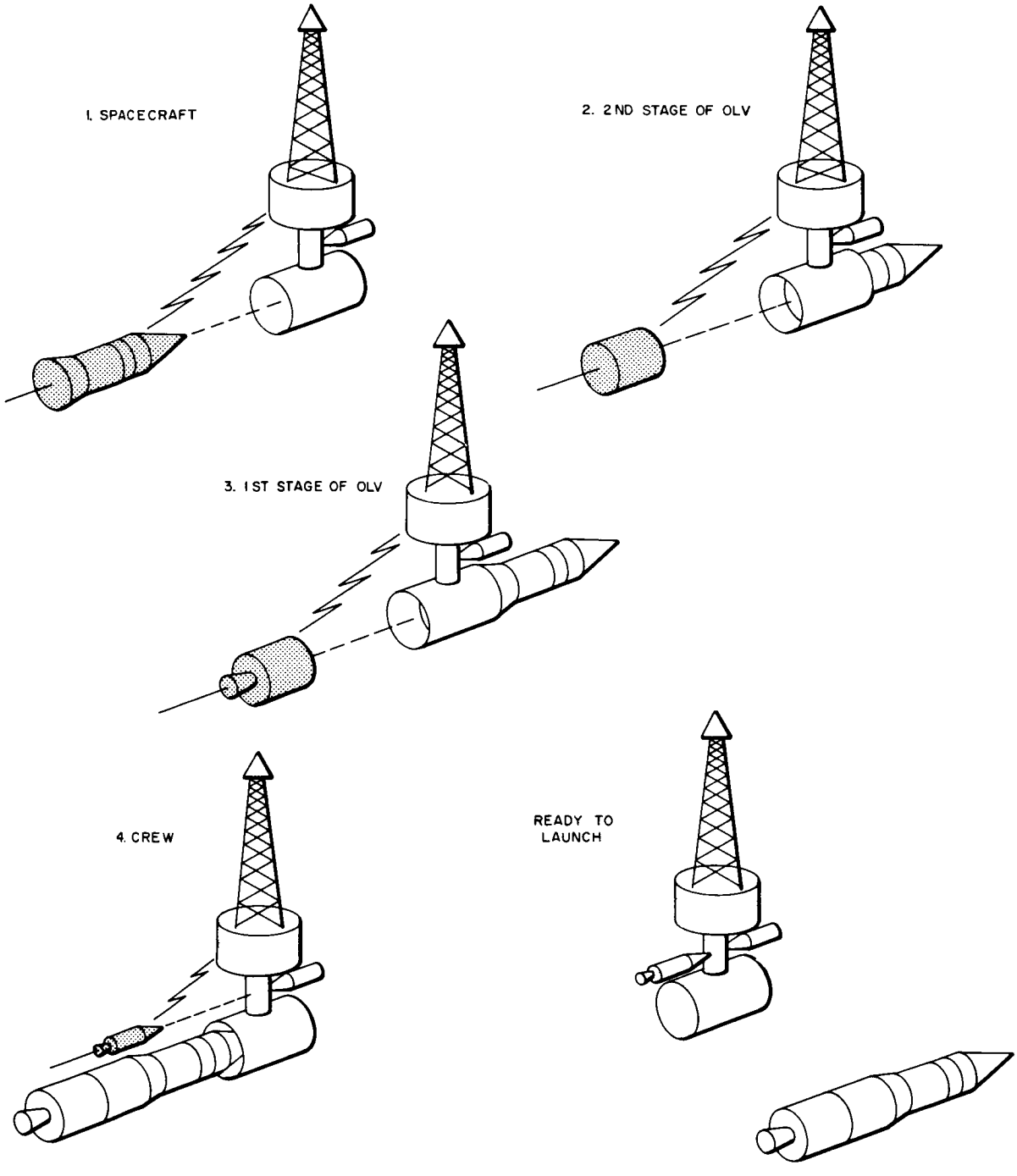


Figure 2-7. OLF Concept Uses Four Rendezvous

- The number of rendezvous dockings which must be accomplished
- The thrust capability of the payloads
- Allowable earth launch time deviations.

When each of these three variables is fixed, the operating concept is largely specified, insofar as guidance is concerned. The following paragraphs indicate the ranges of these variables within which the best operating concept will probably lie.

a. Number of Required Dockings

The existence of an OLF will increase the probability of success of any given docking operation because the OLF can provide more spares and tools and a more hospitable environment than the NO OLF environment. As the number of docking operations increases, the need for an OLF becomes more acute. It has been arbitrarily assumed in this study, therefore, that an OLF will be required if the number of required dockings is greater than two.

The number of dockings required for a mission serves to identify two families of operational concepts, those which include an OLF, and those which have NO OLF.

b. Payload Thrust Capabilities

The thrust capability of the payloads can be divided into three categories,

- Sufficiently fine vernier control so that the payload may be remotely commanded to perform the docking maneuver.

- Gross thrust capability, wherein the payload can be commanded to within a few miles of the OLF or spacecraft, but can not complete the docking maneuver using its own engines.
- No thrust capability.

The docking maneuver is assumed completed when all relative motion of the two rendezvous vehicles has been nulled out and some kind of physical connection has been accomplished.

c. Launch Time Deviations

Deviations in launch time cause two kinds of problems. First, the launching point on earth moves away from the orbital plane of the target, and, secondly, the target moves along its orbit, away from its planned location at the ideal instant of launch.

The first effect requires that the orbit of the ascending payload or other vehicle be rotated to place the vehicle in the plane of the target's orbit. This is feasible if the launching deviates only a few minutes from the ideal time of launch. If the payload launching is delayed more than eight minutes, however, the launch plane may be more than two degrees from the target orbital plane, and a weight penalty of more than ten percent will be required to place the payload orbit in the target plane. Launch time deviations greater than eight minutes will therefore be considered aborts.

The second effect is significant only if the target (OLF or OLV) is in a synchronous orbit, such that when the launching point on earth passes through the target orbital plane, the target will be in the proper position so

that a payload can be launched directly to it. This requires that the payload be launched within a time period of a few seconds, because the target is moving along its orbit, relative to the launching point on earth, at a speed of the order of five miles per second. For example, if the launching is delayed from the ideal time by two seconds, the payload will still be launched in a plane very close to the orbital plane of the target, but the target will have moved approximately ten miles away from the point of arrival of the payload.

When the payload arrives in the orbit of the OLF, but there is a significant phase difference, either because the launching was delayed or because the OLF is not in a synchronous orbit, this difference can be made up by injecting the payload into a phasing ellipse instead of circularizing its orbit when the correct altitude is reached. The character of the phasing ellipse would be dependent on the amount of phase angle to be made up at time of injection. If the target vehicle ephemeris is well known prior to launch, a simple stored program could be inserted into the injection program to select a ΔV for injection that is dependent on the launch time deviation (that is, phase angle). The maximum miss distance that can be closed in one orbital revolution, assuming a 300 nm target orbit, is 1150 nm or 18 degrees which corresponds to about a 4.8 minute launch time deviation. Provided the ascent vehicle lags the target at injection, these phasing maneuvers can be conducted at no fuel penalty.

In addition to using a stored program for phasing ellipse injection, the ground facilities or the OLF (if present), could be utilized provided they can track the ascent vehicle and have sufficient time to compute and relay the correct commands for the injection.

If there are no failures, this phasing ellipse maneuver can be used to bring the two vehicles within a few miles of each other where the retrieval and docking phases can be initiated. This is assumed to be the normal mode of injecting payloads into orbit for both the OLF and NO OLF families in this report.

2-7. FAMILIES OF OPERATIONAL CONCEPTS

The ranges of the operational variables within which the best operating concept will probably lie can be defined as follows: There will be two families of concepts, one including an OLF and one without an OLF (NO OLF). Within each of these, there are three possible degrees of payload thrust capability. Table 2-1 shows the vehicles required for each possibility, particularly the relation of the tug to the spacecraft and payload. In each case, it is assumed that the guidance will be sufficiently accurate to keep the OLF of spacecraft in a synchronous orbit if desired, and in addition, bring the payload in, from a considerable distance, if it does not arrive close to the OLF.

2-8. SEQUENCE OF OPERATIONS

In order to provide a background against which the guidance functions can be assigned to the vehicles in the Orbital Launch Operation, this subsection outlines a likely sequence of operations during an orbital rendezvous and preparations for lunar launch. Since the operation will vary somewhat, depending upon the existence or non-existence of an OLF; upon whether the payloads are powered or not, etc. two of the six concepts in table 2-1 have been taken as

TABLE 2-1

OPERATIONAL CONCEPTS

Family	Payload Thrust Capability		
	None	Gross	Vernier
NO OLF	* . S/C must be the active vehicle in both the rendezvous and docking maneuvers with P/L. . Ferry vehicle may assist in docking. . Propellant transfer technique used in orbital buildup of space vehicle.	. S/C commands P/L to within 1 nm of S/C. . S/C provides the fine verier thrust for docking with the P/L. . Ferry vehicle may assist in docking.	. S/C commands P/L to within docking range. . S/C need not conduct any thrusting prior to lunar launch. . Ferry vehicle may assist in docking.
OLF	. Tug is required to retrieve and assist P/L in docking with OLF. . Tug must accomplish all thrusting required to move P/L.	. P/L can be commanded to within 1 nm of OLF. . Tug is required to assist in docking P/L with OLF. . Tug can command payload to thrust and can provide vernier control.	*. P/L can be commanded to dock with the OLF with no assist from Tug. . Tug will be required for backup and emergencies. . Vehicle assembly technique used in orbital buildup of space vehicle.

NOTES: 1) * indicates concepts to be studied.
2) Assume phasing ellipse maneuver brings payloads to within a few miles of target.
3) Three man crew is delivered into orbit in C-1 ferry vehicle.
4) S/C - Spacecraft
5) P/L - Payload

illustrative examples, and a sequence of operations for each concept is described. The two concepts are at diagonally opposite corners of the matrix of table 2-1; each is marked with an asterisk in the table.

In the first case, there is no OLF, and the payloads are unpowered. The spacecraft is the chase vehicle that is the active element in docking with fuel payloads. The manned tug/ferry vehicle which delivers the three man crew into orbit can assist the spacecraft in the docking maneuver.

In the second case, there is an OLF, and all payloads are capable of vernier thrust control and can be remotely controlled by the OLF for rendezvous and docking purposes. A manned astrontug will be available to assist in case of trouble, but this is considered a backup mode. The range of this tug is assumed to be 300 nm.

It should be borne in mind that neither of these necessarily represents the "best" or final operational concept. They are two examples from the range of operational variables, within which the best operational concept will probably lie.

a. Operational Concepts - NO OLF

The booster payload configuration for the NO OLF case is shown in figure 2-4 together with the associated weight statement. Figure 2-6 shows diagrammatically one possible arrangement of vehicles during the orbital rendezvous and launch operations. The two C-4 booster concept has been selected from Vought Astronautics data for this case in order to keep the number of dockings within the previously assumed limit for NO OLF operation. It is shown in this figure that only the spacecraft payload and the crew ferry vehicle have engines and are therefore capable of active maneuvering during

the rendezvous. If the initial lunar landing mission is scheduled prior to mid 1967, it is quite possible that C-4 boosters will not be "man rated". In view of high regard for human safety and the high probability of the world spotlight focusing on the initial lunar launch, it has been assumed that the lunar crew will be launched into orbit to rendezvous with the payloads using a C-1 booster with proven reliability. The crew ferry vehicle may assist the rendezvous operation in the role of a tug.

(1) Ascent

(a) Fuel payload (#1) launched into synchronous orbit along a preprogrammed path while tracked by the ground facility. If the launch time was in error or progress along the ascent is not within allowable limits, the ground facility may modify the ascent by determining a correction maneuver and commanding the vehicle accordingly.

(b) Ground facility makes remote confidence checks of P/L #1 after injection into desired orbit. If critical functions malfunctioned during ascent, and are beyond the limited repair capability of the lunar flight crew, a standby fuel payload is launched into orbit.

(c) Spacecraft and OLV payload (#2) launched into compatible rendezvous orbit to within 10 miles of P/L #1. P/L #2 should normally lag P/L #1 for fuel considerations if phasing ellipse technique must be used. Ascent is tracked by the ground facility and commanded accordingly to correct path errors.

(d) Ground facility makes remote confidence checks of P/L #2 and if lethal malfunctions that cannot be repaired are uncovered, a standby payload is launched into orbit.

(e) The C-1 crew ferry vehicle is launched into orbit approximately 24 hours after P/L #2 and follows the same ascent procedures to arrive in the vicinity of the spacecraft payload. Preprogrammed or ground supplied injection information may be updated by the ferry vehicle when the spacecraft payload comes within radar range in order to hold injection accuracy within a few miles.

(2) Terminal Rendezvous and Docking

(a) The crew ferry vehicle determines and executes a thrust program to bring it within 50 feet of the spacecraft. The thrust program may call for an initial maneuver to check out the precision thrust control system.

(b) Docking is accomplished and one man enters the spacecraft to initiate detailed checkout.

(c) After spacecraft is checked out, it is stabilized and position and motion relative to the fuel payload are determined.

(d) The spacecraft determines the thrust schedule required to rendezvous with the fuel payload within 50 feet. Spacecraft executes the schedule first checking the thrust control system.

(e) Ferry vehicle may assist in docking by supplying vernier thrust control to the spacecraft or by serving as a remote observation point.

(3) Propellant Transfer

(a) Propellant transfer lines and the transfer will most probably be controlled automatically by the man in the spacecraft. However, in an emergency one man may leave the crew ferry and assist in positioning a rigid telescopic rod between the fuel payload and the empty OLV. The transfer lines would then be secured in place using the rigid bar to move along and to offset any inertial reaction forces.

(b) Fuel transfer is monitored by the man in the spacecraft and when completed, the remaining crew members enter the spacecraft.

(c) Final preparations and checkouts are made in the spacecraft and the ground is informed of the status.

(4) Launch Positioning

(a) Crew ferry and empty fuel vehicles are moved away under command from the spacecraft.

(b) Ground informs space vehicle of the precise launch time, advises if any orbital plane changes are to be made prior to launch and supplies initial condition data such as orbital parameters, thrust schedule, launch attitude and time hack. In later systems, this operation is done by the spacecraft itself.

(c) Space vehicle is oriented to proper attitude and is launched by the crew on command from the ground.

b. Emergency Operation - Mission Abort

In the event of a major catastrophe, such as might be caused by a very high energy solar flare, all crew members must return safely to earth. The two vehicles capable of reentry are the ferry and the command module of the spacecraft. To maximize human safety, each vehicle must have complete capability to determine the disorbit attitude and thrust schedule and maintain safe passage through the reentry corridor.

c. Operational Concepts - OLF

The operational concepts presented in this subsection assume that an OLF has been launched into a specified orbit which may or may not be synchronous, staffed by the OLF crew, checked out and is in operational status. A manned long range (300 mile) tug is on standby at the OLF to assist in the rendezvous operation if the payloads fail to respond to the OLF commands. The OLF will not thrust out of its scheduled orbit except for station keeping or in an emergency. That is, the payloads and crew ferry will be the active vehicles in the rendezvous.

The booster-payload configuration considered for this case is shown in figure 2-5. Figure 2-7 shows one possible arrangement of vehicles during the orbital rendezvous and launch operations. Three two stage C-4 boosters are required to lift the heavier spacecraft and OLV payloads to orbit and a C-1 booster is again postulated for use with the crew ferry vehicle. The 47,260# weight increase in the spacecraft is the result of using a combination of storable solid and cryogenic propellant systems as compared to an all cryogenic system in the NO OLF case. All payloads are powered by fine thrust control engines to permit precisely controlled maneuvering during the rendezvous.

(1) Ascent

The payload vehicles follow essentially the same sequence developed for the NO OLF case except that when the vehicle comes within range of the OLF's radar, the OLF verifies the vehicle's actual trajectory and determines if the preprogrammed thrust schedule or thrust schedule modified during the ascent by the ground facilities will result in a satisfactory ascent. If corrective action must be taken, the OLF determines new thrusting information and commands the vehicle accordingly.

(2) Terminal Rendezvous

(a) The OLF determines a thrust program to bring the vehicle to within 50 feet of the OLF along a line-of-sight path, rather than using an orbital transfer. The thrust program calls for an initial maneuver to check out the vehicle's attitude stabilization and precision thrust control systems.

(b) The vehicle is commanded along this path by the OLF to within about 50 feet and essentially zero relative velocity.

(3) Docking

(a) Suitable tow bars or cables are remotely attached to the vehicle and the vehicle is towed into the hanger.

(b) Changes in the OLF attitude and station resulting from the docking reaction forces, are sensed and corrected by the OLF.

(4) Space Vehicle Assembly

(a) The first payload to arrive at the OLF is the spacecraft and adapter. Fairly extensive checks are made immediately

to uncover any malfunctions that cannot be repaired in the OLF. If the spacecraft has been seriously damaged during the ascent or rendezvous, the ground is informed that a critical spare part or a replacement vehicle is required. The spare part may be sent up on the next payload. This quick check procedure is used for all payloads.

(b) As the remaining payloads, (2nd and 1st stage of OLV) arrive at the OLF and are checked out, the space vehicle is assembled and pre-launch checks are performed. When the maintenance crew is confident that the space vehicle is ready for its lunar journey, the lunar flight crew is launched from earth and rendezvous with the OLF.

(c) The flight crew transfers to the command module and preparations are made to move the space vehicle from the OLF to the orbital launch area.

(5) Launch Positioning

(a) Space vehicle is moved under control by the OLF to the remote orbital launch area.

(b) Count down sequence is the same as the NO OLF case.

(6) Emergency Operation

Payload or OLF Malfunction - The astrotug, on standby at the OLF, can retrieve a payload within 300 miles of the OLF if malfunctions in the payload or OLF preclude the primary mode of operation. Such malfunctions might be,

(a) Payload will not respond to commands from the OLF or its response may indicate potential control difficulty that would endanger human safety during the last part of the terminal rendezvous.

(b) OLF could not satisfactorily command the payload at long ranges if, for example, the radar malfunctioned. Although the optical ranging device could be used with some degree of accuracy at a range of about 20 miles, the payload may possibly be brought within a few miles of the OLF on command from the ground and the tug then dispatched to complete the rendezvous.

The sequence of operations for the tug is as follows:

(a) Precise orbit determination, thrust schedule and departure time determined by the OLF and supplied to the tug.

(b) Progress along the computed orbit is monitored by the tug and the OLF and when the payload is within range of the tug's radar (from 50-100 miles), the tug may update the thrust program and complete the rendezvous using line-of-sight derived information.

(c) After docking is completed, the tug receives return course information from the OLF, orients itself and the payload, and executes the thrust schedule.

(d) Terminal rendezvous is accomplished the same as step (c) and the docking follows the procedures of primary mode operation.

(7) Mission Abort - Same as the NO OLF case except the tug will in addition, have reentry capability.

2-9. GUIDANCE FUNCTIONS REQUIRED FOR EACH VEHICLE

This subsection enumerates the guidance functions which are required for each vehicle in the Orbital Launch Operation. Table 2-2 and table 2-3 shows the allocation of these among the vehicles, as follows:

- Functions of the payload, if it is powered and guided.
- Functions of the OLF, if it is included in the operation.
- Functions of the tug/ferry.
- Functions of the lunar vehicle, assuming an OLF exists.
- Functions of the lunar vehicle, assuming an OLF does not exist.

In this way, all of the functions required by the vehicles in all of the operational concepts listed in table 2-1 are covered. These functions form the basis for generating the guidance systems which are described in Section 3. The section of recommended guidance systems is described in Appendix C, and is performed in accordance with accuracy requirements stated in Appendix A, and reliability considerations are discussed in Appendix B.

There are basically four types of guidance functions which a space vehicle must perform. These are,

TABLE 2-3 GUIDANCE FUNCTIONS - OLF CONCEPT

Vehicle/Facility	Vehicle Orientation	Thrust Vectoring	Target Tracking	Trajectory Determination
Spacecraft and OLV payloads	• During ascent and injection into near circular orbit or phasing ellipse • During rendezvous and docking with the OLF • For translunar launch • For abort and escape (S/C only)	• During ascent and injection into near circular orbit or phasing ellipse • During rendezvous and docking with the OLF • To inject into translunar trajectory • For abort and escape	not required	• Ascent guidance system determines altitude, velocity and time to properly inject into orbit. This may be updated by ground facilities • OLF determines thrust schedule for rendezvous and docking maneuvers • Later systems will determine launch conditions for injection into translunar trajectory • For abort and escape
OLF	• During Orbit determination • For rendezvous and docking with; Tug, Payload and Spacecraft	• To update orbit for station-keeping • To avoid collisions	• For rendezvous and docking with; Tug, Payload Ferry and Spacecraft • To avoid collisions	• Determines OLF orbit • Determines payload orbit • Determines ΔV for tug to initiate orbit transfer to payload • If no tug, determines path and ΔV for payload
TUG	• During rendezvous and docking with all vehicles • For abort and reentry	• During rendezvous and docking maneuvers with all vehicles • For abort and reentry	• For rendezvous and docking with all vehicles • To avoid collisions	• For rendezvous and docking with all vehicles • For abort and reentry
Manned Ferry (uses C-1 booster)	Same as NO-OLF case except the ferry docks with the OLF and does not act as a tug.	Same as NO-OLF case except the ferry docks with the OLF and does not act as a tug.	• Acquire and track OLF to accomplish rendezvous and docking • To avoid collisions	Same as NO-OLF case

TABLE 2-2

GUIDANCE FUNCTIONS - NO OLF CONCEPT

Vehicle/Facility	Vehicle Orientation	Thrust Vectoring	Target Tracking	Trajectory Determination
Fuel Payload (No Power or Guidance after Injection)	• During ascent and injection into near circular synchronous orbit.	• During ascent and injection into near circular synchronous orbit.	Not Required.	• Ascent guidance system determines altitude, velocity and time to properly inject into orbit. This may be updated or corrected by ground facilities.
SPACECRAFT and OLV Payload (Not Fueled)	• During ascent and injection into near circular synchronous orbit or phasing ellipse. • While tracking the P/L during terminal rendezvous and docking phases. • While thrusting the S/C during the terminal rendezvous and docking phases. • For translunar launch. • For abort and escape.	• During ascent and injection into near circular synchronous orbit or phasing ellipse. • During Terminal rendezvous or docking phases. • To inject into translunar trajectory. • For abort and escape.	• Acquire and track P/L to accomplish rendezvous and docking with P/L. • Track and monitor manned ferry vehicle during rendezvous and docking phases. • To avoid collisions.	• Ascent guidance system determines altitude, velocity and time to properly inject into orbit. This may be updated or corrected by ground facilities. • Convert tracking data during rendezvous and docking to vehicle thrust requirement. • Later systems will determine launch conditions for injection into translunar trajectory. • For abort and escape.
Manned Ferry (uses C-1 Booster)	Same as S/C except: • Ferry docks with S/C. • Ferry does not inject into translunar trajectory.	Same as S/C except: • That there is no ferry translunar injection. • While docked with S/C, assist S/C in rendezvous and docking with P/L by thrusting in direction opposite to S/C thrust.	• Acquire and track S/C to: accomplish rendezvous and docking with P/L. • Track P/L to assist S/C during P/L - S/C docking. • To avoid collisions.	Same as for S/C except: • Ferry does not inject into translunar trajectory.
Ground Facilities	Generate and relay information to ascent vehicle to update or correct, if necessary, vehicle attitude for thrusting properly at orbital injection. Determine S/C attitude required for translunar injection and relay to S/C.	Determine and relay changes to ascent vehicle for proper thrusting at orbital injection. Determine and relay thrust instructions to S/C for translunar injection.	Track and monitor all vehicles during ascent and orbit injection.	Translate tracking data to determine vehicle trajectory and relay information to vehicle when required.

- Orientation of the vehicle.
- Establishment of a thrust vector and a velocity increment.
- Tracking a target object.
- Determining the vehicle trajectory with respect to a given reference.

The detailed assignment of these functions among the vehicles of the Orbital Launch Operation has been used on considerations of,

- Safety
- Functions necessary for successful completion of the mission
- Flexibility of the operation
- Minimization of the number of times which the vernier thrust motor must be operated.
- Fuel economy.

Two of these, safety and flexibility, are particularly pertinent and will be mentioned briefly.

a. Safety

Safety requires that the tug/ferry and the escape portion of the lunar vehicle be capable of reentering the earth's atmosphere and remaining within a safe reentry corridor.

It is also assumed that any manned vehicle in orbit should have the capability of determining its own orbit, at least approximately.

Any manned vehicle which participates in rendezvous operations must be capable of determining the line-of-sight to any approaching body; this capability should preferrably extend to a range of 300 miles, and at least to 100 miles.

b. Flexibility

Flexibility requires that the recommended guidance systems be capable of accommodating any of the operating concepts within the range indicated in table 2-1. Specifically, the guidance should be adaptable to operations with or without an OLF. Similarly, the guidance should be capable of maintaining an orbiting vehicle in a synchronous orbit, if this is desired.

The remaining considerations, for example fuel economy, primarily affect the quantitative accuracy requirements, and are discussed in Appendix A.

SECTION III

RECOMMENDED GUIDANCE SYSTEMS

3-1. GENERAL

This section describes the guidance system configurations which are recommended for the vehicles of the OLO complex. Configurations are recommended for four types of vehicles, the Lunar Spacecraft, the Tug/Ferry, the Orbital Launch Facility, and unmanned Payloads. The recommended systems have sufficient flexibility to accommodate all of the probable operational concepts which are enumerated in table 2-1 of Section II. It is realized that it is quite possible that the operational concept chosen may not require guidance for the payload, or that an Orbital Launch Facility may not be required. The vehicle guidance systems have been synthesized here independent of the Stabilization and Control effort reported in Volume IV. Although the systems described are similar in many respects, care would be taken during the system engineering phase of the study effort to eliminate any unnecessary duplication of equipment.

The recommended guidance systems fall into two classes, guidance for unmanned vehicles, and guidance for manned vehicles. For both classes of systems, the accuracy requirements (stated in Appendix A) are in general, easily attainable with present day guidance equipment, while reliability is a major problem. The process of selection of guidance configurations (described in Appendix C) has consisted of establishing candidates which perform the necessary functions with

the required accuracy, and comparing these primarily on the basis of reliability. The selected candidate is the one which has the greatest probability of achieving the required high degree of operational reliability, which can be proven in orbital flights by 1965, if system design is started now, using presently available components.

Guidance for unmanned payloads id discussed first, since it is simpler. The recommended system, which is selected in Appendix C and described in subsection 3-2, consists of gyros, accelerometers, a horizon tracker, and a computer.

The different kinds of manned vehicles share a great many common guidance requirements. The Orbital Launch Facility is typical of these; the selection of a guidance system for it is therefore discussed in some detail in Appendix C. The systems recommended for the Tug/Ferry vehicle and for the spacecraft are closely related to that for the OLF.

A primary consideration in choosing the guidance configuration for each of the manned vehicles is that the astronauts should be able to determine their own orbit, at least approximately, and should be able to monitor the approach of any other vehicle in the rendezvous operation from a distance of a few hundred miles away. On this basis, the guidance configuration selected for the manned vehicles consists essentially of:

- A manual celestial tracker mounted on an inertial platform.
- A reliable multipurpose radar for determination of altitude.
- Line of sight for rendezvous, and vertical; and the necessary computer and displays.

Included in each of the manned vehicles is a manual periscope that allows the astronaut to look outside the vehicle and perform certain functions both in primary and backup modes of operation. The Lunar Spacecraft has, in addition, midcourse and terminal guidance equipment. If the requirements for orbit determination and for determination of line-of-sight to a few hundred miles are removed, a different candidate guidance system from among those considered in Appendix C might be selected for the manned vehicles.

Two elements of the recommended guidance systems require early initiation of development and orbital flight test programs; each can be designed using presently available components. The first is the multipurpose radar for measurement of altitude, vertical, and line-of-sight for rendezvous, with sufficient reliability (500 hours MTBF by 1965, 1500 hours by 1967) for the OLO and Lunar missions. The second is the manual celestial tracker and inertial platform. Both of these are discussed in Appendix D in sufficient detail to indicate their feasibility and physical characteristics, and to provide a basis for establishing requirements for a development program for each.

3-2. RECOMMENDED GUIDANCE SYSTEM FOR THE PAYLOAD

Various types of payloads will be put into orbit during the OLO operational phase. Among these will be fuel payloads, empty lunar vehicles and astro tugs. Some of these vehicles will have guidance systems of their own, used for boosting the vehicles into orbit or to be used for lunar, abort or rendezvous missions. The system being recommended for further study is intended to be used in remotely controlled payloads that do not require a guidance system for other missions,

or in cases where it is not feasible to use the existing system. The system being recommended meets the reliability and performance requirements specified in Appendices A and B of this volume and performs the functions assigned to it under the operational concepts specified in Section II. The recommended system was selected among a number of candidates. A summary of the candidate selection and tradeoff considerations is contained in Appendix C.

a. System Operation

The primary function of the payload guidance system is to accept commands from either the ground facilities, OLF or Tug and align the thrust vector of the vehicle along a line in inertial space and then to command, measure and cut off the thrust once the required Δ V has been established. To perform this function, the payload guidance system must establish the inertial attitude of the payload. Since the orbit of the payload is known; having been determined by the OLF, Tug or ground stations; inertial attitude may be determined by establishing a vertical reference line to the earth and a reference line in the orbital plane. The recommended system determines attitude in this manner using a horizon scanner to establish a vertical and three orthogonal gyros, used in a gyro compassing mode, to determine vehicle alignment within the orbital plane. The determination of the incremental velocity is performed using accelerometers along the three major axes of the payload and a digital computer performing the integration and computation functions. Figure 3-1 is a block diagram of the complete payload vehicle guidance system.

(1) In addition to the primary function of thrust line and ΔV determination, the payload guidance system must be

DATA LOCATION CHART

Data	Payload	OLF	Tug/Ferry	Spacecraft
System Characteristic, Weight, Power, Volume	Table 3-2	Table 3-5	Refer to OLF	Table 3-10
Component Accuracy Specifications	Table 3-3	Table 3-6	Table 3-8	Table 3-11
System Elements and Their Functions - Primary and Redundancy	Table 3-1	Table 3-4	Table 3-7	Table 3-9
System Block Diagram Primary Mode	Figure 3-1	Figure 3-4	Figure 3-6	Figure 3-9
System Block Diagram with Redundancy		Figure 3-5	Figure 3-7	Figure 3-10
Candidate Guidance Systems	Table C-2	Table C-5	Refer to OLF	Refer to OLF
Reliability Versus Redundancy Curves	Figure C-1	Figure C-3	Figure C-4	Figure 3-11

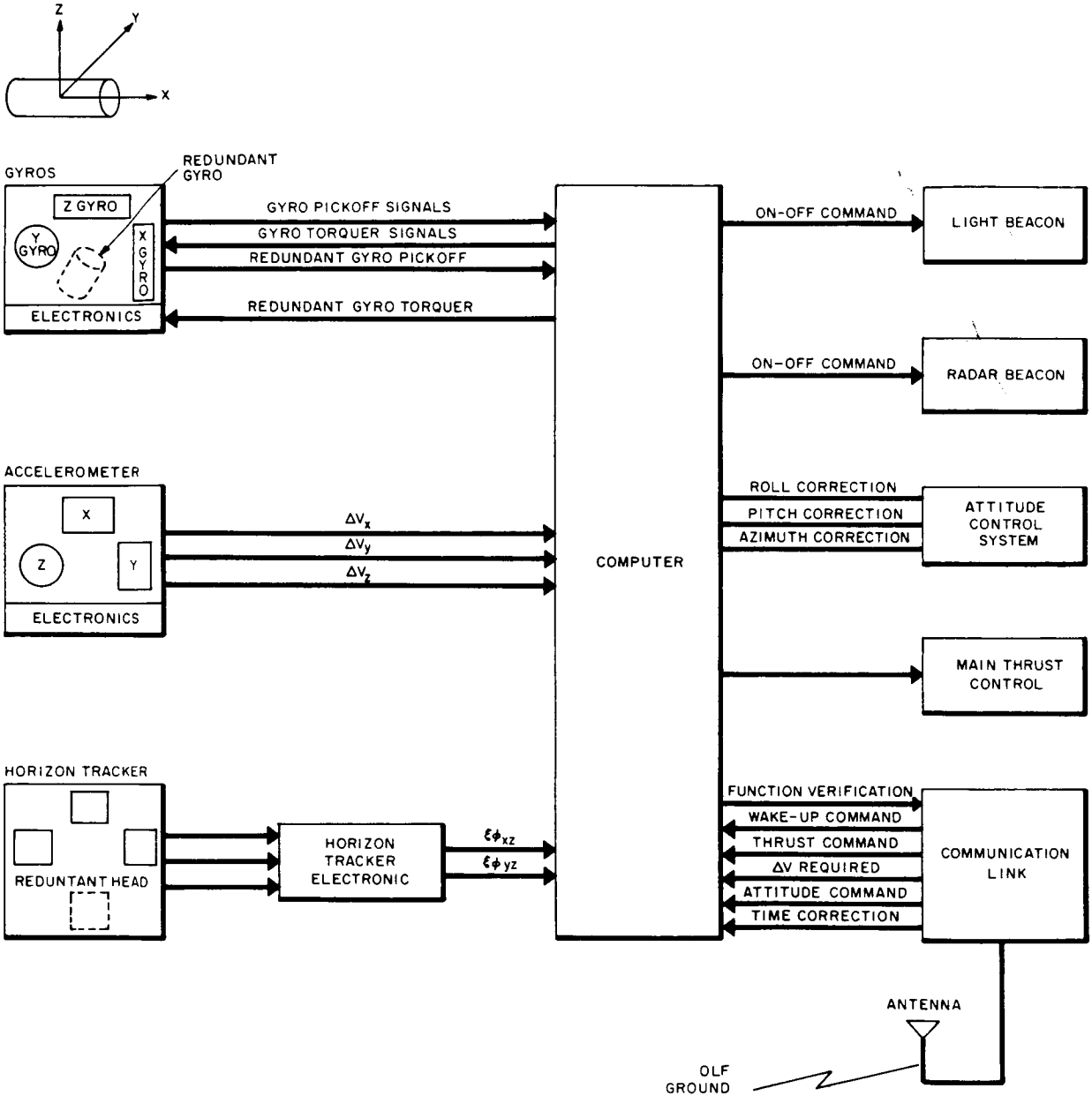


Figure 3-1. Payload Guidance System, Block Diagram

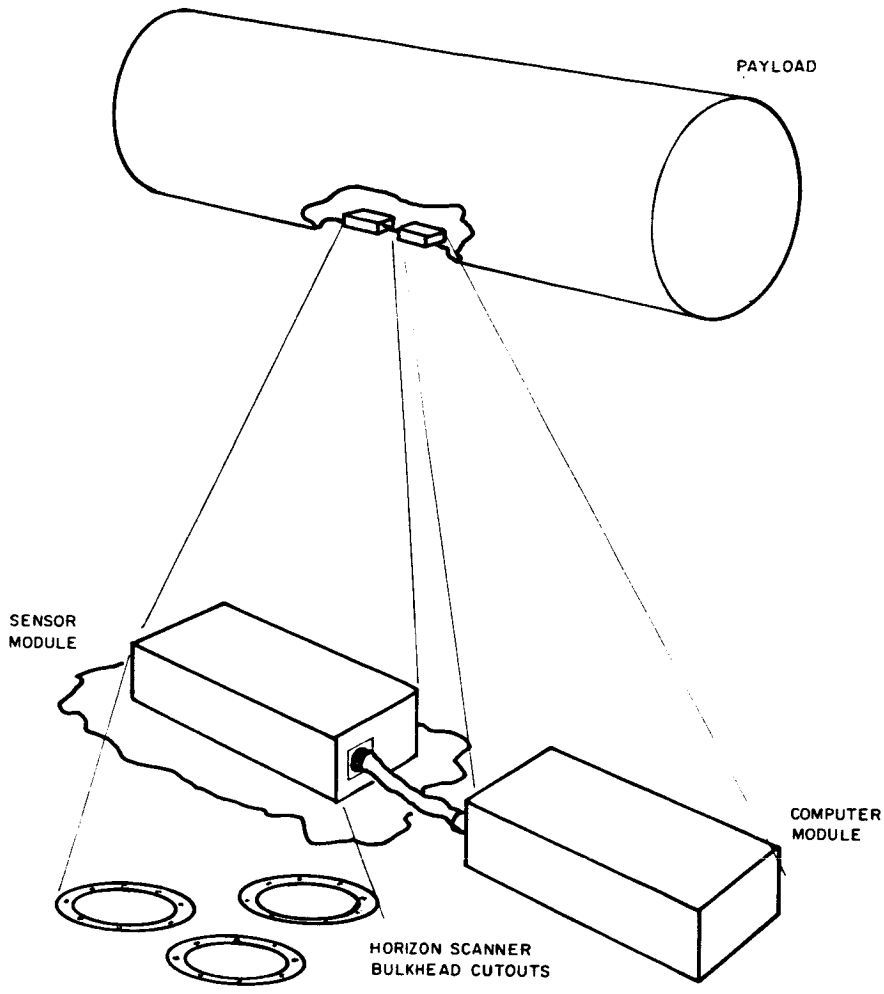


Figure 3-2. Guidance System Location

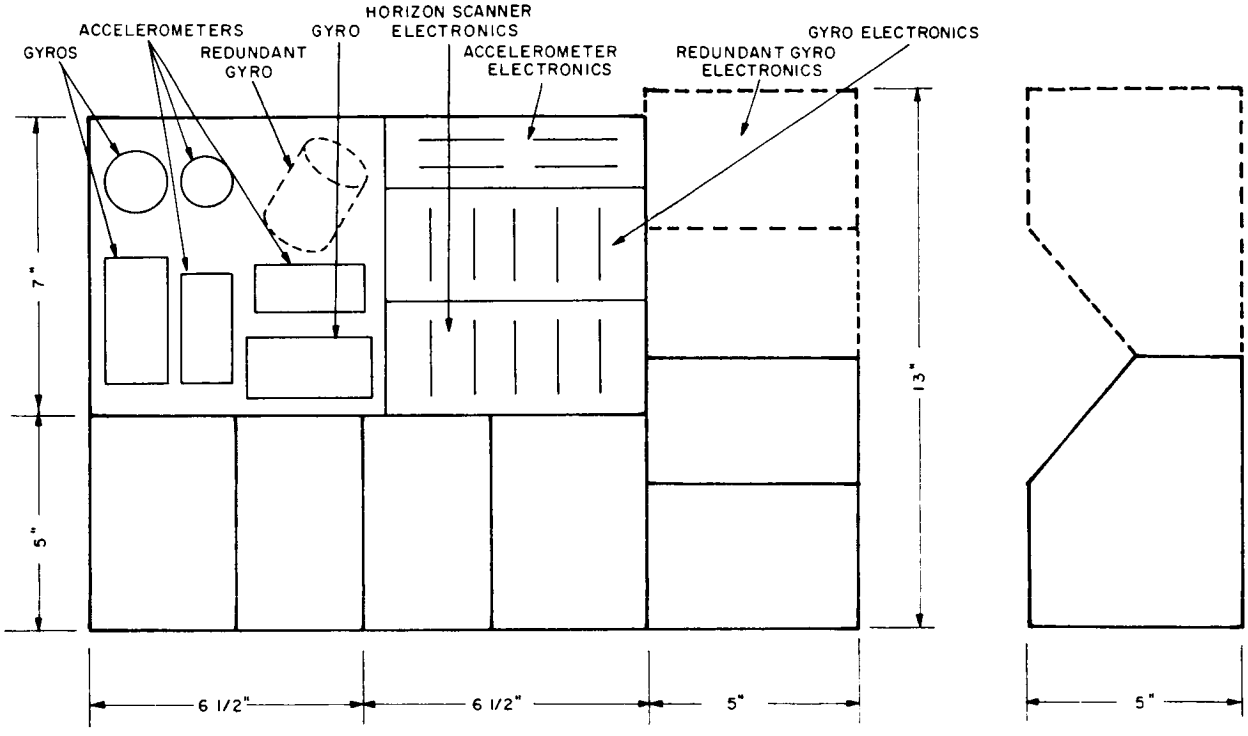


Figure 3-3. Sensor Package Layout

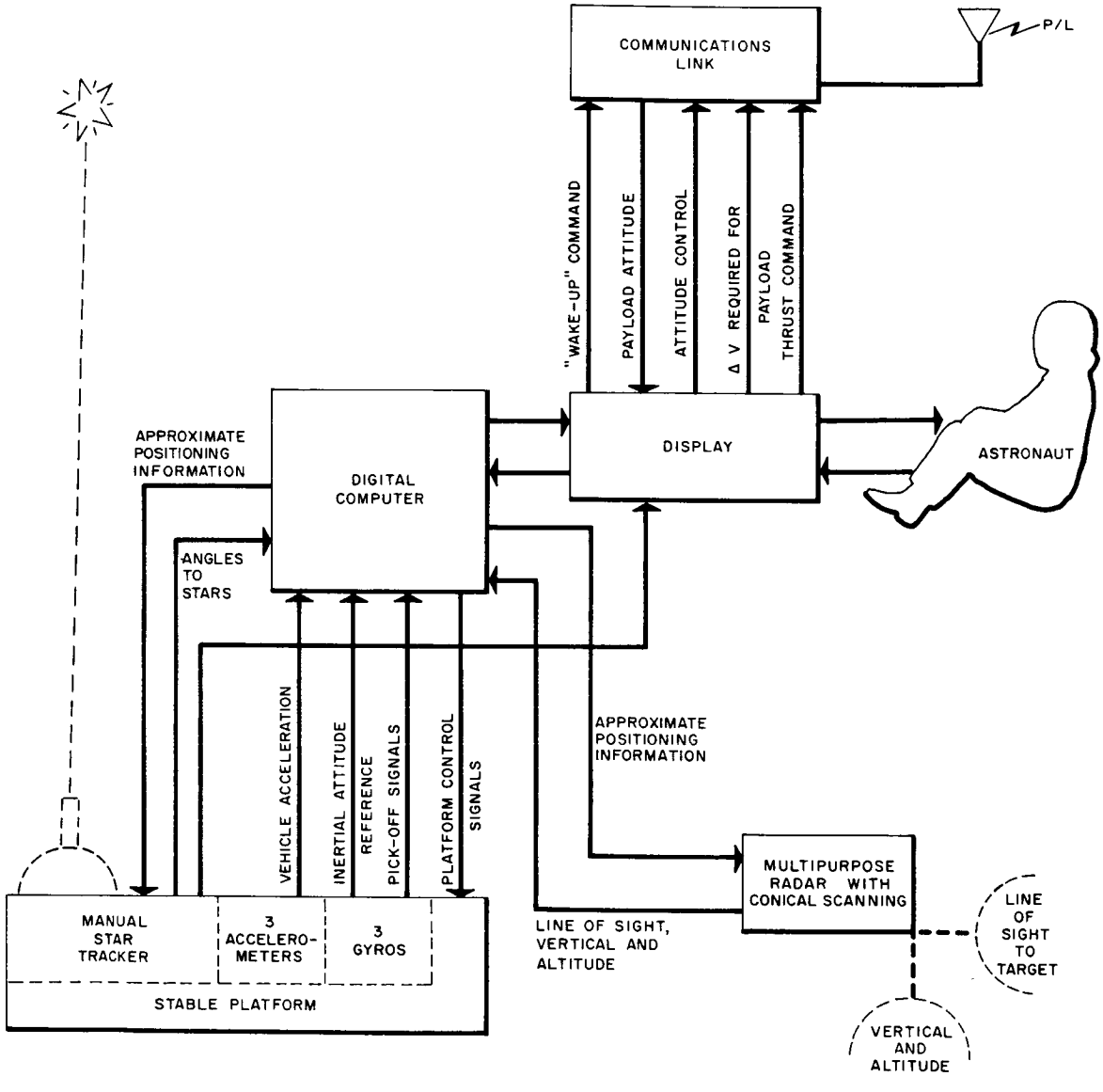


Figure 3-4. OLF Guidance System (Primary Mode of Operation)

capable of stopping the tumbling of the vehicle after it has received a wake-up command. The "stop tumbling" function will be accomplished by utilizing the outputs of the three gyros to compute control signals for the vehicle attitude controls.

(2) Payload injection into orbit can be monitored and controlled by either the ground facilities or the OLF, depending on the particular launch operation concept employed. Where there is a significant phase angle to be made up at the OLF orbit altitude, the payload must have the capability of injecting into a phasing ellipse discussed in Section II. If the phase angle can be predicted at launch, a simple program may be used to compute the incremental velocity to either inject into a phasing ellipse or circularize. Other than alloting a computer program for this function, no other guidance equipment is required.

b. System Description

Table 3-1 lists the system elements and the functions performed by each. Table 3-2 is a tabulation of the weight, power required and volume for each major element in the recommended system. The whole system will be packaged into two modules, a sensor module and a computer module. The sensor module will weight 34 pounds (12 pounds of which is due to beacon required for the OLF guidance system) and occupy a volume of 0.75 cubic feet. The computer module will weight 12 pounds and also occupy 0.75 cubic feet. The whole system will require approximately 200 watts of power. Three bulkhead cutouts, approximately three inches in diameter, will be required in the payload vehicle to permit horizon tracker viewing. Bulkhead cutouts, with respect to guidance system location, are shown in figure 3-2. A tentative sensor

package layout for a strapped down system is shown in figure 3-3. This system has been laid out using existing shelf items.

c. Computer Requirements

The following are a list of functions that must be performed by the digital computer. The computer description is covered in Volume V of this presentation.

- Verify the operation of each function performed by the payload.
- Integrate ΔV's from each accelerometer, determine and command attitude changes and thrust control to achieve required incremental velocity for orbital transfer.
- Upon command from the OLF, or ground, via the communication link, wake-up the system.
- Receive gyro signals, compute attitude signals to stop vehicle from tumbling.
- Determine vehicle alignment in the orbital plane during gyrocompassing mode from gyro and horizon scanner signals.
- Provide thrust signals for orbital injection.

d. Study Recommendations

It is recommended that a study be conducted to determine the guidance system to control system complexity tradeoffs for a strapped down versus a gimballed platform system. This study should include consideration of the various gyrocompassing methods to insure that the best overall system is chosen.

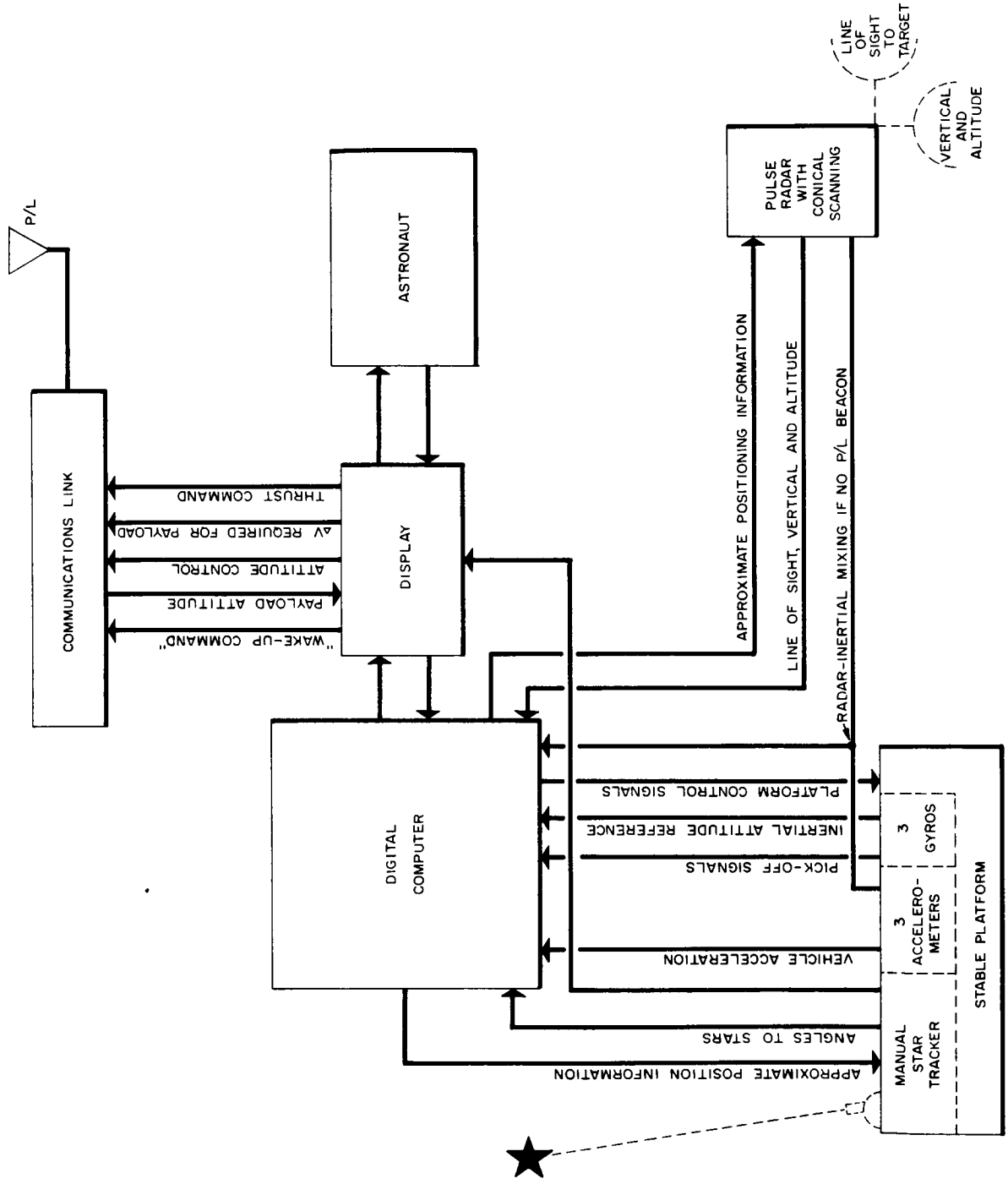


Figure 3-6. Tug/Ferry Guidance System (Primary Mode of Operation)

e. Component Performance Requirements

Table 3-3 indicates the accuracies required for major components of the Payload in order for it to fulfill its mission requirements.

TABLE 3-1

SYSTEMS ELEMENT PERFORMING REQUIRED FUNCTIONS

Function	Sensor	Backup
Wake-up and stabilize	3 Gyros (strapped down)	1 spare gyro
Establish 2 lines in space	(1) Horizon tracker 3 scanning heads	fourth head if desired
	(2) Gyro compass using 3 gyros (strapped down)	spare gyro
Maintain attitude	3 Gyros (strapped down)	spare gyro
Align thrust and compute	3 Accelerometers (strapped down)	

TABLE 3-2

SYSTEM CHARACTERISTICS

Component	Wt No.	Power Watts	Volume Cu. In.
Horizon Tracker (3 heads)	9	6	450
Horizon Tracker Electronics	5		200

TABLE 3-2 (Cont)

Component	Wt No.	Power Watts	Volume Cu. In.
*Radar beacon	12**	40**	40**
4 Gyros	3.8	32	240
Gyro electronics	2.5	6	30
3 Accelerometer	1	-	33
Accelerometer electronics	0.5	1.2	6
Sensor Totals	32.8	85.2	999
Computer	12	100	1200
System Total	53.8	205.2	2199

**This includes the redundancy required to attain a reliability of 0.99 as shown in Appendix C.

*The Radar beacon is the part of the OLF Guidance System located on the P/L (See Appendix C).

TABLE 3-3

PAYLOAD COMPONENT ACCURACY SPECIFICATIONS

Components	Performance Characteristics
Gyros	3°/Hr RMS random drift, 300°/Hr/g, 45°/Hr/g^2
Accelerometers	0.4% RMS, or 0.1 ft/sec. RMS accuracy (whichever is larger)
Horizon Tracker	10 minute RMS accuracy at 300 nm altitude
Beacon	See radar description in Appendix D.

3-3. RECOMMENDED GUIDANCE SYSTEM FOR THE OLF

The guidance system for the OLF must, as stated in Section II, perform these functions:

- Determine the attitude of the vehicle
- Determine the orbit of the vehicle
- Determine the line of sight to the rendezvous target; this involves acquiring target vehicle, determining angle and angular rate of line-of-sight to target, and determining range and range rate to target.
- Generate and control velocity increments, e.g. for station keeping.

a. Recommended Guidance System

The recommended system among the six candidates studied in Appendix C is system number three. The primary mode consists of the following elements:

- Manual star tracker that is platform mounted (one concept is described in Appendix D)
- Inertial platform (3 gyros and 3 accelerometers)
- Reliable multipurpose radar (one concept is described in Appendix D)
- Digital computer
- Radar beacon on payload
- Manual periscope and displays

Figure 3-4 is a signal-flow block diagram of the primary mode.

b. System Operation in the Primary Mode

Table 3-4 lists the major guidance functions and indicates which elements of the system perform each, in the primary

mode. (It also lists redundant modes, as discussed in later subsections.) Briefly, each major function is performed as follows:

- Orbit Determination - The orbital parameters are determined by smoothing the information from a series of celestial position fixes. Each position is made using altitude and vertical information from the multipurpose radar, and determining two lines in the inertial space with the platform-mounted manual star tracker. The star tracker is positioned by the computer and sensors, as is an automatic star tracker; the operator's task is to correct any error.
- Attitude Determination - This is achieved as a by-product of orbit determination. The gyros on the inertial platform store the inertial reference between sightings with the star tracker.
- Line-of-Sight - The multipurpose radar will perform all of the line-of-sight functions (acquisition, angle and angular rate, range and range rate), from a maximum range of 300 nm to a minimum of 1000 feet. Within this minimum distance, visual optics (manual periscope) will be used.
- Control of Velocity Increment - This will be accomplished by the accelerometers on the inertial platform, together with the digital computer.

c. Redundancy

The recommended system contains two levels of redundancy, as indicated in Appendix C, (1) redundancy of the elements of the primary mode, to give full capability

TABLE 3-4

PRIMARY AND REDUNDANT MODES OF OPERATION FOR THE OLF GUIDANCE SYSTEM

Primary Mode and Redundancy with full capability			Further Redundancy, Usually with Reduced capability			
Guidance Function	Primary Equipment	Back up Equipment	Equipment	Remarks	Equipment	Remarks
Orbit Determination						
1) Local vertical	1) Multipurpose radar	1) Redundant electronics.	1) IR horizon tracker	1) Horizon tracker can determine vertical to required accuracy	1, 2, 3 ground tracking	Number of ground stations presently available can not track continuously. A number of vehicle orbits must be permitted to achieve required accuracy
2) Altitude	2) Multipurpose radar	2) Redundant electronics.	2) IR horizon tracker	2) Horizon tracker requires considerable smoothing to achieve required altitude accuracy		
3) Two lines in space	3) Manual star tracker	3) Redundant electronics and servos.	3) Manual positioning and readout			
Altitude Determination and control						
1) Two lines in space	1) Manual star tracker	1) Redundant electronics and servos.	1) Manual positioning and readout			
2) Maintain references	2) Gyro stabilized platform	2) Redundant Gyro and platform elect.	2) Vehical stabilization system	2) Vehical attitude will be maintained to a few degrees		
ΔV Control						
1) Computation	1) Digital computer		1, 2 Manual computation and control	1, 2 Accuracy limited by tolerance on engine vernier thrust		
2) Sensing	2) Accelerometers					
Line Of Sight						
1) Acquisition	1) Multipurpose radar and payload beacon	1) Redundant radar elect. and payload beacon.	1) Periscope	1,2,3,4,5, Periscope requires that target be illuminated by sunlight. Useable range 30 Nm		
2) Bearing angles	2) Multipurpose radar and payload beacon	2) Redundant radar elect. and payload beacon.	2) Periscope		2,3,4,5 ground tracking plus visual observation at close range	

TABLE 3-4 (Cont)

PRIMARY AND REDUNDANT MODES OF OPERATION FOR THE OLF GUIDANCE SYSTEM

Primary Mode and Redundancy with full capability			Further Redundancy, Usually with Reduced capability			
Guidance Function	Primary Equipment	Back up Equipment	Equipment	Remarks	Equipment	Remarks
Line Of Sight (Cont)						
3) Range	3) Multipurpose radar and payload beacon	3) Redundant radar elect. and payload beacon	3) Periscope with Stadiametric ranging	3,4 Stadiametric ranging useable range of 5Nm		
4) Range rate	4) Multipurpose radar and payload beacon	4) Redundant radar elect. and payload beacon	4) Periscope with Stadiametric ranging			
5) Angular rate	5) Multipurpose radar payload beacon plus computer	5) Redundant radar elect. and payload beacon	5) Periscope			

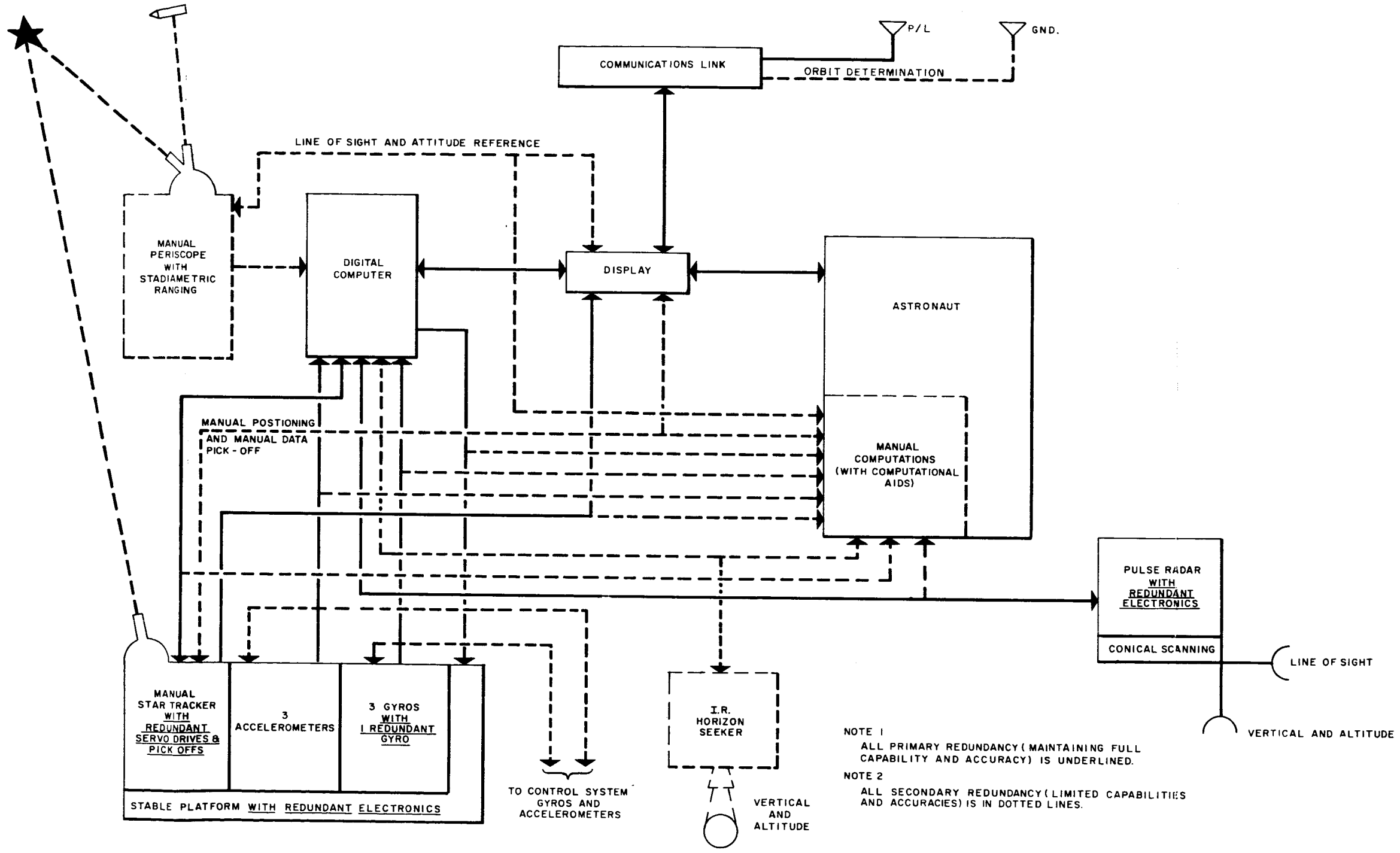


Figure 3-5. Primary and Secordary Redundancy for the OLF Guidance System

performance with a reliability of 0.99 throughout the 20 hour operating requirement stated in Section II, and (2) further redundancy, usually with degraded performance, for the approximately 1 percent of the missions in which the full capability may fail.

Figure 3-5 is a block diagram showing the redundant elements, and table 3-4 lists the primary and back-up means of performing each function.

In order to achieve the reliability of approximately 0.99 with full capability, redundancy is provided for the following, as explained in Appendix C.

- Platform electronics
- Servo drives and data pickoffs for the manual star tracker
- Gyro
- Multipurpose radar electronics
- Radar beacon electronics.

With the recommended level of redundancy, the estimated reliability is 0.9893 rather than 0.99. Tradeoff data are presented in Appendix C (figure C-3 shows added weight versus added reliability) which indicate that a considerable weight increment would be required to raise the estimated reliability beyond the recommended system's 0.9893 to 0.990.

Additional back-up modes, not necessarily with full capability, are provided as follows:

- I R Horizon Tracker for altitude and vertical, in case of failure of these functions of the multipurpose radar .

- A manual periscope with a simple provision for estimating range to the object sighted; this will provide a back-up for acquisition of target, angle, and range, in case of failure of these functions of the multipurpose radar.
- Attitude control system gyros and accelerometers will be used in case of failure of the corresponding units of the inertial platform.
- Manual positioning of the star tracker telescope, and manual readout of data will be available in case of failure of the electronics.
- Essential computations and control functions will be performed manually in case of failure of the digital computer or control elements.

d. System Characteristics

Table 3-5 lists the weight, power and volume of the recommended system; it includes the recommended redundancy.

TABLE 3-5

OLF SYSTEM CHARACTERISTICS

	Component	Weight pounds	Power Watts	Volume cubic in.
1.	Horizon Tracker (3 heads)	9	6	800
2.	Platform - 3 Gyros, 3 Accelerometers + manual Star Tracker	42*	95*	2000*
3.	Radar Antenna + Radar Electronics	41*	110*	3800*
4.	Platform Electronics	20*	70*	500*

*This includes the redundancy required to attain a reliability of 0.9857 as shown in Appendix C.

TABLE 3-5 (Cont)

	Component	Weight pounds	Power Watts	Volume cubic in.
5.	Horizon Tracker Electronics	5	5	300
6.	Periscope with Stadiametric Ranging	ND	ND	ND
7.	Display Console	ND	ND	ND
8.	Radar Beacon	6	20	20
	Total	123	306	6420**

**Does not include computer

ND - Not determined

c. Computer Requirements

Computer description and specifications are discussed in Volume V of this study. The following is a list of functions that must be performed by the OLF digital computer.

- Compute orbit from sensor inputs and stored celestial data
- Compute attitude from sensor inputs and stored celestial data
- Compute signals for multipurpose radar altimeter antenna stabilization
- Compute approximate angular positioning of multipurpose radar interrogator antenna and compute stabilization signals
- Compute range rate and angular rate between P/L and OLF

- Compute ΔV program for P/L rendezvous, including attitude control
- Compute drift of OLF from synchronous orbit
- Compute ΔV program for OLF station keeping, evaluate actual changes from accelerometer output and generate correction signals, if required
- Compute and convert all data to be displayed
- Compute drift of attitude control system gyros
- Compute P/L and OLF predicted ephemerus, weight, and fuel load
- Compute program for torquing platform gyros, if desired.

f. Display Requirements

The following are some of the parameters that must be displayed for line-of-sight, station keeping, station safety and docking:

- Line-of-sight - sighting angle, angular rates, range and range rate to approaching vehicle
- Station keeping - Attitude, altitude, time, position, velocity and deviations from normal orbit
- Station safety - Relative position to nearest ground tracking station, environmental conditions, fuel
- Docking - some kind of visual display showing positions of payload with respect to OLF.

g. Component Performance Requirements

Table 3-6 indicates the accuracies required for major components of the OLF guidance system in order for it to

fulfill its mission requirements. Computer specifications have not been included.

TABLE 3-6

OLF COMPONENT ACCURACY SPECIFICATIONS

Components	Performance Characteristics
Gyros	0.2°/Hr. RMS random drift; 50°/Hr./g 5°/Hr./g^2
Accelerometers	0.02% RMS total error, at least 0.002 ft/sec. resolution
Horizon tracker	6 minutes RMS at 300 nm altitude
Manual star tracker	15 minutes RMS accuracy
Radar and beacon	See radar description in Appendix D.

3-4. RECOMMENDED GUIDANCE SYSTEM FOR THE TUG/FERRY VEHICLE

The primary purpose of the Tug is to assist in the rendezvous operation of two vehicles in the OLO complex, while that of the Ferry is to deliver crew members to either the OLF or Spacecraft. In order to accomplish these tasks, both vehicles must perform virtually the same guidance functions, so that the guidance system recommended in this section is applicable to both the Tug and Ferry vehicles. The functions which must be performed are an outgrowth of the following considerations:

- The OLO operation may or may not include an OLF.
- The requirement of crew safety dictates the need for an abort capability.

In view of these considerations, the guidance system for the Tug/Ferry must be capable of performing the following functions:

- Determine its orbit
- Determine and control attitude
- Determine line-of-sight to target vehicle
- Generate and control ΔV's
- Determine and control reentry path

All but the last function are identical with those required of the OLF guidance system.

a. Recommended System

The equipment required to perform the first four functions listed (orbit, line of sight, attitude, ΔV's) is sufficient to accomplish earth reentry with only minor additions to computer and display requirements. However, because of the similarity between the OLF and the Tug/Ferry functional requirements, the guidance system recommended for the OLF accommodates the Tug/Ferry vehicle as well. Although functionally similar, the accuracy requirements for Tug/Ferry guidance are somewhat more stringent than the requirements for the OLF system due primarily to the earth reentry requirement. Reference to Appendix C will show the six candidates from which the recommended system configuration was selected and will give the reasons for selecting the particular elements which make up this system.

For operation in the primary mode, the recommended guidance system consists of the following elements:

- Reliable multipurpose radar
- Inertial platform containing 3 gyros and 3 accelerometers
- Manual star tracker - platform mounted
- Digital computer
- Radar beacon located on payload
- Displays.

A block diagram of these elements, showing the signal flow in the primary mode, is shown in figure 3-6.

A feature of the tug guidance system is the provision for switching to longer pulse width (30 usec instead of 1 usec), inertial-radar mixing, and longer integration times, in the event of failure of the beacon on the payload, or if it is necessary to rendezvous with targets not beacon-equipped. If this is not done, and the payload beacon fails, the range of the multipurpose radar will be reduced from 300 nm to about 20 nm (assuming the payload has a radar cross-sectional area of three square meters). However, by using this radar-inertial mixing feature (described in Appendices C and D) the radar can measure angle, range and range rate to ranges of about 100 nm (using a four minute integration time).

In this mode of operation, the long-term accuracy is supplied by the radar, and short-term information by the accelerometer, which takes account of thrusting of the Tug/Ferry, etc. Initial information giving the approximate location of the target, to enable the astronaut to bracket the target with the radar range gate and enable the radar to lock-on, will be supplied by the ground, the OLF, or by sightings by the astronaut.

b. System Operation in the Primary Mode

A description of system operation, typical of the Tug/Ferry guidance system, is presented in paragraph 3-3 a. Table 3-7 lists the five primary functions performed by this vehicle and indicates those elements of the system required to accomplish each of the functions. Back-up operation (to be discussed later in this subsection) is also indicated in this table. Since the Tug/Ferry vehicle has the additional capability of earth reentry, several factors that influence system operation are briefly described for the three phases of reentry.

(1) Computation of a de-boost program from a knowledge of the orbit parameters and the desired landing location. This includes determining the attitude and ΔV at the time of retro-thrusting required to obtain the proper flight path angle as the vehicle enters the atmosphere.

(2) Attitude control and measurement of ΔV during retro-thrusting, and control of engine cut-off.

(3) Control of the vehicle's flight path in the atmosphere. This includes keeping the vehicle within the safe limits of the reentry corridor with the least energy expenditure, and adjusting the flight path within the corridor to obtain the desired longitudinal and lateral range. In the primary mode, the guidance system will determine and use such quantities as velocity, altitude, attitude, and range to target to control the flight path. During passage through the reentry corridor, the guidance system should be supplemented with temperature sensors to provide an emergency override if the vehicle approaches too close to the thermal boundary. If problems such

TABLE 3-7

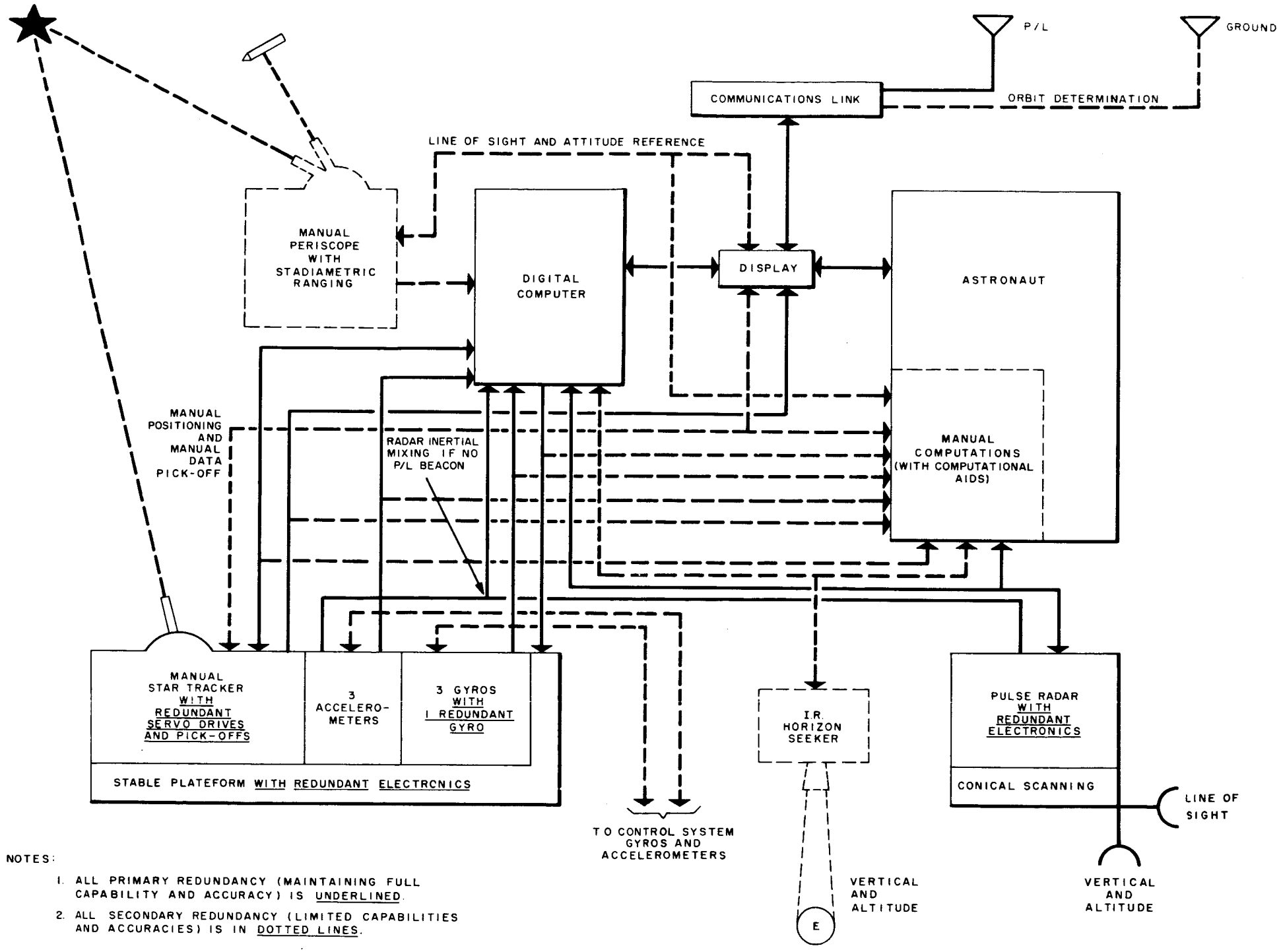
PRIMARY AND REDUNDANT MODES OF OPERATION FOR THE TUG/FERRY GUIDANCE SYSTEM

Primary Mode and Redundancy with Full Capability			Further Redundancy, Usually with Reduced Capability			
Guidance Function	Primary Equipment	Backup Equipment	Equipment	Remarks	Equipment	Remarks
Orbit Determination						
1) Local vertical	1) Multipurpose radar	1) Redundant electronics	1) IR horizon tracker	1) Horizon tracker can determine vertical to required accuracy	1,2,3, Ground Tracking	Number of ground stations presently available can not track continuously. A number of vehicle orbits must be permitted to achieve required accuracy
2) Altitude	2) Multipurpose radar	2) Redundant electronics	2) IR horizon tracker	2) Horizon tracker requires considerable smoothing to achieve required altitude accuracy		
3) Two lines in space	3) Manual star tracker	3) Redundant electronics and servos	3) Manual positioning and readout			
Attitude Determination						
1) Two lines in space	1) Manual star tracker	1) Redundant electronics and servos	1) Manual positioning and readout			
2) Maintain references	2) Gyro stabilized platform	2) Redundant gyro and platform elect.	2) Vehicle stabilization system	2) Vehicle attitude will be maintained to a few degrees		
ΔV Control						
1) Computation	1) Digital computer		1,2 Manual computation and Control	1,2 Accuracy limited by tolerance on engine Vernier thrust		
2) Sensing	2) Accelerometers					

TABLE 3-7 (Cont.)

PRIMARY AND REDUNDANT MODES OF OPERATION FOR THE TUG/FERRY GUIDANCE SYSTEM

Primary Mode and Redundancy with Full Capability			Further Redundancy, Usually with Reduced Capability			
Guidance Function	Primary Equipment	Backup Equipment	Equipment	Remarks	Equipment	Remarks
Line-Of-Sight						
1) Acquisition	1) Multipurpose radar and payload beacon	1) Redundant radar elect and payload beacon	1) Periscope	1,2,3,4,5, Periscope requires that target be illuminated by sunlight. Usable range 30 nm		
2) Bearing angles	2) Multipurpose radar and payload beacon	2) Redundant radar elect and payload beacon	2) Periscope		2,3,4,5 Ground Tracking plus visual observation at close range	
3) Range	3) Multipurpose radar and payload beacon	3) Redundant radar elect and payload beacon	3) Periscope with Stadiametric ranging	3,4 Stadiametric ranging usable range of 5nm		
4) Range rate	4) Multipurpose radar and payload beacon	4) Redundant radar elect and payload beacon	4) Periscope with Stadiametric ranging			
5) Angular rate	5) Multipurpose radar payload beacon plus computer	5) Redundant radar elect and payload beacon	5) Periscope			
Re-entry						
1) Attitude Reference	1) Gyro stabilized platform	1) Redundant gyro and platform elect	1) Horizon tracker and periscope	1) This backup can be used during the initial phase of re-entry. Can not be used in the atmosphere.	1) Vehicle Attitude Stabilization System	1) Coarse information
2) ΔV Control computation sensing	2) Digital computer accelerometers		2) Manual computation and control	2) Accuracy limited by tolerance on engine Vernier thrust		
3) Acceleration (g load)	3) Accelerometers plus computer					
4) Vertical velocity	4) Radar inertial mixing	4) Redundant radar electronics	4) Radar or accelerometer	4) Radar not usable during large period of re-entry due to plasma effect. Accelerometer has build-up error in pure inertial operation.	4) Ground Tracking	4) Ground can not supply complete coverage
5) Altitude	5) Radar inertial mix.	5) Redundant radar elect.	5) Radar or accelerometer plus complete integration	5) Same as 4) above	5) Ground Tracking	5) Ground can not supply complete coverage



NOTES:

1. ALL PRIMARY REDUNDANCY (MAINTAINING FULL CAPABILITY AND ACCURACY) IS UNDERLINED.
2. ALL SECONDARY REDUNDANCY (LIMITED CAPABILITIES AND ACCURACIES) IS IN DOTTED LINES.

Figure 3-7. Primary and Secondary Redundancy for the Tug/Ferry Guidance System

as thermal time lags and establishing accurate temperature re-entry path relationships can be overcome, then temperature sensors can be used to control the vehicle's flight path until such time that the guidance system can again resume control. Further study must be conducted to determine the degree to which this type of control can be satisfactorily used.

c. Redundancy

Within the framework of operational concepts discussed in Section II, it has been shown that the Tug/Ferry vehicles may support the orbital launch operation in several ways depending on whether an OLF exists or not. Although these variousoperational possibilities do not influence the type of guidance system required for the vehicle, they do effect the amount of redundancy to be added to achieve the reliability goal of 0.99 because the system operating time, is different for each operational concept. For example:

- Vehicle operates as both a crew ferry and space tug if an OLF does not exist. Estimated system operating time - 20 hours.*
- Vehicle operates in emergency capacity as a space tug and another vehicle serves as a crew ferry if an OLF exists. Estimated system operating time 15 hours and 5 hours, respectively.

It is realized that a comparison of such factors as system weight put in orbit, logistics, space maintenance and operational flexibility may show it to be more economical to equip all vehicles of this type with identical guidance systems rather than tailor the amount of back-up in each system to fit the specific application.

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*Refer to Appendix B.

For this study, the required redundancy has been developed for system operation of 15 hours (the estimated operating time for the astrotug). For purposes of comparison, the reliability of the OLF system, having the same redundancy, is listed for system operation of 20 hours.

The recommended system contains two levels of redundancy, (1) redundancy of the primary mode elements, to give full capability performance with a reliability of 0.99 throughout the 15 hour operating requirement stated in Appendix B and (2) further redundancy, usually with degraded performance, for the approximately 1 percent of the missions in with the full capability may fail.

Figure 3-7 is a block diagram showing the redundant elements required for the recommended system. A list of primary and back-up means of performing each function is shown in table 3-7.

In order to achieve the reliability of approximately 0.99 with full capability, redundancy is provided for the following:

System Operation - 15 hours

- Platform electronics
- Servo drives and data pickoffs for the manual star tracker
- Gyro
- Multipurpose radar electronics
- Radar beacon electronics

With the recommended level of redundancy, the estimated reliability is 0.992; slightly greater than the reliability goal of 0.99. Tradeoff data are presented in Appendix C,

figure C-4, which show that this reliability has been attained at a cost of about 32 pounds of weight. For the same system, with the same degree of redundancy, system operation for 20 hours would decrease the reliability to 0.9893. If the recommended system was installed in a vehicle requiring only 5 hours operation (for example the crew ferry in the OLF case), the reliability goal would be far exceeded or, as an alternative, some of the redundancy could be removed thereby reducing the weight of the system.

Additional back-up modes, not necessarily with full capability, are provided as follows:

- IR horizon tracker for altitude and vertical, in case of failure of these functions of the multipurpose radar.
- A manual telescope with a simple provision for estimating range to the object sighted; this will provide a back-up for acquisition of target, angle, and range, in case of failure of these functions of the multipurpose radar.
- Attitude control system gyros and accelerometers will be used in case of failure of the corresponding units of the inertial platform.
- Manual positioning of the star tracker telescope, and manual readout of data will be available in case of failure of the electronics.
- Essential computations and control functions will be performed manually in case of failure of the digital computer or control elements.

d. System Characteristics

Table 3-5 lists the weight, volume, and required power of each major component in the guidance system. This system is essentially the same as the system recommended for the OLF.

e. Computer Requirements

A description of the computer is covered in Volume V of this report. The following are a list of functions which must be performed by the digital computer.

- Compute orbit from sensor inputs and stored celestial data
- Compute attitude from sensor inputs and stored celestial data
- Compute signals for radar altimeter antenna stabilization
- Compute approximate angular positioning of pulse radar interrogator antenna and compute stabilization signals
- Compute range gate reference signals from inertial information during periods of thrusting
- Compute range rate and angular rates between Tug/Ferry and Target
- Compute ΔV program for rendezvous with target
- Compute drift of Tug/Ferry from synchronous orbit, if required
- Compute Tug/Ferry predicted ephemeris, weight, fuel load

- Compute and correct drift of attitude control system gyros
- Compute required attitude and ΔV for passing through re-entry window
- Compute flight path control signals to remain in reentry corridor.

f. Display Requirements

The following signals should be displayed to assist the astronaut in monitoring and controlling the various guidance functions:

- Line-of-sight - Target acquisition signal, range, range rates, and bearing angles to target
- Station keeping - Attitude, altitude, time, position relative to earth, velocity
- Control of ΔV's - Comparison of programmed ΔV's and integrated accelerometer outputs
- Terminal rendezvous and docking - Display showing the relative positions of Tug/Ferry and Target, and an indication of the maximum safe closing rate
- Payload control - All quantities relating to P/L thrust control, such as P/L attitude, engine on/off signals, fuel flow rates, etc.
- Re-entry- Altitude, velocity, attitude, vehicle environmental conditions
- Station safety - Relative position to nearest ground tracking station, vehicle environmental conditions, fuel.

g. Component Performance Requirements

Table 3-8 indicates the required accuracies for major components of the Tug/Ferry in order for it to fulfill its mission requirements. Computer specifications have not been included, since these may be seen as a portion of the Computer System presentation. (Volume V)

TABLE 3-8

TUG/FERRY COMPONENT ACCURACY SPECIFICATIONS

Component	Performance Characteristics
Gyros	0.1°/Hr RMS random drift, 0.15°/Hr/g, 0.1°Hr/g^2
Accelerometers	0.03% RMS
Horizon scanner	6 minutes RMS at 300 nm altitude
Manual star tracker	5 minutes RMS accuracy
Radar and beacon	See radar description in Appendix D.

3-5. SPACECRAFT GUIDANCE SYSTEM

The spacecraft mission includes several phases, some of which are shared by other vehicles of the OLO complex, and some of which are unique:

- The spacecraft is sent into orbit as an unmanned payload
- Once in orbit, it is assembled or fueled and checked out. During this time, it may assume some of the functions of an OLF, particularly if the operating concept does not include an OLF

- It must be provided with necessary guidance information, and must be properly aligned for the orbital launch; the launching thrust must be accurately controlled
- During the midcourse phase of the lunar journey, the astronauts in the spacecraft must be able to determine their orbit, and correct it as necessary
- The spacecraft must contain lunar approach and terminal guidance
- The return from the moon includes lunar launch, midcourse, and earth approach, and reentry into the earth's atmosphere, and landing.

The discussion of operational concepts in Section II indicates that, if all goes as planned during the Orbital Launch Operation, the spacecraft is essentially an unmanned payload during its ascent, and following this can always be close to and under the control of either the OLF or the Tug/Ferry vehicle. Strict adherence to this would lead to the conclusion that the only guidance functions required for the spacecraft are:

- Guidance functions of the unmanned payload
- Functions for alignment of the spacecraft prior to orbital launch, and storage of data for the lunar journey
- Measurement and control of velocity increment during orbital launch.

There is a very large possibility, however, that the spacecraft may assume other functions during the Orbital Launch Operation, either planned or unforeseen. The spacecraft may

serve as living quarters, or as a workroom for checkout or repair work. It may become actively involved in the rendezvous, either as an independent vehicle or as a control center. It might receive an unanticipated velocity increment, either through misfire of an engine, or an abortive attempt at orbital launch. This would leave the spacecraft in an unknown and possibly dangerous orbit, which must be determined as soon as possible. Ground tracking will not be present continuously.

Since these possibilities exist, the determination of spacecraft guidance functions must follow the principle that, since it is a manned vehicle and is part of the rendezvous operation, the spacecraft should be able to determine its own orbit, and determine the line-of-sight to other vehicles in the OLO complex, in the same way as the OLF and the Tug/Ferry vehicle.

a. Spacecraft Guidance Functions

The spacecraft guidance system must therefore perform the following functions:

(1) Determine the spacecraft orbit, approximately. (Accurate orbit information prior to orbital launch will be supplied from the ground.)

(2) Determine the line-of-sight (acquire, determine angle and angular rate, and range and range rate) to other vehicles in the rendezvous operation, to a distance of 300 nm.

(3) Perform the guidance functions of the unmanned payload during ascent into orbit.

(4) Determine and control attitude. Attitude determination must be sufficiently accurate to enable the orbital launching to be accomplished accurately.

(5) Measurement and control of velocity increment, with sufficient accuracy to enable the orbital launching to be successfully accomplished.

(6) Midcourse guidance

(7) Lunar approach and terminal guidance

(8) Earth approach and reentry guidance.

b. Recommended Guidance System

These functions include all of the functions required for the OLF plus a number of others; the selection of guidance system elements to perform the OLF-type functions in the spacecraft can be based on the same candidates and choices as in the selection of the system for the OLF. The recommended guidance system for the spacecraft is determined, then, by commencing with a system similar to that recommended for the OLF (Section 3-3 and Appendix C) and examining two further considerations:

- What must be added to the OLF guidance system to enable it to perform the remaining functions required of the spacecraft in the Orbital Launch Operation?
- Can the resulting guidance system be efficiently combined with elements necessary to perform the midcourse and terminal phases (non-OLO) of the spacecraft mission?

Referring to the required guidance functions which are enumerated in paragraph a of this subsection, the following equipment must be added to the basic OLF guidance system for each OLO.

(1,2) None

(3) Add IR horizon tracker; program the computer to activate system to perform functions of unmanned payload guidance system during ascent.

(4) None (manual star tracker on inertial platform is sufficient).

(5) Accelerometers on inertial platform must be more accurate and have greater range than in OLF guidance system.

(6) An optical sensor probably must be added to the basic OLF guidance system. See paragraph d of this subsection.

(7) Minor changes to provide radar inertial mixing. Radar can then measure range and range rate to 4000 nm from moon.

(8) Minor-similar to Tug/Ferry guidance system.

Only minor changes and the addition of an IR horizon tracker are required to enable the basic OLF guidance system to perform the guidance functions required of the spacecraft in the Orbital Launch Operation. The question of midcourse and terminal guidance is discussed in paragraph d of this subsection where it is shown that the elements of the basic OLF guidance system can satisfy the requirements of these mission phases, if small changes are made and an optical sensor added

for midcourse guidance. Also, the operating time required of the spacecraft guidance system will full capability and 99 percent reliability is 40 hours (from Appendix B) compared to 20 hours for the OLF guidance. Paragraph e of this subsection discusses the addition of redundancy to achieve the required reliability during this longer operating time.

The basic (primary mode) system recommended for the spacecraft guidance consists of:

- Inertial platform with manual star tracker (one concept is described in Appendix D)
- Multipurpose radar (described in Appendix D)
- Digital computer
- IR horizon tracker
- Optical sensor for midcourse guidance (one concept is described in paragraph d of this subsection)
- Displays.

c. System Operation in the Primary Mode

Table 3-9 lists the primary functions performed by the spacecraft vehicle and indicates those elements of the system required to accomplish each of the functions. Back-up operation (to be discussed later in this subsection) is also indicated in this table. System operation of the Spacecraft guidance system for the earth orbiting and reentry phase of the lunar mission is identical to the combined operation of the payload and Tug/Ferry and is presented in subsections 3-2, 3-3 and 3-4. In addition, the Spacecraft will have the capability of midcourse and terminal guidance, the operation of which is briefly described.

(1) Midcourse - Once the incremental velocity program, required to place the Spacecraft in a trans-lunar trajectory, has been completed, the Spacecraft guidance system will be deactivated. During the midcourse portion of the lunar mission, the astronaut will be required to make a number of manual position fixes using the multiple image optics and performing the manual computations. If his calculations indicate deviations from an allowable error volume around the nominal trajectory, the system will be activated and the manual position fix information inserted into the computer, which will then initial a corrective thrusting program. It is anticpated that during a normal midcourse mission, the system will be activated twice for vernier trajectory correction.

(2) Terminal Guidance and Lunar Approach - As the Spacecraft approaches the moon, the spacecraft guidance system will be reactivated. At a range of approximately 20,000 nm, the IR horizon tracker will acquire the moon and supply the direction of the vertical. The IR horizon tracker will also provide range and range rate data of sufficient accuracy to position the radar range gate enabling the radar to lock on at ranges in the order of 4,000 nm. The radar data is then mixed with the accelerometer derived data to supply accurate range and range rate information that may be used up to the final descent before touchdown. The vertical information will be supplied by the horizon tracker to an altitude of 250 nm, below this altitude the IR horizon tracker vertical becomes inaccurate and the vertical will be derived from the multipurpose radar using its conical scanning capabilities.

Primary Mode and Redundancy with fu

Guidance Function	Primary Equipment
Orbit Determination	
1) Local Vertical	1) Multipurpose radar
2) Altitude	2) Multipurpose radar
3) Two lines in space	3) Manual star tracker
Attitude Determination and Control	
1) Two lines in space	1) Manual star tracker
2) Maintain References	2) Gyro stabilized platform
ΔV Control	
1) Computation	1) Digital Computer
2) Sensing	2) Accelerometers
Line Of Sight	
1) Acquisition	1) Multipurpose radar and payload beacon

TABLE 3-9

PRIMARY AND REDUNDANT MODES OF OPERATION FOR THE SPACECRAFT GUIDA

11 capability		Further Redundancy,
Back up Equipment	Equipment	Remarks
1) Redundant radar	1) IR Horizon Tracker	1) Horizon Tracker can de mine vertical to requi accuracy
2) Redundant radar	2) IR Horizon Tracker	2) Horizon Tracker requi considerable smoothing achieve required altit accuracy
3) Redundant electronics and servos	3) Manual positioning and readout	
1) Redundant electronics and servos	1) Manual positioning and readout	
2) Redundant Gyro and platform electronics	2) Vehicle Stabilization System	2) Vehicle Attitude will maintained to a few degrees
1) Redundant digital computer	1,2 Manual computation and control	1,2 Accuracy limited by tolerance on engine vernier thrust
1) Redundant radar and payload beacon	1) Periscope	1,2,3,4,5, Periscope requ that target be illumina by sunlight. Useable r 30 nm.

Primary Mode and Redundancy with full cap

Guidance Function	Primary Equipment	Back
Line Of Sight (Cont)		
2) Bearing angles	2) Multipurpose radar and pay-load beacon	2) R a b
3) Range	3) Multipurpose radar and pay-load beacon	3) R a b
4) Range rate	4) Multipurpose radar and pay-load beacon	4) R a b
5) Angular rate	5) Multipurpose radar payload beacon plus computer	5) R a b
Re-entry		
1) Attitude Reference	1) Gyro stabilized platform	1) R a e
2) ΔV Control Computation Sensing	2) Digital computer accelerometers	2) R d
3) Acceleration (g load)	3) Accelerometers plus Computer	
4) Vertical Velocity	4) Radar inertial mixing	4) R
5) Altitude	5) Radar inertial mix	5) R

TABLE 3-9 (Cont)

RIMARY AND REDUNDANT MODES OF OPERATION FOR THE SPACECRAFT GU

ability	Further Redundancy	
up Equipment	Equipment	Remarks
edundant radar nd payload eacon	2) Periscope	
edundant radar nd payload eacon	3) Periscope with Stadiametric ranging	3,4 Stadiametic ranging useable range of 5 nm
edundant radar nd payload eacon	4) Periscope with Stadiametric ranging	
edundant radar nd payload eacon	5) Periscope	
edundant gyro nd platform lectronics	1) Horizon tracker and periscope	1) This back-up can be during the initial p of Re-entry. Can nc used in the atmosphe
edundant igital computer	2) Manual computation and control	2) Accuracy limited by erance on engine ver thrust
edundant radar	4) Radar or accelerometer	4) Radar not useable du large period of re-e due to plasma effect Accelerometer has bu up error in pure ine tial operation.
edundant radar	5) Radar or accelerometer plus complete integration	5) Same as 4) above

Primary Mode and Redundancy with full capab		
Guidance Function	Primary Equipment	Back u
Terminal Guidance and Lunar Approach (Additional Requirements)		
1a) High Altitude Determination. Set radar range gate	1a) Horizon tracker	1a) Re zo he
1b) Determine altitude range, range rate	1b) Radar inertial mixing	1b) Re pl tr ra
2) Moon observation	2) Periscope	

TABLE 3-9 (Cont)

MARY AND REDUNDANT MODES OF OPERATION FOR THE SPACECRAFT GUID

ility	Further Redundancy,	
) Equipment	Equipment	Remarks
lundant hori- n tracker ad	1a) Periscope with stadiametric ranging.	1a) Limited accuracy
lundant gyro, atform elec- onics and lar	1b) Horizon tracker plus computer	1b) Limited accuracy

Primary Mode and Redundancy with full capab

Guidance Function	Primary Equipment	Back u
Payload Functions		
1) Wake-up and Stabilize	1) Inertial platform	1) Redu pla ele
2) Establish two lines	2) Horizon tracker platform gyros (gyro compassing)	2) Redu pla ele
3) Maintain attitude	3) Inertial platform	3) Redu pla ele
4) Align thrust and measure	4) Inertial platform accelerometers and digital computer	4) Redu pla ele
Midcourse Guidance Functions		
1) Determine trajectory (number of pos. fixes)	1) Multiple image optics plus manual aids	
2) Compute incremental velocities to correct trajectory	2) Digital computer	2) Redu tal
3) Sense incremental velocity	3) Accelerometers on inertial platform	3) Redu for

TABLE 3-9 (Cont)

ARY AND REDUNDANT MODES OF OPERATION FOR THE SPACECRAFT GUID

lity		Further Redundancy, U
Equipment	Equipment	Remarks
ndant gyro form tronics ndant gyro form tronics ndant gyro form tronics ndant gyro form tronics		
	1) Manual platform mounted star tracker	1) Platform must be energ Since only one star ca tracked at a time, con erable computation is required.
ndant digi- computer	2) Manual computation	2) Limited accuracy. Ade to permit return to ea vicinity
ndant plat- electronics	3) Manual computa- tion and control	3) Accuracy limited by to ance on engine vernier thrust

d. Midcourse and Terminal Guidance

Discussions concerning the midcourse, lunar approach and terminal phases have not been considered as part of the Orbital Launch Operations. Since it is essential that the spacecraft guidance system be efficiently integrated, however, the requirements and equipment for these phases will be considered briefly.

The guidance functions required during these mission phases are:

Mission Phase	Guidance Function
Midcourse	Determine deviations from desired trajectory; perform required corrections. This requires measurement of angles between moon, earth, and stars plus thrust vectoring.
Lunar approach and orbit	Predict perilune; determine orbital plane and parameters.
Lunar landing	Location on moon; time to deorbit; altitude and rate during descent; vertical.
Return trip	Alignment and midcourse functions.
Reenter earth's atmosphere	Enter atmosphere at acceptable location; maintain vehicle in reentry corridor; land in acceptable area.

(1) Midcourse Guidance

During the midcourse phase of a lunar journey, the guidance system must determine the vehicle's trajectory, compare this to a planned trajectory, predict errors in perilune, and indicate the extent of any corrections necessary. The most probable method of accomplishing this is by suitably smoothing the data from a series of position fixes. Each position fix would be made by measuring angles between stars and the earth or moon. Celestial measurements have the advantage that the heavenly bodies are omnipresent, highly accurate references, which can be utilized by relatively simple instruments which require no energy during operation.

One scheme for celestial position fixing is to use the manual star tracker on the inertial platform, which is described in Appendix D and is a part of the tug guidance system. This has two disadvantages: (a) it requires that the inertial platform be operating to provide a stable base, thus consuming both energy and operating life, (b) a relatively large amount of computation is required since the Spacecraft travels along its (unknown) trajectory between successive measurements.

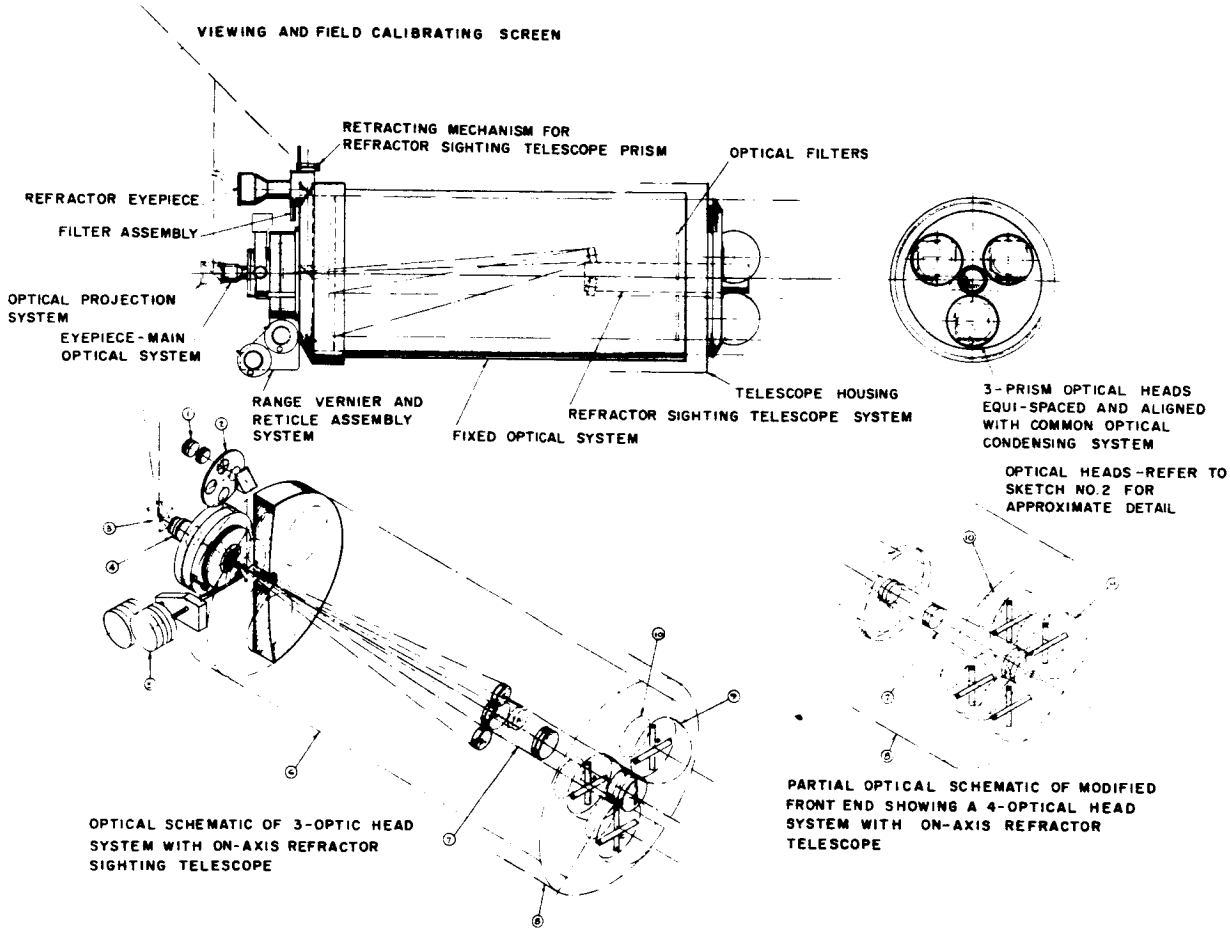
The first disadvantage can be eliminated and the second minimized if an optical guidance element is added which can measure angles between three celestial bodies simultaneously. Figure 3-8 is a sketch of one concept of such a device.

It consists of:

- Sighting head containing three optical heads* for the simultaneous viewing of three celestial bodies

- - - - - - -

*Optical heads contain manual positioned 2 degrees of freedom Dove prisms.



1. REFRACTOR EYEPIECE
2. FILTER ASSEMBLY
3. OPTIC-PROJECTION SYSTEM
4. EYEPIECE-MAIN OPTICAL SYSTEM
5. RANGE VERNIER AND RETICLE ASSEMBLY SYSTEM
6. FIXED OPTICAL SYSTEM
7. REFRACTOR SIGHTING TELESCOPE SYSTEM
8. TELESCOPING HOUSING
9. 3-PRISM OPTICAL HEADS* EQUI-SPACED AND ALIGNED WITH COMMON OPTICAL CONDENSING SYSTEM * OPTICAL HEADS - REFER TO SKETCH NO.2 FOR APPROX. DETAIL
10. OPTICAL FILTERS

Figure 3-8. Multiple Image Optics - One Concept

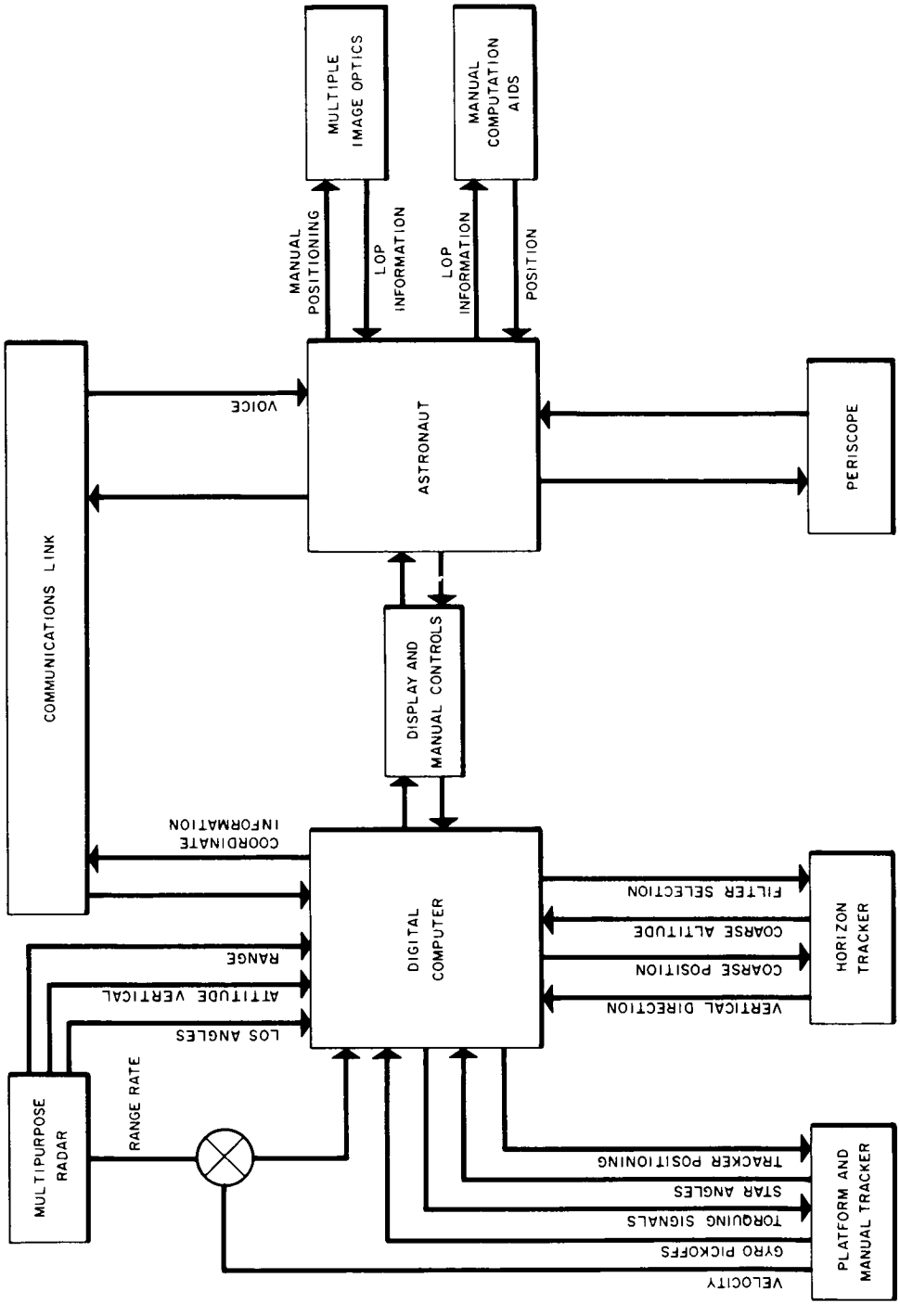


Figure 3-9. Spacecraft Guidance System, (Primary Mode of Operation)

and a refractor telescope system which serves as a spotting scope to establish a base line sighting reference.

- Viewing head containing operator controls, manual readouts, main optical system reflection mirror and operator eyepieces.

Its operation during a typical mission is described.

At intervals throughout the midcourse phase, the present position of the Spacecraft may be defined by the intersection of a line-of-position (LOP) to the moon and to the earth. To establish one LOP, the multiple image optical device is used to simultaneously sight on two stars and the center of one planet. The second LOP is determined the same way but no more than 15-20 minutes later. Angular data manually readout at each sighting is then converted to present position using either manual or automatic computational techniques. Since knowledge of a single position fix in space is not sufficient for the determination of a trajectory; data from a series of position fixes must be smoothed and converted into the desired trajectory information. A digital computer can be normally used for this task, however, this computation could be done manually or with the aid of a mechanical computer.

Figure 3-9 is a block diagram of the spacecraft system in the primary mode of operation. Included in this diagram is the horizon scanner and multiple image optics required for midcourse and terminal guidance.

(2) Lunar Approach and Terminal Guidance

In order to predict perilune of the lunar approach hyperbola, injection into a lunar inspection orbit, and orbital

transfers for rendezvous and lunar landing, it is necessary for the guidance system to measure range, range rate, direction and rate of the local vertical, tangential velocity and determine the selenocentric inclination of the orbital plane. If these quantities can be measured at distances of several thousand miles from the moon, sufficient time is given to the astronaut to verify that the guidance system is performing satisfactorily and to make trajectory corrections before the approach becomes critical. Further, if most of these quantities can be measured directly in selenocentric coordinates, the astronaut's computation task will be considerably simplified.

With no additional guidance elements and only minor changes made to some of the existing elements, the extended configuration guidance system can perform all of these terminal functions with suitable accuracy to ranges of about 3000 nm. The changes to be made are principly within the radar area* (see Appendix D) and are as follows:

- Radar provided with a longer pulse width to determine range to moon from 3000 to 4000 nm
- Radar provided with a shorter prf
- Radar-inertial mixer provided with a longer integration time
- Horizon tracker or periscope tie-in with mixer to provide approximate range and direction to moon or earth at about 20,000 nm and to enable radar to lock-on at about 4,000 nm.

- - - - - - -

*Further information may be found in a Sperry technical paper "LASSO - A Landing and Approach System for Space Vehicles" to be presented at the 1962 Military Electronics Convention in Los Angeles, California.

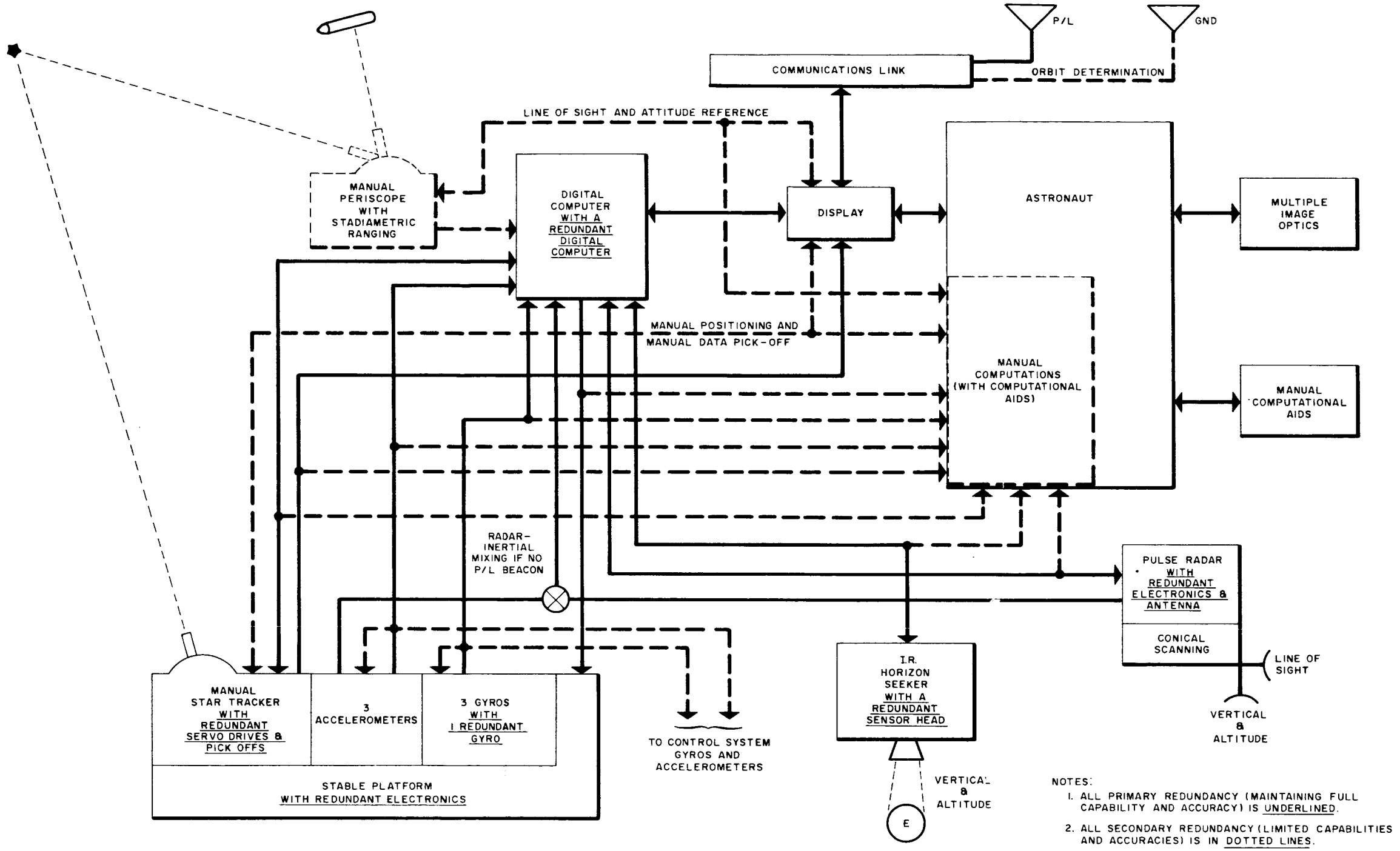


Figure 3-10. Primary and Secondary Redundancy for the Spacecraft Guidance System

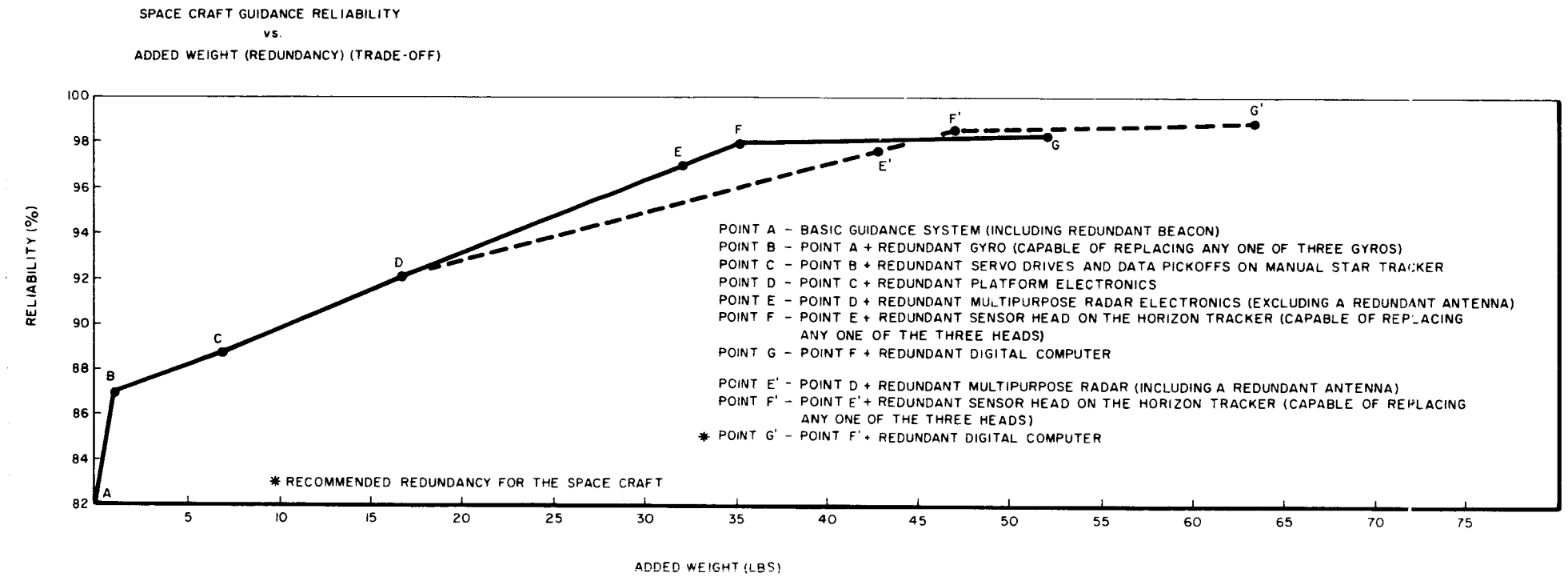


Figure 3-11. Spacecraft Guidance Reliability versus Added Weight (Redundancy) (Tradeoff)

e. Redundancy in Spacecraft Guidance

The spacecraft guidance system includes two levels of redundancy as do the OLF and Tug/Ferry guidance: (1) redundancy is added to the primary mode elements to attain 99 percent reliability during the required 40 hour operating time, with full capability; (2) additional backup, usually manual and usually having reduced capability, is added for use during the approximately 1 percent of the missions when failure of the full capability equipment might be expected.

Figure 3-10 is a block diagram of the recommended guidance system for the spacecraft, showing these two kinds of redundancy. Table 3-9 lists the primary and redundant means of performing each guidance function.

(1) Full Capability Redundancy.

The elements of the primary mode guidance system contribute to the overall unreliability of the spacecraft guidance system in the following order:

- Three gyros
- Multipurpose radar
- Platform electronics
- Manual star tracker
- Horizon tracker
- Digital computer
- Three accelerometers
- Radar beacon
- Multiple image optics

Figure 3-11 is a tradeoff curve showing the increase in reliability for 40 hours operating time, with full capability, as weight is increased by adding redundancy. In order to approximately reach 99 percent reliability, the following redundancy is included in the recommended system:

- Redundant multipurpose radar (antenna included)
- Redundant gyro (capable of replacing any one of the three)
- Redundant servo drives and data pickoffs on the manual star tracker
- Redundant platform electronics
- Redundant sensor head on the horizon tracker (capable of replacing any one of the three heads)
- Redundant digital computer.

This added redundancy increases the spacecraft guidance reliability from 0.8217 to 0.9882.

(2) Additional Backups

Backups not necessarily with full capability, are provided as follows:

- IR horizon tracker for altitude and vertical, in case of failure of the multipurpose radar
- A manual periscope with a simple provision for estimating range to the object sighted; this will provide a back-up for acquisition of target, angle and range, in case of failure of these functions of the multipurpose radar.

- Attitude control systems gyros and accelerometers will be used in case of failure of the corresponding units of the inertial platform
- Manual positioning of the star tracker telescope, and manual readout of data will be available in case of failure of the electronics
- Essential computations and control functions will be performed manually in case of failure of digital computer or control elements.

f. Computer Requirements

The following is a list of functions which must be performed by the spacecraft digital computer. A description of the computer is covered in Volume V of this study.

- Compute orbit from sensor inputs and stored celestial data
- Compute attitude from sensor inputs and stored celestial data
- Compute signals for radar altimeter antenna stabilization
- Compute approximate angular positioning of pulse radar interrogator antenna and compute stabilization signals
- Compute range gate reference signals from inertial information during periods of thrusting
- Compute range rate and angular rates between Tug/Ferry and target
- Compute ΔV program for rendezvous with target
- Compute drift of Spacecraft from synchronous orbit, if required

- Compute Spacecraft predicted ephemeris, weight, fuel load
- Compute and correct drift of attitude control system gyros
- Compute required attitude and ΔV for passing through reentry window
- Compute flight path control signals to remain in reentry corridor
- Perform gyro compassing operation to determine vehicle alignment in orbit
- Process manually obtained celestial position data during midcourse portion of lunar mission
- Compute ΔV program for trajectory correction during lunar mission
- Perform mixing of radar and accelerometer data during lunar approach.

g. Display Requirement

The following signals should be displayed to assist the astronaut in monitoring and controlling the various guidance functions. Additional study should be given to the selection of the types of displays and the size and location of the display unit in the spacecraft.

- Line-of-sight - Target acquisition signal, range, range rates, and bearing angles to target
- Station keeping - Attitude, altitude, time, position, velocity relative to earth or moon
- Control of ΔV's - Comparison of programmed ΔV's and integrated accelerometer outputs

- Terminal rendezvous and docking - Display showing the relative positions of Spacecraft and target, and an indication of the maximum safe closing rate
- Reentry - Altitude, velocity, attitude, vehicle environmental conditions
- Station safety - Relative position to nearest ground tracking station, vehicle environmental conditions, fuel.

h. System Characteristics

Table 3-10 lists the weight, power and volume of the recommended system; it includes the recommended redundancy.

TABLE 3-10

SPACECRAFT SYSTEM CHARACTERISTICS

Component	Weight Pounds	Power Watts	Volume Cubic Inches
Inertial platform	42*	95*	2000*
Platform electronics	20*	70*	500*
Horizon tracker	12*	8*	1000*
Horizon tracker electronics	5	5	300
Multipurpose radar	20*	70*	500
Multiple image optics	25	-	2000
Manual computation aids	3	-	150

*This includes redundancy required to attain a reliability of 0.9882.

TABLE 3-10 (Cont)

Component	Weight Pounds	Power Watts	Volume Cubic Inches
Periscope with stadia-metric ranging	ND	-	ND
Display console	ND	ND	ND
**Total	127**	248**	6450**

**Does not include digital computer

ND Not Determined

i. Component Performance Requirements

Table 3-11 indicates the accuracies required for major components of the Spacecraft in order for it to fulfill its mission requirements. Computer specifications have not been included, since they may be seen as a portion of the computer system presentation.

TABLE 3-11

SPACECRAFT COMPONENT ACCURACY SPECIFICATIONS

Components	
Gyros	0.05^{o}/Hr random drift; 0.05^{o}/Hr/g; 0.015^{o}/Hr/g^2
Accelerometers	0.002%
Horizon scanner	6 minutes at 300 NM altitude above earth

TABLE 3-11 (Cont)

Components	
Manual star tracker	25 seconds RMS
Radar and beacon	See radar description Appendix D
Multiple image optics	30 seconds RMS for determing earth or moon center line

SECTION IV

MAN-MACHINE RELATIONS DURING RENDEZVOUS

4-1. INTRODUCTION

While the previous section lists the recommended guidance equipment for each of the vehicles in OLO, the purpose of this section is to consider the factors that influence how this equipment is to be used in combination with man to obtain an effective utilization of both man and machine. The more important factors influencing the rendezvous operation are

- The formulation of effective guidance laws
- The allocation of tasks between man and machine (that is, degree of automaticity)
- The selection of an interface display configuration.

In the previous sections, it has been possible to establish fairly detailed requirements and candidate systems for the "hardware", the sensors and the computers, and to evaluate these and select the best. In the case of the "software", the guidance rule, and the allocation of tasks between the hardware and the astronauts, and in the selection of the display, it is not possible to proceed so directly. Selection of the software and the display configuration requires extensions of the present study. These components receive inputs from the OLO concept and vehicle design as they are firmed up, and include simulator studies with human operating models of the recommended guidance

equipment in the simulator. This section therefore indicates certain problem areas, and outlines the fields from which the software and display candidates will be established. Examples of a few attractive display schemes are presented in subsection 4-4.

4-2. THE SELECTION OF GUIDANCE LAWS

This particular subsection deals with the little discussed area of "software", that is, the formulation of laws upon which the utilization of equipment and other decisions can be based. Certain of these laws can be postulated by mission considerations such as

- Number of required rendezvous
- Guidance accuracy
- Energy available
- Time constraints.

The laws, however, can only be finalized by including specific system design characteristics such as

- Propulsion characteristics
- Vehicle characteristics
- Role of the ground facilities
- Human safety.

Certain problem areas warrant further investigation before any guidance laws can be finalized. These include some of the ground rules that were necessarily postulated in Section II, based on preliminary

a. Reliability Model

Reliability will be defined, in the most general way, as the probability that the system will perform its intended mission satisfactorily. For the rendezvous manuever, the system includes the astronaut, his vehicle (OLF, Tug or Spacecraft), and the payload. Performance criteria for terminal rendezvous may be specified by position, attitude and associated rate limits which must not be exceeded for the start of the docking maneuver. These limits define gates for satisfactory entry into the docking phase of the operation. The reliability of the system is numerically the probability that the terminal rendezvous maneuver will end within these gates.

Consider the system presented in block diagram form in figure 4-1. The primary control/display system may or may not include the human operator, dependent on whether the system is manual or fully automatic. In either case, the performance of the primary system is monitored by the astronaut through the monitor system. If the astronaut decides that the performance of the primary system is unsatisfactory, he will switch to the standby system. The latter may be fully automatic or manual.

There are several distinct reliability components which must be considered in evaluating the total reliability of the system R_T (table 4-1). These are:

R_p = probability that performance of primary system will be satisfactory

R_M = conditional probability that the astronaut will be satisfied with system performance, given satisfactory primary system performance

$R_{\bar{M}}$ = conditional probability that astronaut will detect degraded system performance, if the primary system performance is in fact degraded

R_S = conditional probability that performance of standby system will be satisfactory, if the astronaut elects to use it.

Six possible, mutually exclusive conditions involving system performance are itemized in table 4-1. Of these, conditions 1, 2 and 4 augment system reliability, and the sum of their respective probabilities represents the system reliability R_T. The components of unreliability (Q) are the probabilities for conditions 3, 5 and 6. Since the six conditions are mutually exclusive, ΣR and ΣQ are complementary, that is, their sum is unity.

The probabilities R_P and R_S are the results of hardware reliability considerations if these systems are fully automatic. With manual inputs, R_P and/or R_S include both hardware reliability and human performance, as obtained from suitable simulated tests which provide the required statistical data.

The equation for R_T in table 4-1 shows all the component elements of system performance R_P, R_S, R_M and $R_{\bar{M}}$, and their relative influences. This relationship provides a basis for system design decisions. The parameters $(\partial R_T/\partial R_X)$ are the augmentation in system reliability (ΔR_X). There are four partial derivatives involved, one for each of the probabilities R_P, R_S, R_M and $R_{\bar{M}}$. These parameters may be used in trade-off analyses for system design.

investigations, to serve as a basis for the remainder of the effort contained in this volume. The problem areas requiring further study are

- The effects of earth launch time deviations
- Use of synchronous orbits for the OLF
- The tradeoff between energy and time to determine number of revolutions of the phasing ellipse for a given phase angle
- Criteria for thrusting during terminal rendezvous phases
- Criteria for thrusting to correct orbital plane
- Role of ground facilities
- Smoothing philosophies for orbit determination
- The role of the astronaut in accomplishing the guidance functions.

As an example of the ramifications involved in these types of studies, consider the use of synchronous OLF orbits with respect to an earth launch site. This type of orbit greatly simplifies the earth launch operation schedule in addition to maintaining the OLF in a position locus consistent with a scheduled lunar launch. Since this type of orbit allows for one launch per day into compatible rendezvous, this insures a total characteristic velocity for the payload that is near minimum. The feasibility of maintaining a particular orbit in synchronism is dependent upon the following factors:

- The ability to predict and account for the drag and oblateness perturbations

- The fuel required for corrective maneuvers
- The requirements of other disciplines such as communications, orbital maintenance, and logistics and vulnerability.

The alternative to synchronous orbits is the use of a phasing ellipse whose apogee is tangent to the OLF orbit. Where large phase angles are involved, the decision between the number of revolutions and energy expended for the ellipse must be made. If the number of revolutions is more than a few, this may affect the design of the payload guidance and control system, and may imply long range detection capabilities on the OLF.

This discussion on the use of synchronous orbits is indicative of the myriad of problem areas that must be studied for solutions prior to the formulation of guidance laws. These laws are necessary to specify the requirements for the allocation of tasks between man and machine.

4-3. ALLOCATION OF MAN-MACHINE TASKS

The distribution of tasks between man and machine in a space cockpit must satisfy the following inter-related requirements:

- An acceptable level of system reliability must be achieved
- The human operator (astronaut) must not be overloaded.

Systematic approaches to the solution of the problem require models of the system which provide quantitative data for design decisions. Each of these is considered in the following paragraphs.

TABLE 4-1

SYSTEM RELIABILITY BREAKDOWN

Condition No.	Performance of Primary System	Effectiveness of Monitor Function	Performance of Standby System	Reliability (R) and Unreliability (Q) Components
1	Satisfactory	Unsatisfactory	-	$R_1 = R_P R_M$
2	Satisfactory	Unsatisfactory	Satisfactory	$R_2 = R_P (1 - R_M) R_S$
3	Satisfactory	Unsatisfactory	Degraded	$Q_3 = R_P (1 - R_M)(1 - R_S)$
4	Degraded	Satisfactory	Satisfactory	$R_4 = (1 - R_P) R_{\bar{M}} R_S$
5	Degraded	Satisfactory	Degraded	$Q_5 = (1 - R_P) R_{\bar{M}}(1 - R_S)$
6	Degraded	Unsatisfactory	-	$Q_6 = (1 - R_P)(1 - R_{\bar{M}})$
				$\Sigma R + \Sigma Q = 1.0$

$$R_T = R_1 + R_2 + R_4 = R_P R_M + R_P (1 - R_M) R_S + (1 - R_P) R_{\bar{M}} R_S$$

$$= R_P \left[R_M + R_S (1 - R_M - R_{\bar{M}})\right] + R_{\bar{M}} R_S$$

If $R_M = R_{\bar{M}} = 1.0$, $R_T = R_P + R_S - R_P R_S$

b. Operator Loading

The tasks of the astronaut in the cockpit of his space vehicle are manifold, and these may be categorized in a number of ways. For system design purposes, it may be convenient to classify them as control, monitor and communication functions, which may be considered orthogonal as indicated in figure 4-2. The three coordinates may be scaled in behavioral units representing degrees of operator loading. The total load on the astronaut at each instant during the mission may be represented by the loading vector L from the origin to the terminus in space corresponding to the three load components prevailing at that instant. During the course of the mission, the astronaut's loading generates a curve in three dimensional space. The data for describing the curve may be obtained from a time line analysis of operator tasks. The limits of capability for the astronaut is represented by a surface in this same space. The preclusion of operator overloading dictates that no point on the load curve should be outside the capability envelope.

The foregoing analysis may be used to ensure against overloading the astronaut in the course of the initial system layout. This procedure will establish in a preliminary way that the tasks allocated to the astronaut by virtue of the system configuration are reasonable. System performance in a particular system configuration must be determined by the reliability analysis presented in paragraph 4-3 a.

4-4. INTERFACE DISPLAY CONFIGURATIONS

Recent literature in the display area has been mainly concerned with the optimum utilization of man's

capabilities in the rendezvous mission and the best manner of presenting the information he needs to safely accomplish the mission. The simulation programs conducted by NASA[1] and other agencies indicate that trained pilots can perform the mission, even under severe operating conditions, such as thrust misalignments, with less than 20 percent fuel expenditure above that of the two-impulse nominal amount. The discussion here will be mainly concerned with some of the problem areas that have been identified by the mentioned literature. One of these problem areas is the mixing of dialed instruments, visual displays and actual "eyeballing" to obtain the optimum interface configuration. These discussions apply to the Tug, the OLF and the Spacecraft. The fact that the OLF remotely controls the payload is mainly a telemetering problem. A previous study[2] as well as extrapolation of aeronautical experiences in somewhat similar operations such as aerial refueling, indicates that the terminal maneuvers will probably be accomplished most effectively in a manual-visual operation involving a human astronaut. This approach may be contrasted with the use of an automatic system for accomplishing the precise vernier corrections of position, attitude and their rates for rendezvous. The conclusion in favor of manual operation is dictated by matters of system complexity, weight and reliability.

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[1]Brissendun, R. F., Burton, B. B., Foudriat, E. C., and Whitten, J. B.: Analog Simulation of a Pilot-Controlled Rendezvous NASA TN-D-747. April 1961.

[2]Levin, E. and Ward, J.: Manned Control of Orbital Rendezvous, The Rand Corporation, October 1959.

The terminal rendezvous maneuver for which the display has been generated is shown in figure 4-3 where the payload is closing on the target OLF during the last portion of the phasing ellipse on the direct ascent from earth. The vehicle is assumed to have bi-directional longitudinal and normal thrusting, as well as pitching moment capability for attitude control. All controls may be either on-off or proportional type.

It is assumed that terminal closure will be effected by manually controlled impulse inputs by the astronaut who is being given suitable information to accomplish these tasks. It is generally considered that the lowest weight in orbit and the maximum reliability will be obtained when system complexity is minimized and these minimal systems use the abilities of the pilot to a practicable maximum (references 3 and 4). Even with the advent of more sophisticated automatic control systems for terminal operation, direct manual control will still be useful for standby purposes in case of system malfunction.

Two classes of displays have been considered, a projective display and a more conventional panel mounted instrument display. These represent display classifications with significant differences in complexity with respect to to the instrumentation art. The more complex projective display may naturally be expected to provide operational

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[3]Wolowicz, C. H., Drake, H. M. and Videan, E. N.: Simulator Investigation of Controls and Display Required for Terminal Phase of Coplanar Orbital Rendezvous. NASA TN D-511, October 1961.

[4]Kurbjun, M.C., Brissenden, R. F., Foudriat, E.CC. and Burton, B. B.: Pilot Control of Rendezvous. IAS Paper No. 61-37, January 1961.

display may naturally be expected to provide operational advantages to the astronaut in comparison with the technically simpler panel mounted symbolic instruments. The decision regarding which display system to employ, or perhaps to synthesize a composite system, will depend on the factors mentioned in the following paragraphs a. and b.

a. Projective Display

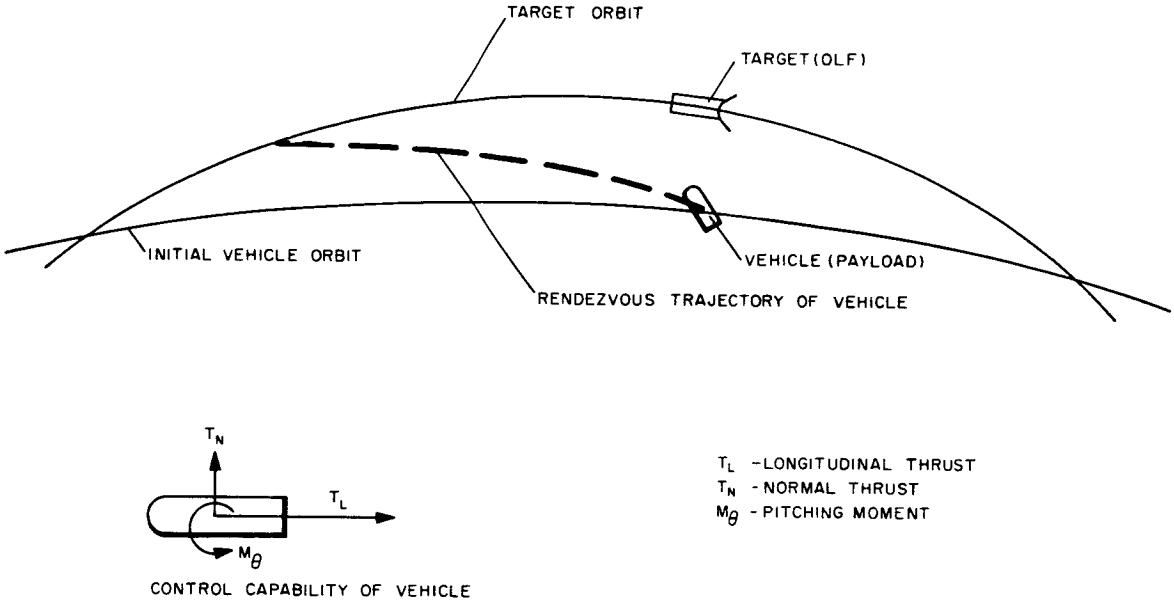
A projective display is one in which collimated images are presented to the human operator at optical infinity, so that binocular convergence and accommodation vision cues are eliminated. These images are therefore phenomenally superimposed on the external visual field of the observer. This implies that the imagery will seem of external origin in the observer's preceptual frame of reference and they will appear superimposed on the external visual field, whether this be the earth or another satellite body. The optical scheme for presenting such projected imagery is presented in figure 4-4. The equipment in the lower half of this illustration, consisting of the CRT, the plane mirror, collimating lens and the combining glass, represents the essential optics. The CRT represents a highly flexible means of generating a large range of images, but the system need not be limited to this display generator. Moving reticles may also be used, but their capability is limited. The orientation of the plane mirror is varied to change position of the entire image constellation within the astronaut's field of view.

The external visual field may be presented simultaneously through the periscope system illustrated in the

upper half of figure 4-4. The essential optics consists of the cylindrical lens and the concave cylindrical mirror, with the plane mirrors providing the periscopic displacement.

The projective display may be adapted to visual flight, instrument flight, or any of a combination of these two modes. Closing the external periscopic aperture makes the operation all instrument, while turning off the projected imagery produces a simple periscope. The utilization of both modes of operation provides a compatible visual-instrument operation in which transition from one mode of operation to another may easily be effected by the astronaut. Either the entire external visual field or any select portion may be subjected to any desired degree of magnification by the interposition of appropriate telescopic optics. The single curvature of the field lens and the concave mirror may be selected to provide a real life (1:1) magnification for the docking operation.

A typical view of the projective display in operation is presented in figure 4-5. The curved surface of the earth represents the true horizon as viewed from the vehicle. The attitude and heading of the vehicle are determined by the relative position and orientation of the attitude reference lines with respect to the reference indices. The indices may be manually adjusted for reference inputs. The transverse velocity vector r_{YZ} represents the relative velocity of the vehicle with respect to the target in a plane normal to the longitudinal (X) axis of the vehicle. The orientation of the target image in the display is a function of the position of the target with respect to the vehicle and the attitude of the vehicle. The size of the target image is made a function of target range to a suitable scale.



CONTROL CAPABILITY OF VEHICLE

T_L - LONGITUDINAL THRUST
T_N - NORMAL THRUST
M_θ - PITCHING MOMENT

Figure 4-3. Rendezvous Trajectory and Vehicle Control Capability

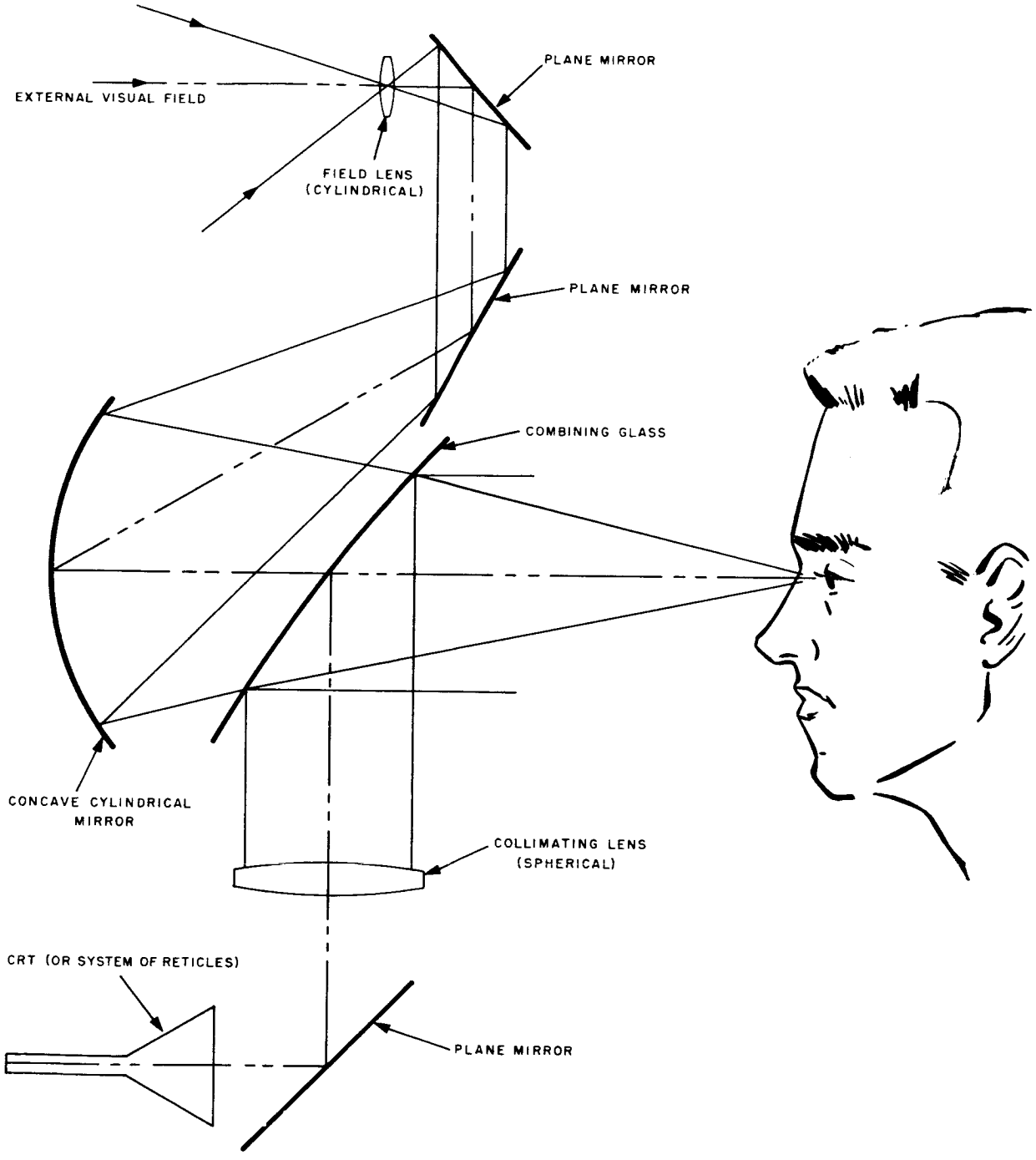


Figure 4-4. Schematic of Projected Display

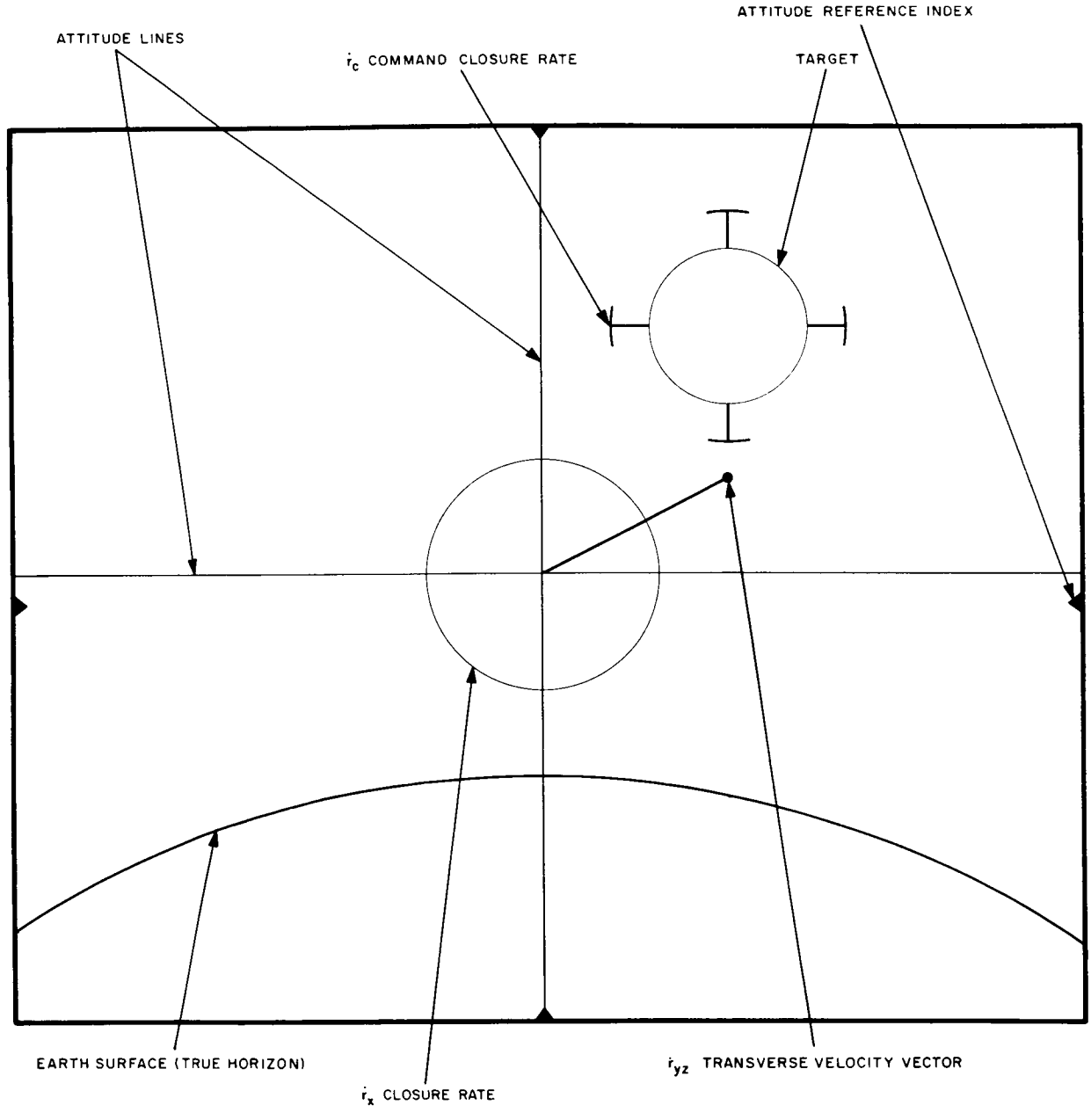


Figure 4-5. Projective Display Presentation

The four peripheral arcs on the target image represent a command closure rate ($\dot{r}_c$) which is a function of target range. The actual closure rate is represented by the circle at the intersection of the attitude lines. When the target image is centered on the display, the closed loop operation for closure may be implemented by tracking the closure rate against the command closure rate, that is, superimposing the closure rate circle on the command closure rate rate arcs as represented in figure 4-6. The difference between the $\dot{r}_x$ and $\dot{r}_c$ images represents a closure rate error to an increasing scale as the target range gets shorter and the size of the target image increases.

b. Panel Display

The use of conventional panel mounted instruments may generally be expected to lead to designs which are technically simpler and lighter in weight. An example of such an array for the terminal rendezvous maneuvers is presented in figure 4-7. These instruments are all of the direct meter movement type, save for the attitude indicator which is a panel mounted servo-driven unit.

Roll, pitch and yaw and their respective rates are presented in the horizontal cluster of four instruments in the center of the display. The line-of-sight to the target from the intercepting vehicle is represented by the group of two cross-pointers above the attitude display. The vertical needle represents azimuth angle and the horizontal needle elevation angle so that the intersection symbolically represents the orientation of the target in the visual field. The crossing of the needles on the center of the instrument represents a situation in which the target is "dead ahead".

Range and range rate to the target are presented in the combined two pointer instrument below the attitude instrument. The needle represents range and the moving index on the perimeter of the display represents closure rate. The matching of the range pointer to the rate index represents closed loop operation in which the closure rate and the range to the target are related by some predetermined guidance law. The sensitivities of the range and rate displays may be selected or varied as a function of distance to the target, so that the display scale is effectively expanded at shorter ranges. The relative sensitivity of the range and range rate pointers will provide the effective coupling constant for the closure maneuver.

c. Simulation Programs

The determination of the effectiveness of any particular display, in advance of rendezvous missions in space, can only be determined by a comprehensive simulation program conducted in a suitable simulator. This device should preferably be of the moving base type with a simulated external visual field with a projected moving target whose size varies as a function of range. The simulated visual field should also contain appropriate imagery representing the view of the earth, and a stellar background. A spherical dome enveloping the moving base simulator, with the astronaut's normal head psotion (mid-interpupillary point) at the center of the sphere will provide a good projection surface for the external field. A sketch of this arrangement is shown in figure 4-8.

The physical finish of this surface and the illumination level should be specified to eliminate visual cues

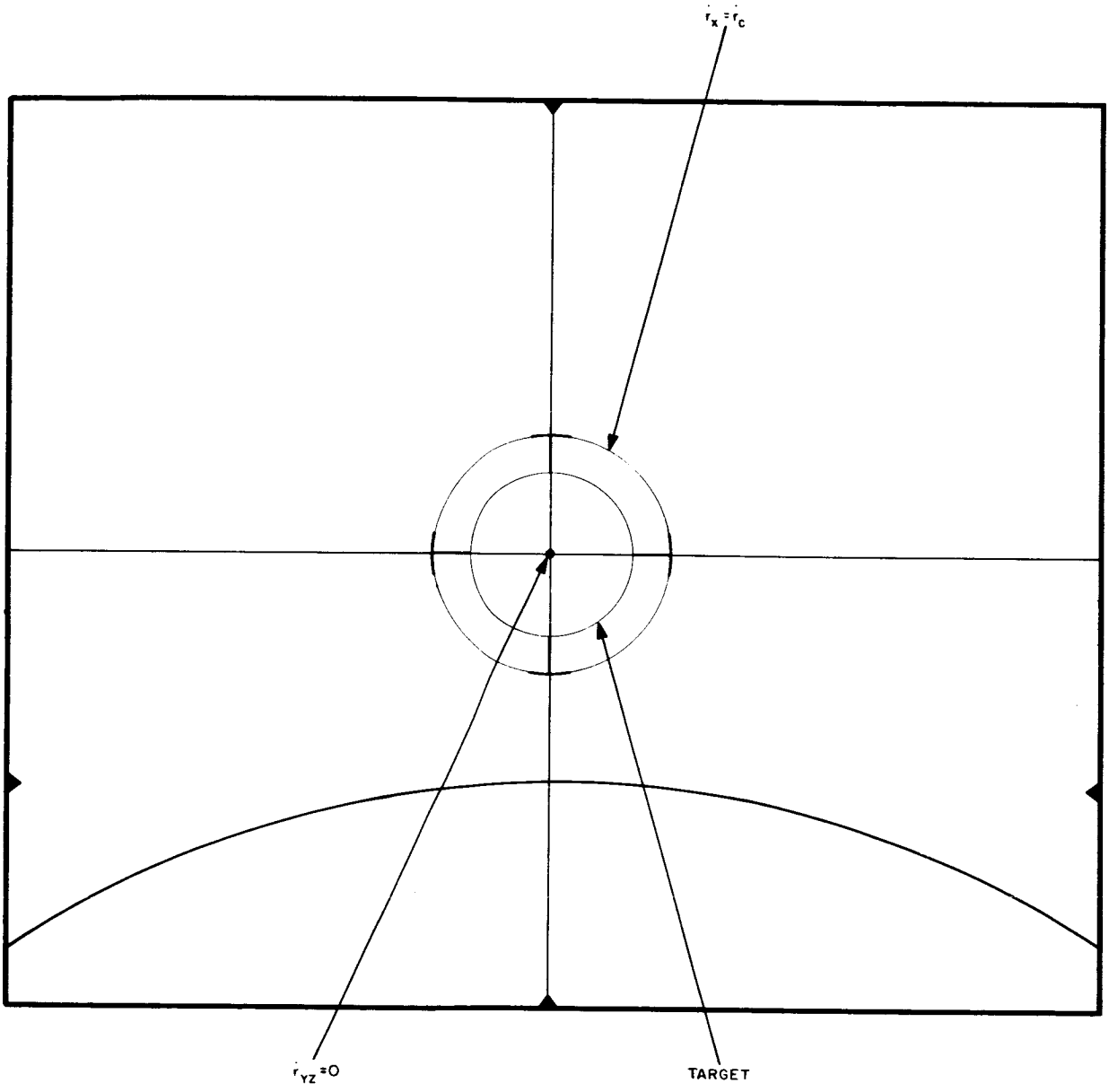


Figure 4-6. Projective Display with Vehicle Tracking Target

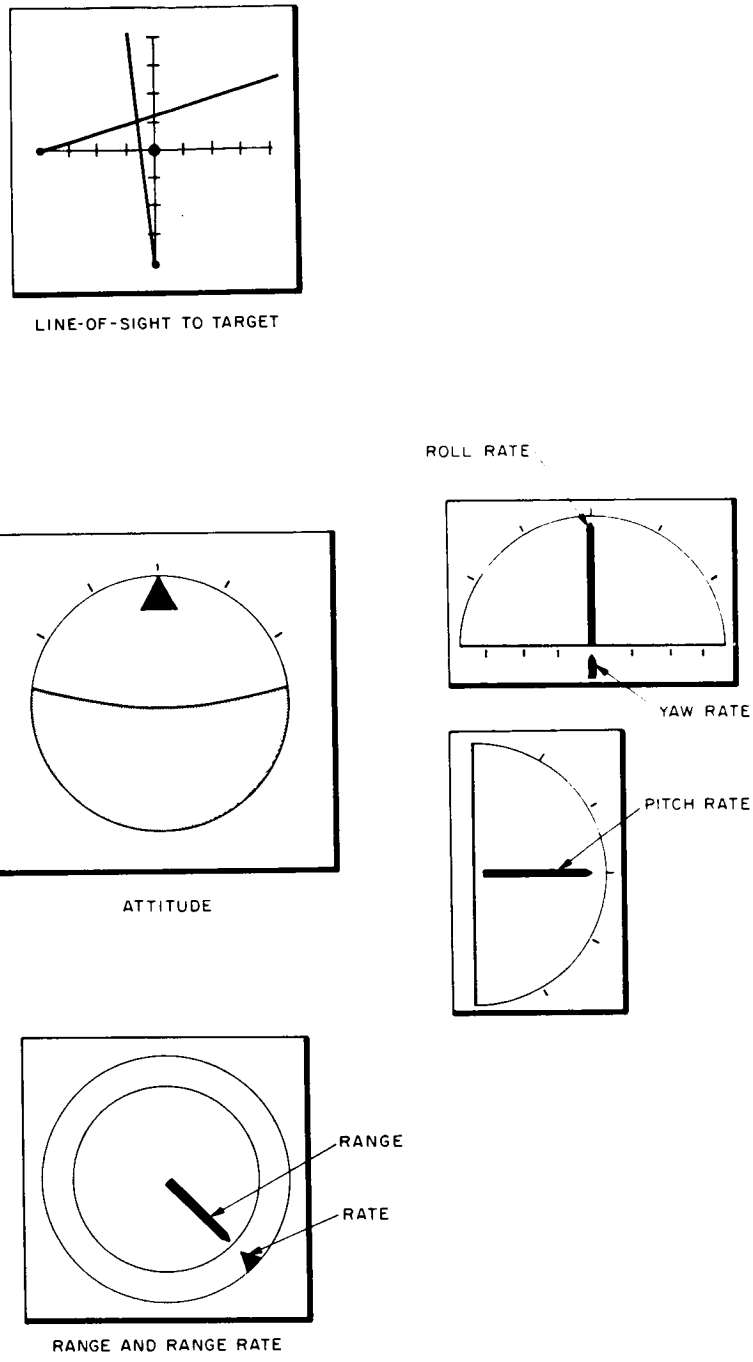


Figure 4-7. Instrument Panel Display

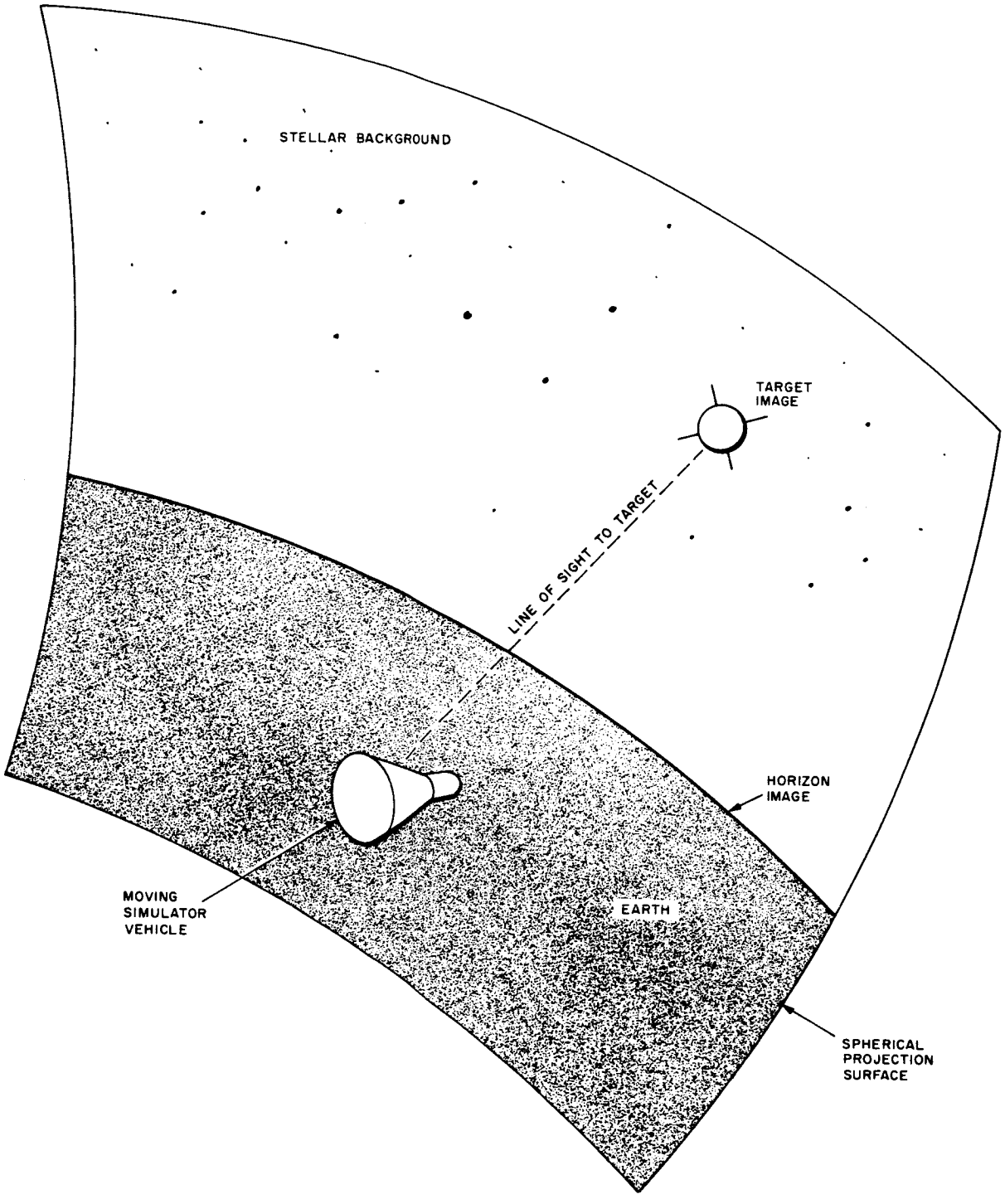


Figure 4-8. Moving Base Simulator with Projected Visual Field

revealing the actual distance of the dome from the simulator vehicle. The six degrees of freedom representing the motion of the vehicle should be suitably divided between simulator base motion and the motion of the project imagery. Combinations of these motions suitably phase may be required to simulate maneuvers involving large angles. This arrangement does not reproduce all the kinesthetic and vestibular cues of vehicle motion for the astronaut, but this is a limitation of in a terrestrial environment.

The simulator will permit the determination of the performance levels achieved by the astronaut, in terms of both trajector and fuel consumption, for a range of initial intercept conditions with each type of display. The range of permissible end values of the terminal rendezvous maneuver will permit a measure of maneuvering success, representing the fraction of terminal maneuvers which result in end values within the docking "gate". A reasonably large number of maneuvers, approximately one hundred per test condition, will be required to produce the statistical data from which performance inferences with narrow confidence limits may be drawn. The moving base simulator may also serve as a trainer for the astronauts, after a particular system configuration has been selected. Extensive training to raise astronaut performance levels for space missions is indicated in view of the associated hazards and costs of these operations.

Rendezvous performance levels may be determined for conditions in which all equipment is operating within normal limits, and for those in which equipment performance is degraded or there are actual malfunctions. Hardware reliability studies will indicate the probabilities associated

with the incidence of equipment malperformance. The simulator studies will yield the conditional probabilities for satisfactory rendezvous performance, given an equipment operating level. The combination of equipment and astronaut performance will therefore provide a direct index of mission success for the terminal rendezvous phase. This data may be used in a quantitative tradeoff analysis involving display reliability, weight and system performance, as a basis for the selection of a display configuration for rendezvous.

SECTION V

PROGRAM CONSIDERATIONS AND RECOMMENDATIONS

5-1. GENERAL

This section summarizes the conclusions from the study and indicates what steps are needed to design guidance systems for the OLO vehicles with efficiency and with reliability demonstrated before Orbital Launch Operations begin.

The study work reported in the previous sections, and in more detail in the Appendixes, has considered the Orbital Launch Operation and has selected a family of operating concepts within which the actual operating concept will probably lie. The vehicles which will require guidance during the operation have been defined, and the guidance functions and their associated accuracy and reliability requirements have been established. Candidate guidance elements and systems have been generated A major effort was devoted to determining realistic estimates of reliability as a means of comparing and selecting the best candidate. Only candidates for which development can begin immediately, using available components and techniques, were considered seriously, and only these are listed in the tables of candidates in Appendix C.

One of the ground rules used in the study is that any vehicle which is manned should have a guidance system capable of completely independent operation. This includes the ability to determine its own orbit, and also to determine the line-of-sight to other vehicles in the rendezvous operation.

One of the major conclusions of the study is that, while meeting the accuracy requirements does not pose a particularly difficult problem, reliability must be a major objective throughout the development, manufacture and test of guidance equipment for OLO.

From the comparison of a relatively large number of ways of accomplishing the required guidance functions, it has become very clear that two specific guidance elements are necessary. These two elements will make possible guidance systems which will perform the functions required in OLO with light weight and high reliability. They are:

- A multipurpose radar for measuring altitude, vertical, angle to a rendezvous target, range and range rate. Reliability is the primary goal in developing this radar.
- A manual star tracker mounted on an inertial platform This is essentially an automatic star tracker in which the sensor and signal processing electronics are replaced by the astronaut's eye, thus eliminating a major reliability problem.

Both of these can be developed using existing techniques and components, and are described conceptually in Appendix D.

After a study of six candidates, the study recommends that the basic guidance system for the manned vehicles in OLO consist of the two guidance elements just described, plus a digital computer and displays. Sufficient redundancy is added to attain the required reliability. Two of the more important features of this system are:

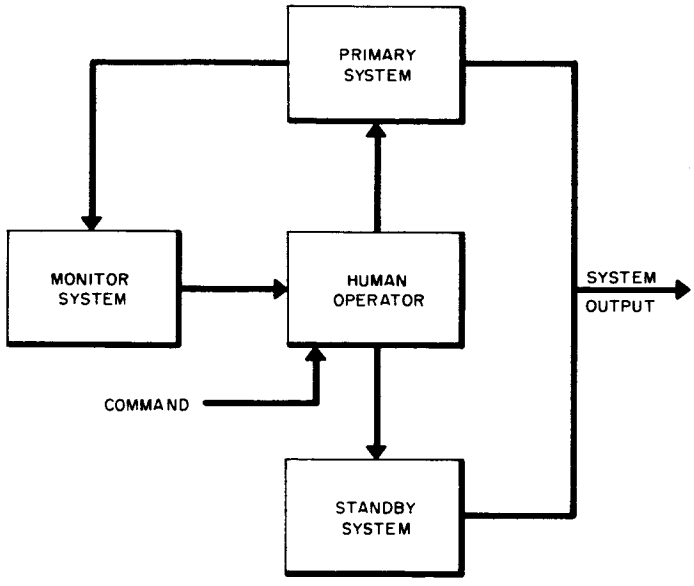


Figure 4-1. Reliability Model for System Including Human Operator

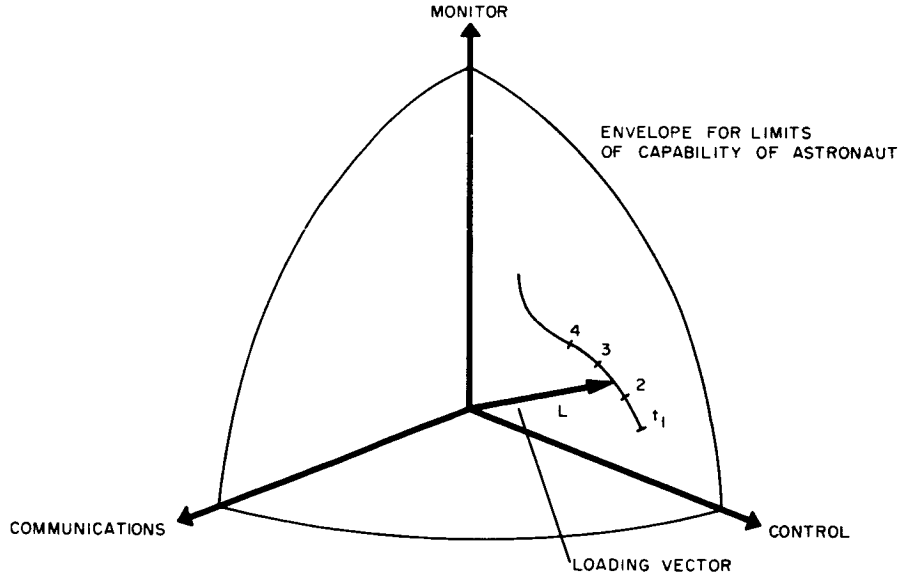


Figure 4-2. Operator Loading Analysis

- The same basic guidance elements will be used for all the manned vehicles in the Orbital Launch Operation. This will allow considerable economies in development, production, training and logistics, compared to the costs if different guidance systems were used in different vehicles.
- The OLO operational concepts are not bounded by guidance. The recommended system can accomodate any of the probable operating concepts discussed in Section II, existence or non-existence of an OLF, thrust capability of payload, etc.

5-2. RECOMMENDATIONS

The three tasks listed are the next steps in the program to achieve the recommended guidance system for the vehicles in OLO. They proceed directly from the present study, and should be investigated in parallel.

- Formulate a detailed development plan for the OLO vehicles' guidance systems. This plan should have reliability as its primary objective, and should aim for demonstrated reliability of each critical guidance element, by successful orbital flight tests, before Orbital Launch Operations begin.
- Prepare specifications and develop breadboards of the critical elements of the guidance systems; specifically, the multipurpose radar, the inertial platform and manual star tracker, and the recommended guidance system for the unmanned payloads.

- Integrate the breadboards of the critical guidance elements into an operational simulator, to optimize the man-machine relationships.

5-3. TIME SCALE

The following time scale has been selected for this study and is representative of several which have been associated with the Orbital Launch Operation. It has the following milestones:

- Operational rendezvous capability achieved by the second quarter of 1965.
- Circumlunar and lunar orbit missions during the fall of 1965 through spring of 1966.
- First manned lunar landing in the third quarter of 1966.

It is most important that the reliability of the models of the essential guidance elements listed be demonstrated by successful orbital flight tests in manned and unmanned satellites before the Orbital Launch Operations begin. When a development, manufacturing and test program which includes these orbital flights is fitted between the present time and the OLO milestones (as is done in figure 5-1) it is obvious that the three studies recommended should be started without delay. Each is discussed in more detail.

5-4. FORMULATE DETAILED DEVELOPMENT PLAN

This is a short study program which will define the individual tasks which make up the development of the guidance

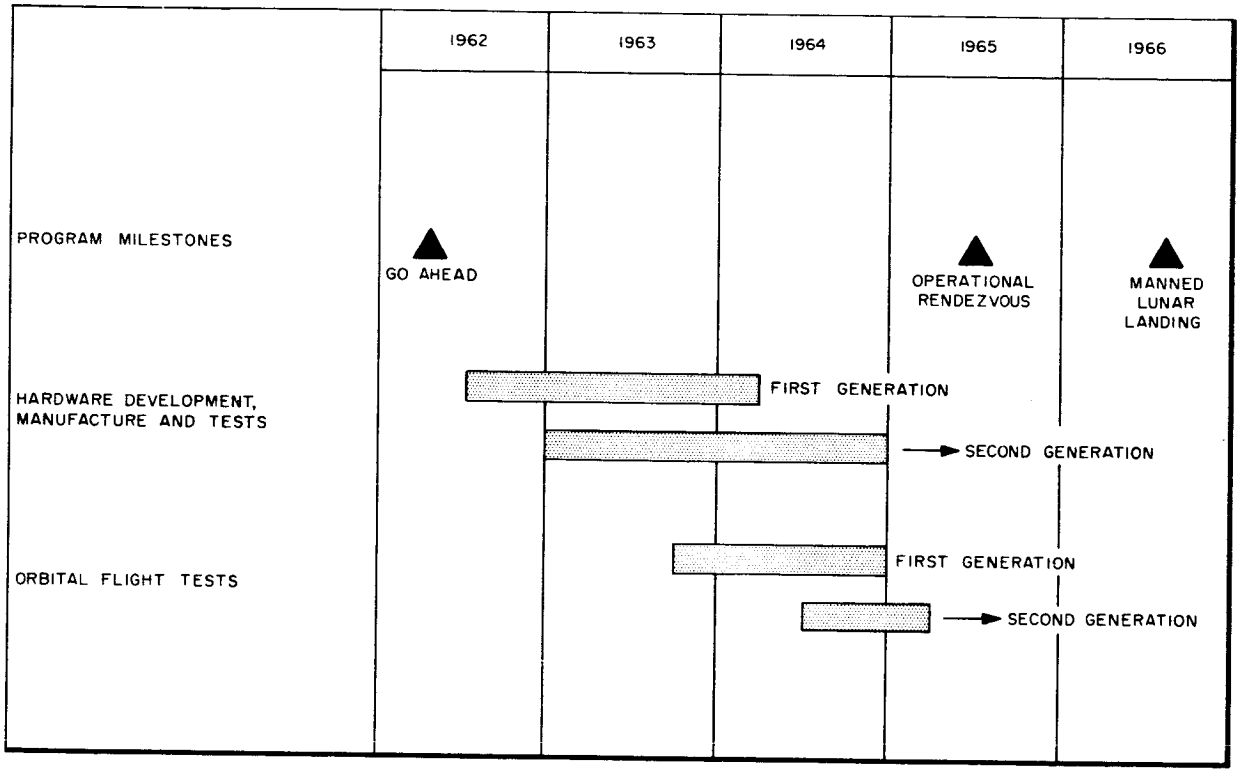


Figure 5-1. Development Program Schedule

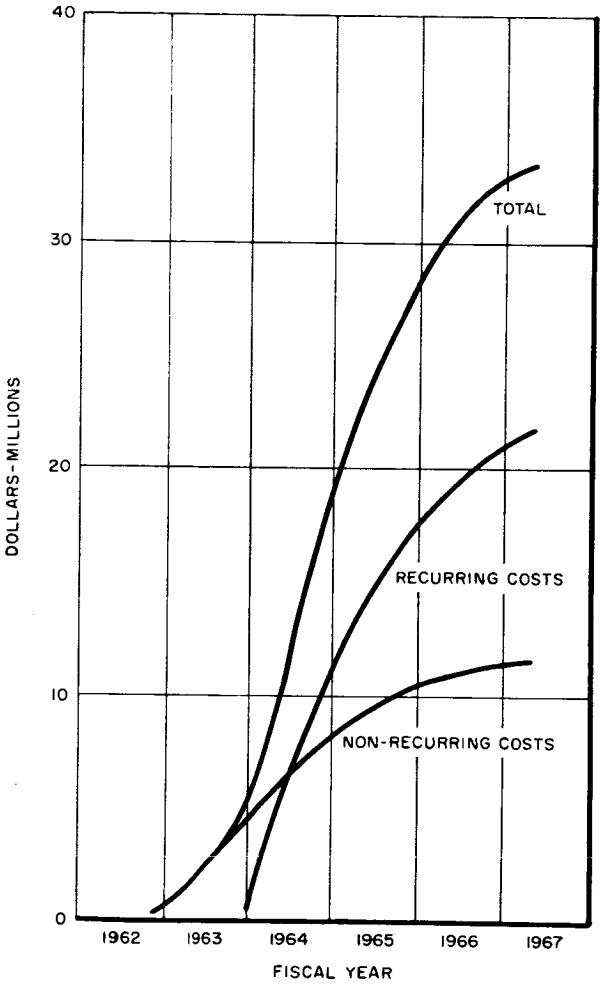


Figure 5-2. Cost Estimate

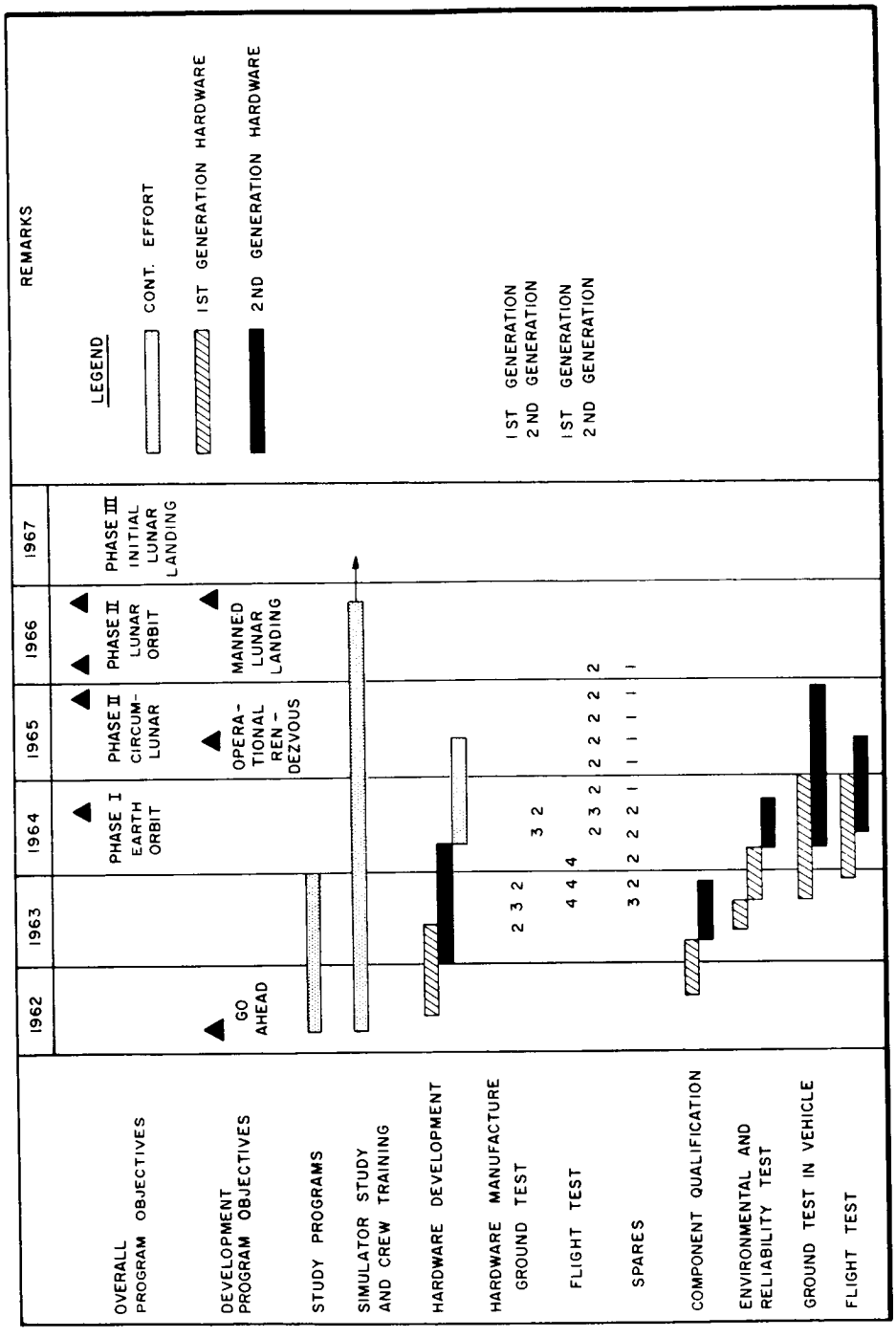


Figure 5-3. Project Schedule - Development Program for Navigation and Guidance System

system, and will integrate these tasks into an effective overall program. The study will define the objectives, the work content, and the schedule for each task. There will be three kinds of tasks, (1) the hardware development tasks for each element of the guidance system, (2) system integration tasks, and (3) tasks which will insure the development of the required amounts of the system's properties, reliability, maintainability, weight, etc.

The hardware development tasks will include the analysis, preparation of specifications, breadboards, design, allocation of models for environmental and other tests, manufacture, and testing.

The system integration tasks will include the coordination and control of information flow, coordination with vehicle design, equalization of difficulties among system elements by adjustment of specifications, system integration tests, flight tests.

The tasks to foster and control properties of the system will include, for each property, establishment of a mathematical model, specifications, single-property physical models (breadboards and mock-ups), design review, special tests and quality control, deficiency reporting and correction, and evaluation. These tasks will be planned to cover such properties as, basic function and accuracy, reliability, weight and volume, and maintainability. A PERT-type analysis will also be established to promote and assure accomplishment of the tasks and availability of the equipment on schedule. A primary purpose of this group of tasks is to insure that the resulting OLO guidance equipment has the required balance among its properties, that reliability is not neglected, for example, because of pressures of schedule, or

weight, or desire for more accuracy. The objective of these tasks is to plan a development program which will include tests and other activities necessary to produce equipment capable of successful operation in space.

5-5. SPECIFICATIONS FOR MULTIPURPOSE RADAR, MANUAL STAR TRACKER ON PLATFORM, AND UNMANNED PAYLOAD GUIDANCE

The studies reported in this volume have shown the need for these guidance system elements, based on the system requirements and consideration of a number of candidates. The studies should be continued by investigating the first tasks in the development of these systems, namely, the analysis, preparation of specifications, and breadboarding of critical parts. There are no unusual problems which would justify delaying the start, or jeopardize successful completion. This task should be started now, in order that the remainder of the development of these elements can be completed in time for models to be flight tested in orbit to demonstrate their reliability before Orbital Launch Operations commence.

5-6. INTEGRATE BREADBOARDS INTO OPERATIONAL SIMULATOR

As stated in Section IV, the hardware, sensors and computers, for OLO can be determined now by establishing candidates and selecting the best. However the "software", the guidance rule, and the allocation of tasks between the hardware and the human, and the choice of displays cannot thus be determined at this early stage. The "software" and displays require studies which are (1) closely

coordinated with the OLO operating concept and the vehicle design as these are finalized, and which (2) include tests with human astronauts, first in simulators, then in flights. The simulator studies should include the breadboards of critical guidance system elements as soon as possible.

It is necessary to commence these studies and plans for the simulation now. The studies will include establishing candidate guidance rules, types of displays, and ranges of the division of tasks among the astronauts, the sensors and the computer. The studies will enumerate specific tests to be performed to evaluate and select candidates, and at what stages additional elements of the problem should be introduced into the tests.

5-7. COST AND SCHEDULE ESTIMATES

a. Schedule

A detailed schedule of the development program is shown in figure 5-3. This program was developed with the program objectives as shown in the top line. The principal feature of this schedule is the two generation approach which is necessary in order to accomplish the system studies and development of reliabile hardware within the available time. The first generation hardware will consist primarily of the basic sensor subsystems, such as the radar and the manual star tracker on the inertial platform. This equipment will be ground and flight tested for reliability development while system and simulator studies are being conducted to determine exactly how the subsystems should be used in the operational system and in what way their information should be displayed to the astronaut. Results of

the system studies would then appear in the second generation equipment which would be designed as a complete system for application in the final manned Spacecraft. Both first and second generation hardware can be tested in unmanned and manual orbital flights. One of the obvious vehicles in which to test this hardware is in the Gemini vehicle.

The quantities of hardware required are the approximate numbers of systems required for both ground and flight test. The hardware designated for ground test will be used for the qualification and environmental test program. The environmental test program should include a test-to-failure-program in order to insure maximum reliability.

b. Cost Estimate

The cost estimate shown in figure 5-2 was based on the program shown in figure 5-3, therefore, it covers development costs through the first manned lunar landing. The total cost of the program is estimated to be 34 million dollars, divided as follows:

(1) Non-recurring Cost

It is estimated that the total non-recurring costs will be 12 million dollars and will cover the following activities.

- Program management and liaison
- System studies including simulator studies
- Hardware design and development, including special ground support equipment.

- Environmental test program
- Engineering support of flight operations
- Manufacturing starting costs.

(2) Recurring Costs

Total recurring costs are estimated to be 22 million dollars and will cover the following activities.

- Manufacturing costs of all hardware including spares. Quantities of systems required as shown in figure 5-3.
- Manufacturing cost of ground support equipment.

The costs of launch booster, etc., are not included in the estimate.

APPENDIX A

GUIDANCE ACCURACY REQUIREMENTS

A-1. PHILOSOPHY

The subject of guidance accuracy for space system applications has been investigated and well documented in much of the current space guidance literature. Possibly several hundred man-years of effort have been focused in this area over the last few years to determine whether the various space missions are feasible from an accuracy point of view by comparing the accuracies required to the component state of the guidance art. The consensus of opinion, just recently crystallized, is that, though there are a few critical mission phases, all of the mission phases present no insurmountable design problems. As a consequence, much of the national effort is beginning to steer towards the more encompassing task of investigating and selecting:

1. Mission operational concepts that will best accomplish the mission (such as discussed in Section II
2. The guidance components that possess a high degree of reliability and versatility which can be used during several of the mission phases (Section III)
3. The principles that dictate the optimum utilization of both the astronaut and equipment (Section IV).

For these reasons, the accuracy requirements presented in this appendix have been derived based on relatively simple models for both the earth orbit and lunar phases of

the mission, for example, near circular orbits and an explicit two-body simulation for earth-moon space. The accuracy values derived with these models are within 10 percent of the more sophisticated values and suffice for the purpose of selecting guidance systems.

A-2. GENERAL

This appendix presents the accuracies for each of the guidance functions that are required to fulfill the Orbital Launch Operation. The guidance functions which are investigated here consist of determining:

- Vehicle orbit
- Thrust velocity increment
- Vehicle attitude
- Rendezvous guidance parameters.

The accuracy requirements for each of the vehicles are summarized in table A-1 and have been established mainly on the basis of fuel economy. In the orbital operations, the level of 20 ft/sec is considered allowable fuel wastage due to a particular functional inaccuracy. In addition to fuel economy, human safety, flexibility of operation and the frequency of engine turn-ons are also considered. The ground rule for alignment for trans-linear injection is that the error in periluna upon arrival should not exceed 70 miles.

A-3. OLF REQUIREMENTS

a. Orbit Determination

The most stringent accuracy requirements for OLF Orbit determination are obtained when it is required to maintain a synchronous orbit, if this should be deemed necessary. The OLF may be put into an orbit which is synchronous with the earth so that there can be one launching per day, from Cape Canaveral, into a direct ascent to compatible rendezvous in which almost the entire ascent takes place in the orbital plane. This results in the optimum combination of minimum fuel and time. While the synchronism would include the fact that the nodal longitude is shifting due to earth's oblateness, there are other smaller perturbations to the orbit which cannot be completely accounted for and would result in orbit drift. The process of updating the OLF orbit, known as station-keeping, requires an orbit determination function since the actual orbit must be known before the desired corrections can be computed. The allowable orbit position error is taken to be 3.0 nm RMS. This is based on the conservative case where, if the entire error was due to a rotation of the orbit plane, the error would amount to 3/3740 = 0.8 milliradians, for a 300 nm orbit. This error would in turn cause a velocity expenditure of 0.8 x 24.52 = 19.7 ft/sec for a direct ascent of the payload.

b. Thrust Velocity Increment

The ΔV accuracy requirement is based upon coplanar transfers to correct vehicle position and upon minimizing the number of times the vernier thrust motors must be

turned on. It can be shown that for a 300 nm orbit, there is an error of approximately 3 nm of forward position error per revolution for each foot per second of forward velocity error. A fairly large transfer corresponds to a Δ V of about 30 ft/sec. If there are 15 orbits per day and a buildup of only 0.3 nm/day is allowed in order to keep rocket engine turn-ons to a minimum, the acceptable error in the 30 ft/sec becomes

$$\left(\frac{0.3 \text{ nm}}{15 \text{ day}}\right)\left(\frac{1 \text{ ft/sec}}{3 \text{ nm}}\right) = 0.0067 \text{ ft/sec}$$

In practice, this requirement is equivalent to a correction of about 0.0067 ft/sec per day or about 0.2 ft/second per month which is quite small. It should be noted here that if rocket engines become very reliable so that they can tolerate more frequent starts, the velocity change tolerance can be raised.

c. Attitude Determination

The attitude requirements for station keeping are considered loose for purposes of thrusting since the corrective thrust levels are quite low. However, more accurate attitude is necessary for any orbit determination of the payload. If the payload error, with respect to the OLF, is allowed to be the same 3.0 nm RMS as the OLF orbit error and is allotted equally to attitude and range, the total error in each is about 3.0 nm/ $\sqrt{2}$ = 2.12 nm which corresponds to an angle of about 24 minutes of arc at ranges of 300 nm. Assuming that the payload is being watched for at least two periods of revolution, then the period and semi-major axis of the ellipse can be determined quite accurately. If the axes are assumed to be in error by

1.0 nm, then the velocity at apogee is incorrect by about 6.5 ft/sec. Allowing at least two maneuvers wherein 6.5 ft/sec of Δ V is wasted plus some smaller corrections, the total Δ V wasted is still less than 20 ft/sec.

Note that the attitude requirement for orbit determination is not included in table A-1 since it is a function of the method or orbit determination.

d. Rendezvous Sensing

The sensing requirements can be categorized by two ranges of OLF-payload separation. These are 300 nm to 1000 feet for terminal rendezvous and below 1000 feet for docking.

Assuming that the OLF-payload separation is 300 nm, the angular accuracy of the line-of-sight from the OLF to the payload can be specified as 24 minutes of arc, based on the discussion in paragraph b. of this listing, as can the range information accuracy of 2 nm. As the separation is reduced to 1000 feet, the percentage range error of 0.67 percent can be applied down to a minimum error of 50 feet. The angular accuracy of 24 minutes can be retained down to 1000 feet since, if this were larger, the terminal rendezvous thrusting program may become too inefficient.

For ranges below 1000 feet, it appears that 1 degree relative attitude between the OLF and payload and 5 percent range error is sufficient for safe docking.

A-4. PAYLOAD REQUIREMENTS

a. Orbit Determination

As is pointed out in Section II, there is no requirement for the payload to determine its own orbit. This function is performed by either the ground or the OLF.

b. Thrust Velocity Increment

The velocity measurement accuracy of the payload can be specified by considering the phasing ellipse maneuver that is controlled by the OLF. Assuming that both vehicles are in identical near-circular orbits and that a 300 nm separation exists between the vehicles, a total ΔV of about 250 ft/sec is required to effect rendezvous in one revolution. An error of 1 ft/sec or 0.4 percent in thrust velocity for this case yields about 3 nm error in position at the end of the transfer. This accuracy is reasonable in light of the closed-loop operation of the terminal guidance maneuver. As the relative distance between the two vehicles is reduced to below one nm, the incremental velocity accuracy required for close rendezvous safety is specified to be 0.1 ft/sec.

c. Attitude Determination

The payload attitude requirements are loose since it is assumed that the OLO operation starts only at the orbital transfer point, or if there is no such transfer, then it starts with the terminal rendezvous. If it is assumed that the transfers occur in multiples of 180 degrees of rotation in order to maximize fuel economy, then

the ΔV requirements are extremely insensitive to attitude errors and in fact, vary approximately with the square of the error so that $\partial \Delta V / \partial$ pitch or $\partial \Delta V / \partial$ azimuth equal zero for small errors in pitch and azimuth. Under these conditions, an error of even 5 degrees has negligible error in azimuth. In pitch however, 5 degrees can cause more than the 20 ft/sec allowed fuel wastage only if it is assumed that the pitch error, in a transfer from a 100 nm circular orbit to a 300 nm circular orbit, has resulted in an altitude difference which must be corrected in a time period of less than 5 minutes instead of waiting another one-half revolution. Therefore, the angular accuracy is determined primarily by the guidance of the payload during the terminal maneuver when it is being controlled by the OLF, at which time the operator wants the payload to follow his directions accurately. Based on some preliminary analog computer rendezvous runs, it appears that a 5 degree pitch limitation is a satisfactory limit. Using this as a 3σ limit and rounding off conservatively, the angular accuracy tolerance becomes 1.5 degrees RMS. Note that this final requirement is caused by fuel considerations since the rendezvous could still be completed with a higher tolerance in closed-loop operation, but at a loss in efficiency and time.

d. Rendezvous Sensing

The payload is not required to sense range and line-of-sight to target, since the rendezvous sensing capability has been allocated to the OLF.

The primary function of the payload guidance system is to accept and execute rendezvous thrusting commands from the OLF.

A-5. TUG REQUIREMENTS

There are three primary mission requirements for the tug that must be investigated. These are:

- Safe reentry
- Return to within 15 nm of a specified landing site
- Rendezvous with other OLO vehicles.

a. Safe Re-entry Considerations

Although the reentry problem presents the greatest hazard to astronaut safety, it does not pose the most stringent accuracy requirements except for attitude measurement in the reentry corridor. Since the reentry corridor may vary from 8 to 12 miles in height, a 2 nm altitude determination error is considered adequate for safety purposes. Assuming the reentry flight path angle (γ) is less than -1 degree, skipping of the tug is minimized, as are the temperatures that may be encountered.

The accuracy requirements for the descent ellipse from a 300 nm, near circular, orbit to establish a nominal flight path angle of -1 degree at atmospheric reentry, 300,000 ft with an accuracy of 0.1 degree are not stringent. Since the percentage error in either attitude or velocity at deboost is given by 100 ton $\gamma \Delta \gamma$, the allowable errors for a 500 ft/sec deboost velocity, become 0.75 ft/sec for ΔV and/or 350 ft for altitude. These then are the orbit determination requirements based on the safe reentry considerations.

b. Specific Landing Site

The allowable position error at time of landing, neglecting the use of ground-based navigational aids, is taken to be 15 nm RMS.

This has been broken up into several major components. The first is six nm rms in final position which is six minutes in ground angle altitude. The second is 8 nm rms due to initial altitude which causes the deboost altitude requirement to be about 75 feet RMS. Also, there is a velocity accuracy of about 0.15 feet/sec RMS which causes 8 nm of error. The thrust error of 10 nm is broken up into 8 nm for an acceleration of 0.03 percent and 6 nm for a pitch error of 10 minutes. These sensitivities are based upon a 10,000 nm ground range. Azimuth error, under these conditions, is negligible since the ground range angle is close to 180 degrees.

This technique is slightly pessimistic because orbital data can be fed to the computer between the time of deboost and reentry into the atmosphere. Any such input will reduce the effect of thrusting errors at the higher altitude by providing feedback of the position error buildup after thrust.

As can be seen, the accuracies obtained for the landing site requirement are more stringent than those for safe reentry. These are then the pacing requirements listed in table A-1.

c. Rendezvous Considerations

For the purposes of this study, the accuracy requirements for Tug rendezvous sensing are taken to be identical to those of the OLF.

A-6. SPACECRAFT REQUIREMENTS

For the purposes of this study, 3 day lunar trajectories, with perilunes varying from 200 nm to 1000 nm, were investigated on the digital computer to determine the various error sensitivities on which to base Spacecraft injection requirements. A 70 nm error in perilune was assumed to be a 3σ maximum. Dividing this by 3 to obtain the rms value and then allotting equal errors to altitude, position altitude and velocity, the allowable error becomes roughly 10 nm for each source of error.

To determine the allowable launch altitude error, a sensitivity of 800 nm of perilune radius to 1 nm orbital altitude error was assumed which yields $\frac{10}{800}$ nm or 75 feet of error.

The injection of the Spacecraft into a lunar orbit allows errors of 0.2 ft/sec RMS in forward velocity and 1.5 ft/sec RMS in vertical velocity. Again, both of these figures will be bettered in actual practice if the horizontal position specification is met.

To determine launch position accuracies, a sensitivity coefficient of 31 nm of perilune radius to 1 nm of orbital position error was selected. This yields an allowable position error of 10/31 nm or about 2000 feet.

It should be noted that, in practice, the horizontal position specification requires that the altitude be determined to a greater accuracy than 75 feet because of the dependence of orbital velocity upon altitude.

To determine the allowable attitude error while establishing the 10,600 ft/sec launch velocity increment,

to be added to the 24,500 ft/sec orbital velocity, a sensitivity of 46 nm in perilune radius to 1 min of pitch error was assumed, yielding an allowable error (10/46) $\frac{35.1}{10.6}$ minutes or roughly 40 seconds.

To determine the allowable launch velocity error, a sensitivity of 32 nm per ft/sec error was selected, yielding an error of 10/32 ft/sec or 0.3 ft/sec. Since there are two sources of velocity error, namely:

1. Due to orbital determination errors
2. Due to thrust measurement errors, $3/\sqrt{2}$ or 0.2 ft/sec velocity error was allotted equally to both sources.

The OLO rendezvous sensing requirement for the Spacecraft are taken to be identical to those of the OLF, for purposes of this study.

TABLE A-1

GUIDANCE ACCURACY REQUIREMENTS

Function	Vehicle OLF	Payload	Tug	Spacecraft
Orbit Determination	3 nm RMS for station keeping.	N/A	6 nm in position 75 feet in altitude 0.16 ft/sec in orbit velocity.	Initial orbit determination by ground; 75 ft. RMS in altitude and 2100 ft. RMS in horizontal position for line-up of spacecraft, 1.5 ft/sec vertical velocity and 1.2 ft/sec horizontal velocity.
ΔV	0.0067 ft/sec RMS error for up to 30 ft/sec (ΔV).	0.4% RMS or 0.1 ft/sec RMS whichever is larger.	0.03% RMS for reentry.	.002% RMS or 0.2 ft/sec whichever is larger.
Attitude	24 minutes RMS for line-of-sight (See following).	1.5° RMS	10 min. RMS for reentry.	40 seconds RMS.

TABLE A-1 (Cont)

GUIDANCE ACCURACY REQUIREMENTS

Function	Vehicle OLF	Payload	Tug	Spacecraft
Range-to-Target	Approx. 2 nm RMS for orbit determination to 300 nm range; 0.67% or 50 ft. whichever is larger for ranges above 1000 ft. 5% for ranges less than 1000 feet.	N/A	(Same as OLF)	N/A
Line-of-Sight Angle-to-Target	Approx. 24 min. RMS for orbit determination at 300 nm; approx 1.0° for ranges below 1000 feet.	N/A	(same as OLF)	N/A

APPENDIX B

RELIABILITY REQUIREMENTS AND CONSIDERATIONS

B-1. GENERAL

Reliability is foreseen as the most important guidance problem and perhaps the most important single problem in the orbital launch operation. This conclusion has unfortunately been supported all too frequently a short distance above the launch pad at Cape Canaveral. The large manpower and financial cost to be expended in the lunar mission together with the high moral responsibility of insuring human safety make it absolutely mandatory that this program be entered into history as a successful and fruitful accomplishment in the United State's program of space exploration and conquest.

The accuracy requirements set forth in Appendix A are generally within the state-of-the-art. Reliability, however, must ultimately be demonstrated in the true operational environment for each element of the guidance system and for the ensemble of elements (the system) before that element is entrusted with human lives or valuable payloads in the orbital launch operation. Since orbital launch operations are scheduled to begin as early as 1965, the basic guidance elements must be selected from the present state-of-the-art to allow sufficient demonstration of their reliability.

One of the problems in this area is that reliability analysis or estimates must be made on a meaningful

basis. For example, it is quite possible to get an estimate that a particular type of guidance element will have a mean time between failure (MTBF) of several thousand hours by 1965, when the best performance in use to date is a MTBF of several hundred. Obviously, any estimate that the reliability of a type of device can be improved by a factor of ten in three or four years must be carefully considered and objectively weighed. An optimistic estimate of improvement of this order may be caused in part by different definitions, some of which exclude certain failure types. However, the operational requirements of the lunar missions demand freedom from all failures and therefore the estimated level of reliability must be demonstrated.

The selection of the winner among candidate guidance systems requires only estimates of relative reliability, for example comparison between a horizon tracker and a gyro, or between a radar and a star tracker. The selection process would be simpler if it could be freed from the individual estimatesof absolute reliability, which have varying degrees of optimism. Therefore the present analysis separates the question of reliability into the following parts and considers each in turn:

- Determine the numerical value of reliability which will be required, and which must be demonstrated for each vehicle guidance system in the Orbital Launch Operation.
- Independent of estimates of absolute reliability, establish a relative basis for comparing the probable reliabilities of candidate guidance systems, in order to evaluate them and make a selection.

- After the relative reliability estimates have been made for the major kinds of guidance elements, these estimates are placed in an absolute reference frame, so that the reliability of the selected system for each vehicle can be estimated, and the amount of redundancy which is required can be determined. This amounts to a consideration of an averaged best estimate of the probable level of reliability of guidance elements in general which can be attained by 1965-6. It can be adjusted later to obtain greater optimism or conservatism, however its value has not influenced the selection among the candidate guidance systems.

B-2. RELIABILITY REQUIREMENTS

As a reliability goal, it is assumed that each vehicle guidance system involved in the Orbital Launch Operation must be 99 percent reliable during its critical period. That is, there should be no more than 1 chance in 100 that the vehicle's guidance system would cause an interruption or degradation in the particular vehicle's mission. The critical period is considered to be that period of operation when maintenance of the guidance system cannot or should not be performed.

To accomplish this, when a system is selected from among the candidates, as described in Appendix C, two levels of redundancy are added:

- Sufficient redundancy is added to the elements of the primary mode guidance system to give 99 percent probability of operation with full capability throughout the critical period.

Additional redundancy, usually manually operated, and usually with reduced capability, is added in order to minimize the possibility of an abort in the approximately one percent of the missions in which failure of the full capability might be expected.

In order to determine the critical periods for each vehicle's guidance system an estimate was made of system operating times for each vehicle in both the NO OLF concept and the OLF concept. The breakdown of these estimates is shown in tables B-1 through table B-3. In the OLF concept two modes of operation were considered in order to bound the range of possible operating times. The first mode assumes no tug operation, that is the payloads are capable of being commanded to the docking point by the OLF. In the second mode, it has been assumed that all these payloads (spacecraft, OLV #1, OLV #2) and the crew ferry vehicle are launched to within 300 nm range of the OLF but are then incapable of completing the rendezvous and therefore require maximum tug operation. These estimates are summarized for the orbital launch operation only (earth launch to orbital launch).

System Operating Time - No OLF Concept

Spacecraft and OLV Payload	25 hours
Crew Ferry Vehicle	20 hours
Fuel Payload	5 hours

TABLE B-1

GUIDANCE SYSTEM OPERATING TIME - NO OLF CONCEPT
(EARTH LAUNCH TO ORBITAL LAUNCH)

Mission Phase	Spacecraft & OLV Payload	Fuel Payload	Crew Ferry
Ascent(1)	4 (hrs)	4 (hrs)	4 (hrs)
Remote Confidence Checks	1	1/2	-
Spacecraft-Ferry Rendezvous	2	-	2
Spacecraft Checkout by Crew	2	-	2
Spacecraft-Fuel P/C Rendezvous(2)	3	-	3
Fuel Transfer	1	-	1
Prelaunch Checkout	3	-	2
Countdown	3	-	-
Abort Re-entry	1	-	1
Contingencies (repair, launch hold, etc.)	5	1/2	5
Total (hours)	25	5	20

Notes: (1) includes boost, coast and a 2 rev phase change maneuver

(2) includes a 1 rev phase change maneuver

TABLE B-2

GUIDANCE SYSTEM OPERATING TIME - OLF CONCEPT WITH NO TUG OPERATION
(EARTH LAUNCH TO ORBITAL LAUNCH)

Mission Phase	OLF	Tug	Spacecraft Payload	OLV Payload No. 1	OLV Payload No. 2	Crew Ferry
Ascent(1)	NOTE (2)	NOTE (2)	4 (hrs)	4 (hrs)	4 (hrs)	4 (hrs)
Spacecraft	2 (hrs)	1 hr.	3	-	-	-
Terminal Rendezvous and initial checkout		(standby)				
OLV No. 1 Terminal Rendezvous	2	1 (standby)	-	2	-	-
OLV No. 2 Terminal Rendezvous	2	1 (standby)	-	-	2	-
Crew Ferry Terminal Rendezvous	2	1 (standby)	-	-	-	2
Monitor Payloads and Ferry Ascent	12	-	-	-	-	-
Prelaunch Checkout	2	-	3	-	-	-
Remote Spacecraft to launch area	1	1	1	-	-	-
Count down	3	-	3	-	-	-
Abort Reentry	-	1	1	-	-	1
Total (hours)	26	6	15	6	6	7

NOTES: (1) includes boost, coast and a 2 rev phase change maneuver

(2) OLF and tug placed in orbit about 5 months prior to the orbital launch operation.

TABLE B-3

GUIDANCE SYSTEM OPERATING TIME

OLF CONCEPT WITH MAXIMUM TUG OPERATION

(EARTH LAUNCH TO ORBITAL LAUNCH)

Mission Phase	OLF	Tug	Spacecraft Payload	OLV Payload #1	OLV Payload #2	Crew Ferry
Ascent (1)	NOTE (2)	NOTE (2)	3 HRS.	3 HRS.	3 HRS.	3 HRS.
Spacecraft Rendezvous and initial checkout	7 HRS.	7 HRS.	-	-	-	-
OLV #1 Rendezvous	7	7	-	-	-	-
OLV #2 Rendezvous	7	7	-	-	-	-
Crew Ferry Rendezvous	7	7	-	-	-	-
Prelaunch checkout	2	-	3	-	-	-
Remote Spacecraft to launch area	1	1	1	-	-	-
Countdown	3	-	3	-	-	-
Abort Re-entry	-	1	1			1
Total hours	34	30	11	3	3	4

Notes: (1) Assumes boost coast and 1 rev phase change maneuver to arrive within 300 mile range of OLF.

(2) OLF and tug placed in orbit about 5 months prior to the orbitol launch operations.

System Operating Time - OLF Concept

	No tug Operation	Max tug operation
OLF	26 hours	34 hours
Tug	6 hours	30 hours
Spacecraft Payload	15 hours	11 hours
OLV #1 Payload	6 hours	3 hours
OLV #2 Payload	6 hours	3 hours
Crew Ferry Vehicle	7 hours	4 hours

Obviously, the two estimates for the OLF concept are extreme limits for several reasons. No tug operation implies that all payloads have vernier thrust capability to complete the terminal rendezvous and that all equipment within each vehicle operate trouble free. Although this thrust capability has been assumed for this study, as discussed in Section II Operational Concepts, it is recognized that gross thrust and even no thrust capability are possible. Furthermore, it is considered unrealistic to presume that all equipments will either operate perfectly or all ascending vehicles will malfunction and must therefore be assisted in completing the rendezvous from 300 nm. More reasonable operating times are therefore considered to fall within these boundaries and are

OLF	30 hours
Tug	20 hours
Spacecraft Payload	12 hours
OLV #1 Payload	5 hours
OLV #2 Payload	5 hours
Crew Ferry Vehicle	5 hours

In arriving at system critical periods (that period of operating time when maintenance of the guidance system cannot or should not be performed) the previously stated operating times must be examined to determine if repairs could be made sometime within this period without seriously affecting any phase of the orbital launch operation. Repair capability must be considered to prevent undue burden being placed on the MTBF requirement of each guidance system. It is this capability that emphasizes the importance of the OLF. In NO-OLF operation, the crew ferry vehicle offers an economical but extremely limited repair facility. The amount of spare parts and test equipment that could be carried into orbit might be sufficient to repair some malfunction within the spacecraft, however, performing maintenance outside the command module does not appear to be practical at this time. In the event a major repair or critical part was required, a maintenance vehicle could be launched into orbit at the expense of time, cost and increasing the complexity of the orbital launch operations. An important consideration in this area is the trade off between system reliability and the amount of maintenance that can be performed in space. For this study it has been assumed that essentially no maintenance can be performed and therefore, the critical periods for the guidance systems in the NO-OLF concept are considered to be the same as the previously stated operating times. When these operating times are combined with the required reliability, the required MTBF for each vehicle guidance system is,

Required MTBF - NO OLF Concept

	Critical Period	MTBF
Spacecraft & OLV Payload	25 hours	2500 hours
Crew Ferry Vehicle	20 hours	2000 hours
Fuel Payload	5 hours	500 hours

Using the spare parts and repair facility offered by the OLF concept the MTBF requirement on the spacecraft and the crew ferry can be significantly relieved. In examining the operating times for this concept it can be reasoned that the OLF, tug and spacecraft do not necessarily have to experience trouble free operation during these total times since repairs could be made, for example between rendezvous, without interfering with orbital operations. Therefore the critical periods for these vehicles have been taken at about 70 - 80 percent of the total operating time. The critical periods for the crew ferry and OLV payloads, however, are the same as the operating times since these represent the time for a rendezvous operation during which repairs cannot be made. Combining these critical periods with the required reliability, the MTBF's for the OLF concept are,

Required MTBF - OLF Concept

	Critical Period	MTBF
OLF	20 hrs.	2000 hrs.
Tug	15 hrs.	1500 hrs.
Spacecraft Payload	10 hrs.	1000 hrs.
OLV #1 Payload	5 hrs.	500 hrs.
OLV #2 Payload	5 hrs.	500 hrs.
Crew Ferry Vehicle	5 hrs.	500 hrs.

The guidance system operating times and critical periods shown for the NO OLF and OLF operational concepts have been estimated for the orbital launch operation, that is the three to four day period from earth launch of the first payload to orbital launch of the lunar space vehicle. It is quite probable that the spacecraft will contain a single, integrated guidance system capable of performing the functions required in both the orbital launch operation and the lunar mission. The guidance system for this vehicle must therefore operate reliably for a period of time considerably longer than that estimated for the orbital launch operation alone. Following the OLO, the spacecraft will most likely embark upon at least a 7 day lunar journey; 3 days out bound, about 1 day on the moon and 3 days inbound. An estimate of system operating times during this period was made to ascertain what impact the total mission would make on the reliability of each guidance element and the degree of redundancy within this spacecraft's guidance system.

Spacecraft Guidance System Operating Time
During Lunar Journey

Earth departure	2 hours
Midcourse *	5 hours
Lunar approach and landing	5 hours
Lunar ascent and departure	3 hours
Midcourse *	5 hours
Earth approach, reentry and landing	5 hours
Total	25 hours

- - - - - - -

* Assumes two velocity corrections - most of system shut down during midcourse except during velocity corrections - computer time may vary from seven hours to full time depending on how manually derived celestial fix data is processed.

Combining this estimate with the critical periods estimated for the spacecraft in the NO OLF and OLF concepts yields,

	Critical Period - Total Mission
NO OLF	50 hours
OLF	35 hours
Value used in reliability analysis of spacecraft guidance.	40 hours

B-3. RELATIVE RELIABILITIES OF GUIDANCE ELEMENTS

Relative reliability ratings are presented in this subsection for most of the elements in the candidate systems in order to form an objective and realistic basis for selecting the more promising system concepts. The relative reliability approach has been adopted in order to prevent unduly optimistic estimates of MTBF's from influencing the selection of the winning candidate. Determining the amount of redundancy in the chosen system requires an estimate of absolute reliability.

The major difference between candidate systems, from which the recommended system for each vehicle is chosen, is in the kinds of sensors used in the different candidates. A comparison is made of the complexity (an inverse measure of reliability) of several kinds of guidance sensors; accelerometers, displays and the digital computer have not been included since these elements are approximately equivalent for all candidates. The sensors have been grouped into three families - optical, inertial and radar. Although the gyro is in general common to all candidates because different candidate systems contain different numbers of gyros, it has

been included to arbitrarily serve as a baseline and has been assigned a complexity of unity.

The relative reliability among the candidate guidance systems, and the absolute reliability of the selected systems were determined in the following steps:

Step 1. Families of candidate sensors were established. Each family contains the units which are considered as candidate guidance elements for OLO, plus at least one similar unit presently in production for which ample reliability experience data are available. The families are:

a. Radars
b. Optical
c. Inertial

Step 2. The members within each family were ranked in order of estimated relative reliability, based upon a careful study of their complexity. The studies were made by design specialists in each sensor. Complexity diagrams used in this study are shown in figures B-1 and B-2 for the radar and tracker families. The members and rankings are,

Radars	Rank of Complexity
APN 59 radar	18
OLO rendezvous radar (altitude, vertical, angle and range tracking capability)	6
OLO navigation radar (includes angle and range tracking functions)	8
OLO spaceborne beacon	3
Trackers	
Hustler B-58 Automatic Star Tracker (includes own platform, sealed housing, etc.)	15
OLO candidate typical automatic star tracker	10

	Rank of Complexity
OLO candidate typical horizon tracker	4
OLO candidate typical automatic sun tracker	2
OLO candidate manual star tracker	1 +

(Note - all OLO candidate star and sun trackers are to be mounted on an intertial platform which is not included in this estimate of complexity).

Inertial	
Sperry ROTORACE® Gyros	1
OLO candidate typical gyro	1
B-58 Navigation Platform Electronics	6
OLO candidate typical platform electronics	2

Step 3. Reliability experience data from field usage or production tests, from Sperry programs such as the B-58 navigation system, is used to relate the relative reliabilities of the members of each family which are presently in production; as follows:

APN 59 radar	18
Hustler Automatic Star Tracker	15
ROTORACE Gyro	1
Hustler Platform Electronics	6

Step 4. From this information a composite ranking list was prepared giving the relative reliabilities of each kind of sensor in the candidate guidance systems considered for OLO. Table B-4 shows this composite ranking structure as well as a summary of the relative ranking of the sensors for each of the listed steps. It is to be noted that in the case of the

TABLE B-4

RANKING OF SENSOR COMPLEXITY - A MEASURE OF RELATIVE RELIABILITY

OLO Sensors	Similar Equipments - 1961 Test Data	Predicted for OLO Equipments	Predicted for Families of OLO Equipments	Composite for all OLO Sensors
Tracker Family				
Star tracker (automatic)	15 (B-58)	10	10	10
Horizon Seeker (automatic)			4	4
Sun tracker (automatic)			2	2
Star tracker (manual)			1+	1+
Passive Sun seeker (stop tumbling) (Solar reference one)			1-	1-
Stadiametric Optics			7	7
			1	1
Inertial Family				
Gyro (one)	1 (Rotorace)	1		1
Platform and electronics (excluding gyros)	6 (B-58)	2		2
Radar Family				
Rendezvous radar (target tracking, altimeter and vertical determination capability)	18 (APN-59)	6	6	4
Position Fixing radar			8	5+
Payload beacon			3	2

Notes:

(1) Based on the relative complexity of equipment similar to that used as candidates in OLO

(2) Larger numbers denote a more complex sensor; ranking arbitrarily normalized to the gyro

(3) Includes about 40% upgrading resulting from availability of more reliable parts and operation at lower power levels.

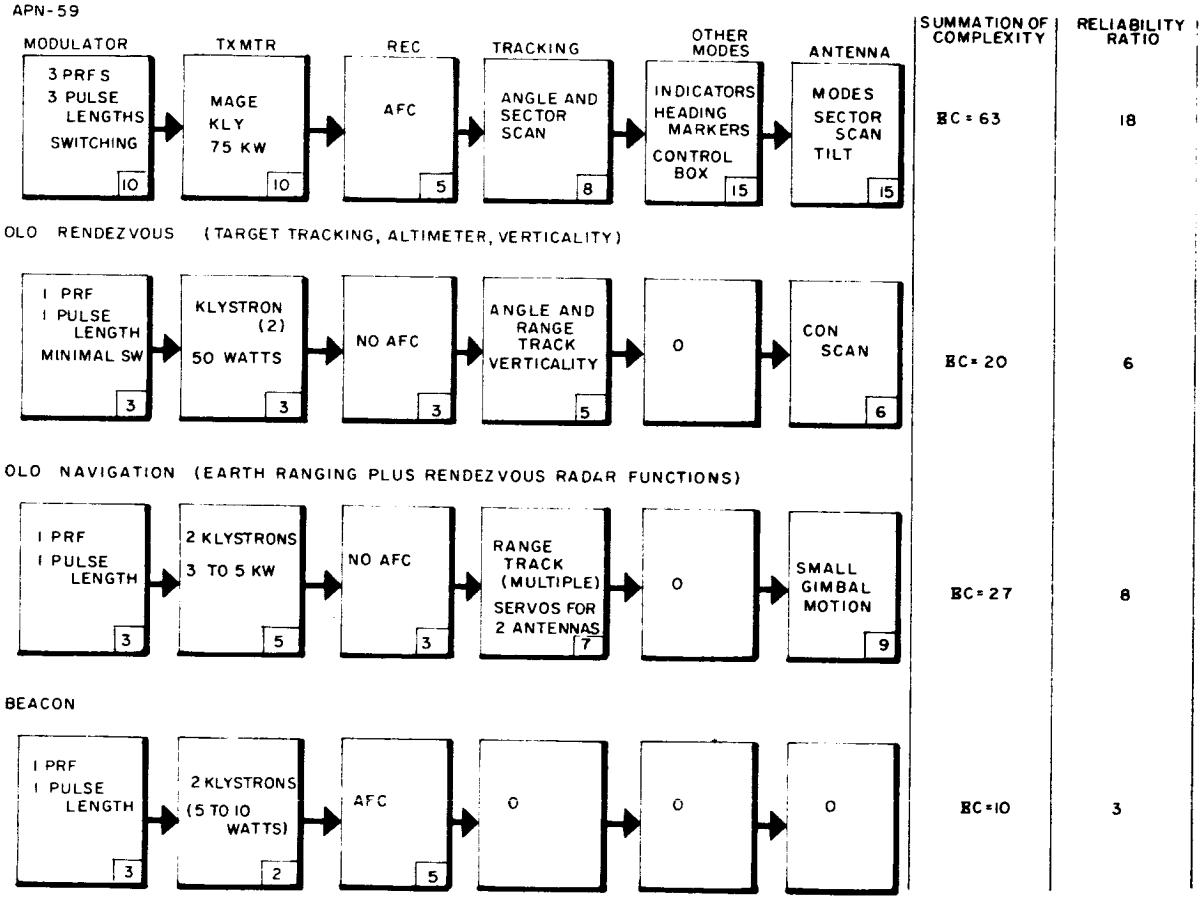


Figure B-1. Complexity Diagrams of the OLO Radar Family

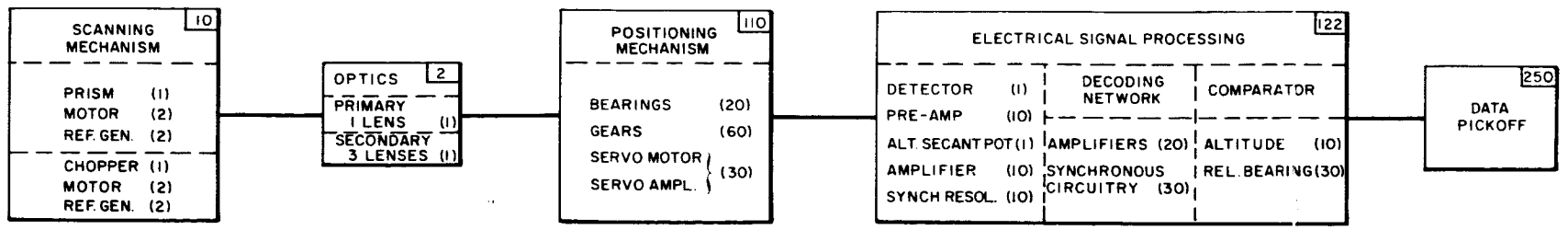
radars the estimated relative reliability of each OLO candidate was increased by about 40 percent above that obtained on the basis of relative complexity alone because - (1) a number of components can be purchased and included in a radar designed today which have higher reliability than their counterparts in the APN 59, which was designed a few years ago, and (2) the OLO candidates operate at much lower power levels than the APN 59.

B-4. ESTIMATE OF ABSOLUTE RELIABILITY

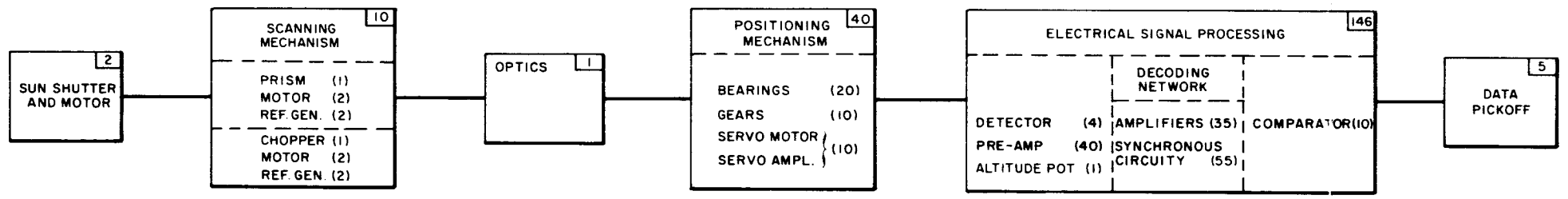
The estimates of absolute reliability are essentially reiterations of the last two steps of the relative estimates, that is, the measured reliability of the existing member of each family is extended to an estimate of the reliability of each OLO candidate, based on relative complexity. As an example this is described in detail for the multipurpose altitude - vertical - angle and range tracking radar. The mean time to failure of this radar, (described in Appendix D Subsection D-2) if designed now, is estimated to be 500 hours. This estimate is conservative, and was obtained in the following manner. The radar described here has been compared in detail to the APN 59, an airborne radar which has been in quantity production and Air Force use for some time. Engineers responsible for the design of the APN 59 made an estimate of its MTBF, using parts failure data from three sources: (1) Sperry Hustler Navigation-Bombing System Electronics Data; (2) ARINC Data; (3) NAA Data. These sources agreed quite closely with each other, and the computed MTBF's were within 20 percent of the MTBF now being obtained with production APN 59 radar sets. The same engineers using the same parts failure data, then estimated the MTBF of the

NAVIGATION

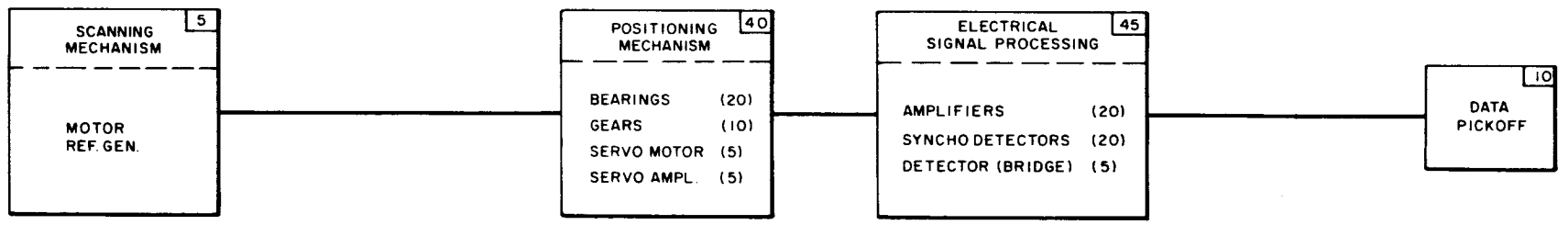
STAR TRACKER



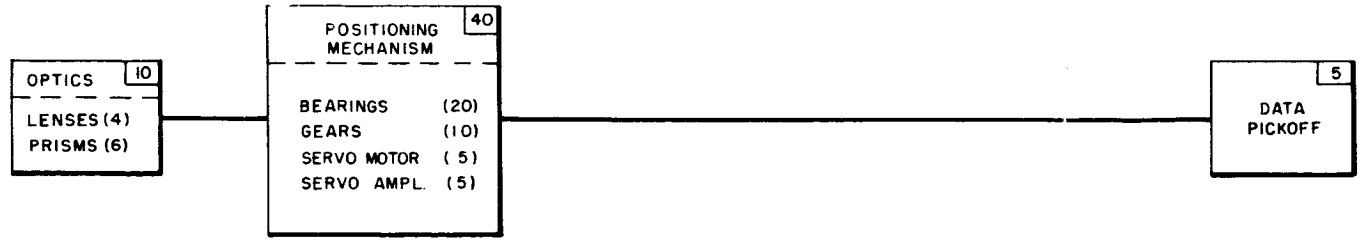
HORIZON SENSOR



SUN TRACKER



NOTE: NUMBERS REPRESENT RELATIVE ORDERS OF COMPLEXITY



SUMMATION OF COMPLEXITY "BITS"	COMPLEXITY RATIO
$\Sigma C = 750$*	15*
$\Sigma C = 494$	10
$\Sigma C = 204$	4
$\Sigma C = 100$	2
$\Sigma C = 55$	1+

*B-58 AUTOMOTIC STAR TRACKER HAS ABOUT THE SAME PARTS GROUPING AS THE OLO STAR TRACKER, HOWEVER, COMPLEXITY IS ABOUT 50% GREATER

Figure B-2. Complexity Diagrams of the OLO Tracker Family

multipurpose candidate radar; this computed estimate is 350 hours. However, a radar designed today can make use of parts which have proven lower failure rates than several of the types used in the APN 59, which was designed some years ago. Also, the lower power will permit the radar described here to achieve somewhat higher reliability. When these factors were taken into account the MTBF estimate was raised to 500 hours.

It should be noted in passing that conventional "parts count" techniques have not been ignored in estimating either relative or absolute reliability. However, they have been used as a reasonableness check rather than as a primary basis for estimate.

The MTBF's estimated for the OLO sensors in the candidate systems are listed below and have been used in the computation of system reliability which serves as one criterion in the selection of the most promising concept.

Candidate OLO Sensor	Estimated MTBF in Hours
Automatic star tracker	200
Automatic horizon seeker	500
Automatic sun tracker	1000
Manual star tracker	1500
Passive sun seeker (stop tumbling)	3000
Passive sun seeker (solar reference)	300
Stadiametric optics (including interferometer, etc.)	2000
Gyro	2000
Platform and electronics	1000
Multipurpose radar	500
Navigation radar	375
Payload beacon	1000

Elements common to all candidates	
Digital Computer (see Vol. V)	10,000
Accelerometer	40,000*

These reliability estimates assume that

The initial failures in individual equipments have been eliminated by production testing.

The system design is capable of withstanding the environment of space, so that failures are random.

This assumes that the development program has included sufficient environmental testing and feedback of information so that the design can withstand space environment. Further, this assumes that there is efficient feedback of failure information and corrective action so that essentially all "weak spots" in the design are discovered and eliminated before the equipment enters the Orbital Launch Operation. For this reason, one primary objective of the development program described in Section V is to achieve early orbital test flights of essential parts of the guidance system, to demonstrate reliability before the beginning of the Orbital Launch Operations.

B-5. RELIABILITY GROWTH

Throughout this study emphasis is placed on guidance systems for which the design can be started immediately, using

- - - - - - -

*Based upon Manufacturer's statement that USAF has officially rated the Arma Vibrating Sting Accelerometer at this figure.

present day components and techniques. The estimated MTBF's are established on this basis.

Starting from the candidates described in Appendix C, for which the MTBF's are estimated, and allowing two additional years for development and improvement of reliability, a considerable increase in the MTBF of some of the important elements can be made with very little or no penalty in weight. Specifically, for equipment which is available and tested in orbital flights by 1967, two years after the equipment in the table above, the following MTBF's can be attained:

Manual star tracker	3000 hours
Gyro	4000 hours
Platform and electronics	2000 hours
Multipurpose radar	1000 hours
Payload beacon	2000 hours

These improved MTBF's are based on two factors: (1) the use of improved components which are continually becoming available through programs such as the Minuteman Reliability Program, and (2) intensive follow-up of failures which are reported during tests and orbital flights of the equipment recommended in this study. It is assumed that all necessary design changes will be made to remedy the causes of failures reported, and the only other design changes which will be made will be substitutuions of parts of known improved reliability.

APPENDIX C

GUIDANCE SYSTEM SELECTION

C-1. GENERAL

This Appendix describes the selection of the guidance system for each vehicle in the OLO complex. Guidance for the payload is discussed first. This is followed by the OLF, the Tug/Ferry and finally the Lunar spacecraft. The guidance system for these three manned vehicles are similar in many respects. In order to present the information upon which the astronaut must base his decisions, a display system is required. In addition, it is assumed that the astronaut will be given a periscope to enable him to perform certain functions. These functions may include:

- Wide angle star pattern recognition
- Final rendezvous and docking
- Crude determination of attitude
- Crude range finding.

Starting with these basic elements (displays and periscope) for the manned vehicles, the selection of the guidance systems can proceed in the following steps:

1. For each of the guidance functions which are defined in Section II, a list of candidate guidance elements for each vehicle is prepared. Each candidate has the means for

generating that particular function, and is sufficiently developed to be included in systems which will fly in 1965, and which will meet the accuracy requirements of Appendix A.

2. For each candidate guidance element, pertinent data on estimated weight, reliability, burden on the human, etc., is assembled for comparison.

3. The most promising among the candidate guidance elements are combined into candidate guidance elements.

4. The candidate systems are compared on the basis of weight, reliability, etc.; the best and the runner-up systems are selected. Reliability is the prime consideration in this selection.

5. The reliability of each selected system is compared to the reliability requirement established in Appendix B, and where necessary, sufficient redundancy is added to the selected system to enable it to meet the requirement for its vehicle.

As was stated in Section III, the selection of guidance systems is conducted independent of the effort reported in Volume IV entitled "Stabilization and Control".

C-2. GUIDANCE FOR THE PAYLOAD

a. Preliminary

The following operational concepts, among those which are stated in Section II, are reiterated here because they are particularly pertinent to payload guidance:

1. The payload will be unmanned

2. The payload will normally be placed within a few miles (ten or twenty) of the orbiting target (OLF or lunar vehicle) by the boost phase of the operation, which will be controlled by the boost guidance system. This may be done either by a direct ascent to the target, or by placing the payload in a phasing orbit so as to close in on the target after one or more revolutions. There will be instances when this does not occur and the payload guidance system must then be capable of being remotely controlled so as to complete the rendezvous from distances as great as 2,000 nm.

3. The type of payload guidance selected is determined by the thrust capability of the payload. It is not affected by the existence or non-existence of an OLF.

4. The payload may be controlled by the OLF, a tug, the ground facilities, or the lunar vehicle, if there is no OLF.

5. In extreme cases, the payload might be in orbit for as long as 30 days before rendezvous is completed.

6. The payload is assumed to be tumbling in a random manner at the beginning of the rendezvous operation.

7. Since the payload is unmanned, maintenance before rendezvous is not feasible.

b. Guidance Functions

As indicated in Section II, the payload guidance system must be capable of performing the following functions:

1. Upon receipt of a signal, the payload guidance system should be capable of waking-up and stabilizing the attitude of the payload.

2. Establishing an attitude reference in inertial space.

3. Maintaining this attitude reference under the varying conditions of orbital flight.

4. Upon receipt of a ΔV signal, the payload guidance system must be capable of aligning the thrust of the payload vehicle along the direction of the required ΔV and of accurately measuring the applied thrust so as to command thrust cutoff within the ΔV error allowance specified in Appendix A.

In addition to performing these functions, the payload must provide the radar beacons, reflectors and/or lights to enable successful tracking of the payload from the ground, OLF or lunar vehicle. The payload must also be capable of verifying each function as it is completed and of transmitting the results of each verification attempt so that emergency action may be taken if a malfunction occurs.

c. Candidate Guidance Elements

(1) General. A number of alternate means of performing the required guidance functions are listed in table C-1 and are briefly described in the order in which the functions will be performed in a rendezvous.

TABLE C1 - PAYLOAD SYSTEM FUNCTIONS

Candidate Sensors	Functions: Wake-up & Stop Tumbling	Establish a Line in Inertial Space (two Lines Required)	Maintain Attitude References	ΔV Determination
1	Gyro	Star tracker	Gyro	Accelerometer
2	Passive sun tracker	Sun tracker	Continuous star track	Radar tracking
3	Horizon seeker	Passive sun seeker	Continous horizon scanner	Optical tracking
4		Horizon scanner	Switch celestial references using inertia of P/L as stable reference during switching (no gyros)	
5		Gyro compass Track the transmitter on OLF or OLV		

(2) Wake Up and Stabilize

- Gyros - Three gyros will be used with each gyro input axis orthogonal to the other two and located along one of the axes of the payload. The torque currents required to maintain null signals at the gyro pickoffs will be proportional to the angular rates around each gyro input axis. These torque signals will be sent to a digital computer which will compute and command the required thrust for each of the vehicle attitude jets to stop vehicle tumbling or change the vehicle attitude as required.
- Passive Sun Sensors - A number of sun sensors, each consisting of one or more photo diodes, are located on the vehicle to provide complete coverage of the celestial sphere. Each diode will receive light power that is proportional to the cosine of the incident angle of the sun's light rays with respect to the diode surface. The photo diode outputs will be sent to the digital computer which will use this information, along with stored information of each sensor location with respect to payload vehicle axes, to determine angular rates and command attitude control signals. A total of 14 sun sensors will be sufficient to indicate and stop the tumbling motion of the vehicle, however the payload attitude will be arbitrary when stopped with this small number of sensors.
- Horizon Scanner - A horizon scanner can stop the payload vehicle from tumbling along two of the three vehicle axis but will require two modes of operation. The first mode will be used to reduce

the angular rates around each of the two vehicle axes to less than one rpm. This will be accomplished by locking the scanning heads and allowing the vehicle motion to provide a coarse scanning pattern. Information from each scanning head will be sent to the digital computer where coarse angular rates are computed and correcting thrusts commanded. Once the vehicle motion has been reduced to less than one rpm, the horizon seeker will determine the vertical in its normal operating mode. From the rate-of-change of this vertical, the computer will determine vehicle angular rates and command the attitude control system.

(3) Establish Lines in Space (Two lines required)

- Automatic Star Tracker - The tracking of distant stars, which are relatively low energy sources as compared to the sun, requires accurate pointing information. Initial alignment may be achieved by adding a star map-matching capability to the basic star tracker and using some of the storage capacity of the digital computer. The star map-matching technique consists of determining the relative angles between stars brighter than a certain magnitude for a portion of the celestial sphere. This information is then sent to the computer which scans its stored map library to match the information. Once the group of stars comprising the map is known, a line in inertial space has been determined. The process is then

repeated for another portion of the celestial sphere and a second line is obtained, and inertial attitude is determined. If a 20 degree conical scan is used and only the 100 most brilliant stars are considered each map need consist of only 3 stars. If four celestial bodies are used all ambiguity caused by planet movement is eliminated. This technique requires approximately 400 words (20 bits/word) of computer storage for the star library. Accuracies in the order of seconds of arc are possible using star trackers.

- Horizon Tracker - Three sensor heads mounted along two mutually perpendicular axes in the payload vehicle are capable of; establishing the local vertical to the earth to within six minutes of arc, of sensing pitch and roll errors relative to the vertical, and of producing signals that are proportional to these errors. In addition, sufficient information is available to determine the altitude of the vehicle, if this is desired.

- Sun Tracker - Tracking of the sun may be used to establish a line in inertial space. Since the earth is rotating about the sun, relative sun position must be computed before a line may be established. A sun tracker has an advantage over a star tracker in that only coarse pointing information is needed since the object being tracked is a high energy source. This advantage results in simple mechanical parts, simpler optics and electronics. In addition, greater degradation in components may be permitted before a malfunction occurs. The star tracker however can establish

both lines to determine attitude, while the sun tracker must be supplemented by a horizon scanner to determine attitude. In addition, the sun tracker cannot track continuously since part of the orbital flight will be on the dark side of the earth. During this portion of the orbital flight a gyro may be used to maintain the reference line. Accuracies on the order of one minute of arc are possible using sun trackers.

- Passive Sun Sensors - A system similar to, but much more complex than, the one described under the "Wake-up and Stabilize" portion of this Appendix can also be used to establish a line in inertial space, once the relative sun position is known. Approximately 75 sun sensors are required to achieve accuracies in the order of 1.5 degrees. The recommended system will utilize signal strength from each diode in the determination of the inertial line to the Sun. The relative gain of each diode must be within 10 percent of nominal if only 75 sensors are to be used. The signal output of each diode is sent to the digital computer for processing. The processed data is then analyzed using correlation techniques and the line to the sun determined. If only on-off data is utilized from each sensor approximately 4000 photo diodes must be used to achieve accuracies in the order of 1.5 degrees.

- Gyro Compass - Since the orbit of the payload in inertial space is known, a reference line may be established with respect to the payload orbit. A system consisting of three orthogonal gyros,

used in a gyro compass mode, can be used to determine a line along the orbit if a vertical reference to the earth is available. By keeping the vehicle's roll and pitch axes perpendicular to the vertical reference, each gyro will sense the angular rotation about its input axis. Knowing the position of each gyro with respect to the vehicle axis, the attitude of the vehicle with respect to the vehicle orbit can be computed.

- Tracking of a Transmitter on the OLF or Spacecraft - A line in space may be established with respect to the known OLF or Spacecraft orbit. This could be accomplished by having an automatic homing system on the payload, however, accuracies of better than 30 minutes at ranges of 2000 nm are not feasible at this time without the addition of excessive equipment on the OLF or Spacecraft and a large directional antenna on the payload.

(4) Maintain Attitude Reference

A number of possible candidates for maintaining attitude reference were considered and these candidates are listed in table C-1. The three-gyro system is the most feasible approach since it is capable of maintaining the attitude reference continuously along all three axes of the vehicle. Each of the other candidates are either not capable of supplying continuous information or require additional equipment to maintain a third axis.

(5) ΔV Determination

A number of candidates were considered to perform the ΔV sensing. Since ranges out to 2000 nm must be considered to command Hohmann orbit transfers, the only candidate capable of meeting the requirements established in Appendix A are the accelerometers located on the payload. Radar and optical tracking is feasible but only at much shorter ranges.

d. Candidate Guidance Systems for the Payload

Table C-2 lists six candidate guidance systems for the payload, briefly indicates their operation, and lists estimated weights, reliabilities, and other information which serves as a basis for comparison. The six candidates have been generated by taking two basic systems, one of which (No. 1) makes use of a star tracker, the other (No. 2) a sun tracker and a horizon tracker. These are then altered in steps to obtain candidates which contain neither star nor sun tracker (No. 3 uses passive sun seekers, and No. 6 uses only gyros and a horizon tracker), or no gyros (No. 4). The remaining system (No. 5) is a variation of the sun tracker - horizon tracker scheme which requires only one gyro.

In addition, each system will require three accelerometers and a digital computer to determine the required ΔV's for orbital transfer. Strapped down components were selected over platform components based solely on accuracy requirements. Further studies are required to insure that the strapped-down configuration does not cause a stabilization problem due to the fact that the gyros and accelerometers are not isolated from the control system. If the strapped-down configuration introduces a severe control problem for the attitude

stabilization system, it is recommended that the platform configuration be investigated so the tradeoffs of more complex guidance hardware but a less complex attitude stabilization system can be evaluated.

The digital computer characteristics, not shown in table C-2, such as computer complexity and size will be determined, in general, by the ΔV computation and the function verification task, both of which are inherent to each of the candidates. Candidates requiring additional computer requirements, and could affect computer complexity, have these requirements tabulated.

e. Recommended Guidance System

(1) The system recommended for use as the payload guidance system is candidate No. 6. This system utilizes three orthogonal gyros to stop vehicle tumbling and maintain the inertial reference lines once established. The reference lines used are: (a) a horizon tracker derived vertical and (b) a gyro compass derived line with respect to the orbit. This system was selected because it represents the simplest system from an operational point of view, has the highest reliability to weight ratio, and requires little additional computer storage and computation. In addition this system is independent of position in orbit.

(2) The system recommended for "runner up" is candidate No. 1. This system is similar to candidate No. 6 but differs in that the second reference line is established by a sun tracker instead of the gyro compass operation. This system is desirable because of its flexibility in handling malfunctions. If the redundant modes of operation are considered,

its reliability to weight ratio is as good as candidate No. 6's. However, it is a more complex system, requiring considerable additional computation and decision making by the computer.

f. Redundancy

The reliability of the recommended guidance system for the unmanned payload is dependent upon the following components and their estimated reliabilities (see Appendix B):

- 0.990 - horizon seeker
- 0.99994 - 4 gyros (1 redundant)
- 0.9995 - digital computer
- 0.999625 - 3 accelerometers

The payload will also contain a radar beacon, but since its reliability is accounted for in the OLF or Tug/Ferry, it has not been included in the payload except in the estimates of weight, power and volume (see table 3-1). Assuming the only redundancy is the recommended fourth gyro, the probability of success based on a five hour trip is 0.989.

If higher reliability is desired, a survey of the possibilities of further redundancy indicates that a redundant horizon sensor head should be added to the horizon tracker (see figure C-1). The addition of this head increases the system probability of success to 0.9985 which exceeds the reliability goal of 0.99, and also increases the total weight of the system by approximately three pounds.

C-3. GUIDANCE SYSTEM SELECTION FOR THE OLF

a. Preliminary

The following operational concepts and assumptions, from among the information presented in Section II, are particularly applicable to the OLF. These concepts are restated since they form an important part of the basis for selecting the recommended guidance system.

(1) The OLF should be self-sufficient insofar as guidance is concerned, that is, it should be independent of information from the ground. (The one exception to this is that the ground must determine the orbit of the OLF with sufficient accuracy for Spacecraft launching.)

(2) The OLF will control the final phases of rendezvous and docking. The OLF will normally assume control when the approaching payload is a few miles (less than 20) away. There are a number of foreseeable circumstances in which the OLF must assume control when the approaching payload is at a greater distance; this is arbitrarily assumed to be 300 nm.

(3) During every rendezvous, the OLF should be capable of monitoring the approaching payload and the movement of a tug, if one is operating, from a distance of 300 nm.

(4) The ground will supply information to the OLF concerning the location of approaching payloads so that the uncertainty volume, within which the OLF must search for the payload, will have a radius no greater than ten miles.

(5) The OLF will have sufficient spare parts for scheduled maintainance of the more critical guidance components.

b. OLF Guidance Functions

As indicated in Section II, the OLF guidance system must be capable of performing the following functions:

(1) Orbit Determination. The OLF must be capable of evaluating the parameters which define its orbit, so that its velocity and position in space at any instant are known. This must be done with sufficient accuracy to enable the OLF to remain in synchronous orbit, if this is desired, but not necessarily accurate enough for the Spacecraft launch operation.

(2) Line-of-Sight Determination. The OLF must be able to establish the line-of-sight to a payload, tug, or other vehicle in order to accomplish rendezvous with, or otherwise direct, the vehicle. This guidance function involves first acquiring the target, and then determining the angle, angular rate, range, and range rate between the target and the OLF. Different schemes might be used to determine line-of-sight over widely different ranges. During the terminal rendezvous and docking maneuver, for example, the payload might be directed with the aid of a TV camera or simply by the operator "eyeballing" it through a calibrated window.

(3) Attitude Determination and Control. The OLF guidance system must establish lines to the stars and planets in order to define the OLF attitude with respect to earth.

Once defined, the attitude of the OLF in orbit must be maintained, and further must be controlled during rendezvous, docking, and other operations. The guidance system must therefore be capable of supplying inputs to the attitude control system for calibration and control.

(4) Generation of ΔV's. Using the previously determined orbit information, the OLF must evaluate deviations from synchronous orbit. It must then calculate a ΔV program which will correct such deviations with minimum fuel expenditure and maintain the desired orbit within the allowable tolerances.

(5) Computer Functions. The OLF computer must be capable of performing the computations associated with all of the previously listed guidance functions, in both the primary and backup modes. In addition, it must have the capacity to assimulate information received from the ground or inserted by the OLF crew. A more detailed and complete list of computer functions is included in Section III.

(6) Display Function. A clear, integrated display must be presented to the OLF crew to enable them to monitor and supervise the various functions which they must perform. During the rendezvous operation, for instance, such quantities as range, range rate, angular line-of-sight, on-off thrust commands, ΔV's, fuel flow rates, etc. would be displayed. For station keeping, the crew would have available displays of attitude, time, position, velocity, fuel, ΔV's, deviations from nominal orbit, etc..

c. Candidate Guidance Elements

(1) General

The major differences among the candidate guidance systems for the OLF lie in the choise of equipment for the determination of the OLF orbit, its attitude, and the line-of-sight to the payload or tug. (Line of sight, as used here, includes the functions of acquisition, determination of angle, angular rate, range and range rate.) The remaining functions of generating ΔV, performing computations, and displays, are straightforward, once the means of accomplishing the orbit and line-of-sight functions have been chosen.

(2) Determination of Orbit

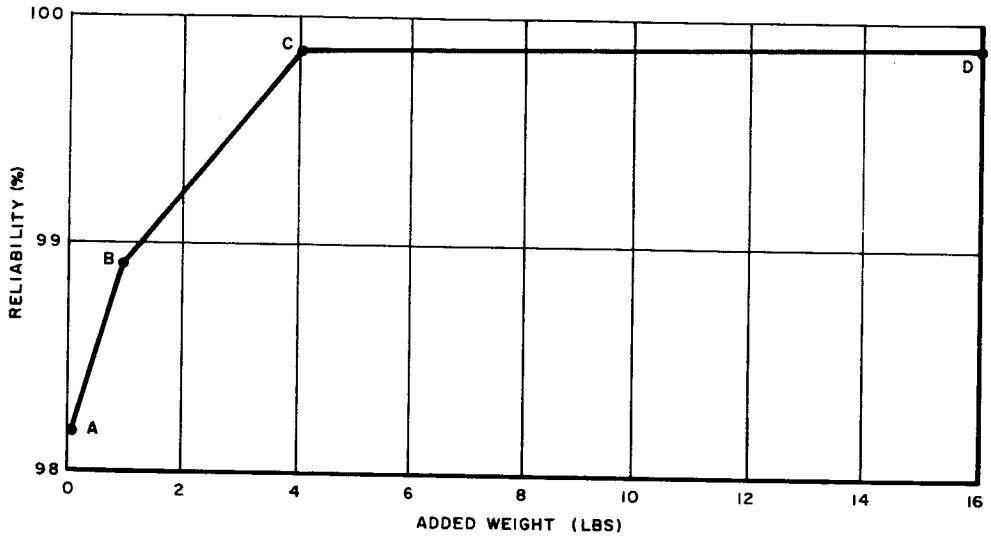
Table C-3 lists the important characteristics of six methods of orbit determination. Each of these is basically a means of making a sequence of position determinations, which are then smoothed by the on-board computer. The orbit parameters are generated from the smoothed data. The six candidates fall into two classes. The candidates of the first class (No's 1, 2 and 3 in table C-3) make position fixes by determining the vertical, the altitude and two lines in space. A radar with conical scan is used for both altitude and vertical (see the next paragraph (3)). The two lines in space are determined either by an automatic star tracker (candidate No. 1), an automatic sun tracker and gyrocompassing (candidate No. 2) or by a manual star tracker (candidate No. 3). Each tracker is mounted on an inertial platform. A description and sketch of such a manual star tracker mounted on an inertial platform is given in Appendix D.

The central element of the second class of candidates is a radar which measures the range to beacons at known locations on the ground. When three beacons are available simultaneously, (candidate No. 4), no other information is needed to complete the determination of position. Appendix D contains a description of this scheme and a discussion of its characteristics. As noted in Appendix D, if only two beacons are available, a horizon tracker must be included in the system to provide vertical information (vertical is required only for pointing the radar accurately) and altitude must also be obtained from the radar. (This is Candidate No. 5.)

The final candidate (No. 6) in table C-3 belongs to the first class of candidates but uses a horizon tracker to determine altitude as well as vertical direction; it has been included in order that the possibilities of a system which does not require a radar are considered. The altitude obtained from the horizon tracker would be combined with vertical and orbital information and smoothed over one or more orbital periods in order to obtain a more accurate altitude determination.

It should be noted that the guidance elements considered for orbit determination in candidate systems 1, 2, 3 and 6 yield attitude information as well. Candidate systems 4 and 5 would require separate attitude sensors. Any of the attitude determining systems mentioned in paragraph (4) of this listing could be used in conjunction with the radar beacon systems (4 and 5) in the synthesis of an overall OLF guidance system.

Table C-3 is comprised of candidate guidance elements whose operational reliability can be proven by 1965, based on the present state-of-the-art. All of these candidate systems will also fulfill the accuracy requirements



POINT A - BASIC GUIDANCE SYSTEM
* POINT B - POINT A + REDUNDANT GYRO
POINT C - POINT B + REDUNDANT HORIZON TRACKER HEAD
POINT D - POINT C + REDUNDANT DIGITAL COMPUTER

* RECOMMENDED REDUNDANCY FOR PAYLOAD GUIDANCE SYSTEM

Figure C-1. Payload Guidance System Reliability versus Added Weight for Redundancy (Tradeoffs)

TABLE

Candidate	Operation	Estimated Reliability for 5 Hour Mission	Estimated Average Power (1)
4. Horizon scanner Star tracker (map matching) Passive sun seekers	1) Horizon scanner and passive sun seekers used to stop tumbling 2) Star tracker establishes both lines in inertial space 3) Tracker and computer smoothing maintains attitude	0.9631*	15 watts
5. Horizon scanner Sun tracker 1 gyro	1) Horizon scanner and single gyro are used to stop tumbling 2) Horizon scanner and sun tracker establish 2 lines in inertial space 3) Horizon scanner and 1 gyro maintain references	0.9817*	21 watts
6. Horizon scanner 3 gyros plus 1 redundant	1) Gyro signals are used to stop tumbling 2) Horizon scanner establishes vertical to earth. Line in orbit plane established by gyro compassing 3) Gyros maintain reference	0.9817* 0.9891**	40 watts

* Normal operating mode - redundancy not considered - computer and ac
**All redundancy, added or inherent, considered
(1)Figures are for strapped - down inertial system
(2)Shelf items are defined as items that are proven designs and requir

C-2 CANDIDATE GUIDANCE SYSTEMS FOR PAYLOAD (Cont)

Estimated eight (1)	Back-up Operation	Additional Computer Requirements
30#	None	1) Contains star map library and performs matching operation 2) Point star tracker 3) Provides smoothing to maintain reference
21#	limited-gyro can back up sun tracker for limited time	1) Compute relative sun position 2) Compute tumbling rates for horizon scanner signals and switch scanner modes 3) Provide smoothing to maintain vertical reference
20#	1) Redundant gyro For a single gyro malfunction	1) Perform gyro compass operation 2) Detect gyro malfunction and switch to redundant one

elerometer reliability included

only fabrication from existing drawings

TABLE

Candidate	Operation	Estimated Reliability for 5 Hour Mission	Estimated Average Power (1)	E W
1. Horizon scanner Sun tracker 3 gyros	1) Gyro signals used to stop tumbling	0.9768*		
	2) Horizon scanner and sun tracker establish 2 lines in inertial space	0.9891**	37 watts	
	3) Gyros maintain reference			
2. Star Tracker with map matching capability 3 gyros plus 1 redundant	1) Gyro signals used to stop tumbling	0.9672*		
	2) Star tracker establishes both lines in inertial space	0.9744**	39 watts	
	3) Gyros maintain references			
3. Horizon scanner Passive sun seeker 3 gyros	1) Gyro signals used to stop tumbling	0.9768*		
	2) Horizon scanner and sun seekers establish 2 lines in inertial space	0.9891**	32 watts	
	3) Gyros maintain references			

* Normal operating mode - redundancy not considered - computer and accele
**All redundancy, added or inherent, considered
(1)Figures are for strapped - down inertial system
(2)Shelf items are defined as items that are proven designs and require onl

C-2 CANDIDATE GUIDANCE SYSTEMS FOR PAYLOAD

stimated eight (1)	Back-up Operation	Additional Computer Requirements
25#	1) Gyros for sun tracker (gyro compassing) 2) Horizon scanner in place of 2 gyros plus computer smoothing for stop tumbling and reference maintenance.	1) Determine relative sun position 2) Perform gyro compassing in back-up mode
21#	1) Redundant gyro for a single gyro malfunction	1) Contains star map library and perform matching operation 2) Point star tracker 3) Detect malfunctioning gyro and switch to redundant one
21#	1) Gyros for sun seeker (gyro compassing) 2) Horizon scanner in place of 2 gyros plus computer smoothing for stop tumbling and reference maintenance	1) Correlate sun seeker data to determine line to sun 2) Compute relative sun position 3) Perform gyro compassing operation in backup mode

ometer reliability included

y fabrication from existing drawings

TABLE C-3. CANDIDATE SYSTEMS FOR ORBIT DETERMINATION

Candidate (exclusive of computer)	Operation	Estimated Wgt	Estimated Power	Remarks
1. Horizon seeker Stable platform (3 gyros) Auto. star tracker Radar altimeter	Series of position fixes determined from vertical, altitude, and 2 lines in inertial space with respect to the vertical. Gyros provide smoothing.	81#	196 watt	1. Also yields attitude information 2. Radar can also be used for line of sight.
2. Horizon seeker Stable platform (gyro-compassing) Auto. sun tracker Radar altimeter	Gyrocompassing gives line in orbital plane. Sun tracker gives line to sun. Position computed from this information plus altitude and vertical. Gyros provide smoothing and memory on dark side of earth.	76#	190 watt	1,2 - Same as candidate (1) 3. Additional computer capability required here as compared with candidate (1).
3. Horizon seeker Stable platform (3 gyros) Manual star tracker Radar altimeter	Same as (1) except operator acquires and sights on stars.	82#	194 watt	1. Also yields attitude 2. Telescope or Radar can be used for acquiring and tracking P/L
4. Pulse radar interrogator to (3) ground beacons.	Position determined by ranging from OLF to (3) ground beacons at known locations. Information smoothed by celestial mechanics over one or more orbital periods.	56#	140 watt	1. Same basic radar with a few minor modifications can be used for altitude and line-of-sight 2. Vertical can be computed from ground ranging and orbit data. 3. Need additional equipment for attitude.
5. Pulse radar with a) Interrogation to (2) ground beacons b) Altimeter with conical scan.	Vertical and altitude determined by conical scan altimeter. Position determined from range to 2 ground beacons and altitude. Data smoothed over one or more orbital periods	56#	140 watt	1. Radar can also be used for line-of-sight. 2. Rough vertical information (10°) required for conical scan radar. 3. Same as candidate (4)
6. Horizon seeker Stable platform (3 gyros) Manual star tracker	Horizon seeker gives vertical and rough altitude, which is combined with star tracker information for approximate orbit. Data is correlated with vertical information and smoothed over one or more orbital periods for accurate orbit determination.	56#	84 watt	1. Larger computer needed here to handle required curve-fitting. 2. Also yields attitude.

TABLE C-4. CANDIDATE ELEMENTS FOR LINE OF SIGHT DETERMINATION

Candidate	Operation	Capabilities (1) Acquisition	Range	Range Rate	Relative Angles	Wgt	Power	Range	Accuracy	Automaticity	Remarks
1. • Manual telescope • Interferometer • Light beacon on target	1. Attitude stabilized telescope determines line-of-sight angles to target 2. Interferometric ranging determines range to target	x	x		x	20# (including stabilization) 4# on target	35W 100W on target	50 mi	Angles ∠ 1 min Range 0.1% Rate 1%	Semi-automatic Operator tracks target	1. Range rate obtained by differentiation of range 2. Need point source of light on target 3. Cannot be used if target is backgrounded by light body
2. • Manual telescope with stadiametric ranging	1. Attitude stabilized telescope determines line-of-sight angles to target 2. Stadiametric tracking used to determine range	x	x		x	20# (including stabilization)	35W	5-10 mi stadiametric ranging	Angles ∠ 1 min Range 2-10% Rate 1%	Semi-automatic Operator tracks target	1. Need stadiametric markings around circumference of target. 2. Cannot be used if target is backgrounded by light body. 3. Differentiate range for rate.
3. • Pulse radar interrogator • Beacon on target	1. Radar scans, acquires and tracks target and gives line-of-sight angles. 2. Range determined from time delay of beacon return.	x	x		x	26# 6# on target	110W 20W on target	300 mi	Angles 0.002 rad Range 0.1% Rate 2%	Primarily automatic	1. Need beacon on target 2. Differentiate for range rate 3. Integration time (T_D) = milliseconds Pulse width (τ) = 1 usec Average Power Pav = .01 watt
4. • Pulse doppler interrogator • Beacon on target	1,2 Same as 1,2 of candidate (3) 3. Doppler shift of beacon pulse used for range rate	x	x	x	x	37# 10# on target	120W 25W on target	300 mi	Angles 0.002 rad Range 0.1% Rate 2%	Primarily automatic	1. Need beacon on target with frequency stabilization to 1 part in 10^8
5. • Non-cooperative pulse radar • Corner reflectors on target	1. Radar scans, acquires, and tracks target and gives line-of-sight angles 2. Range determined from time delay of echo	x	x		x	35#	120W	100 mi*	Angles 0.002 rad Range 0.1% Rate 2%	Primarily automatic	1. *Based on Target Cross Section (σ) = $3m^2$ Integration Time (T_D) = 50 sec Pulse Width (τ) = 10 usec Average Power Pav = 1 watt 2. Differentiate range for range rate
6. • Infrared	1. Infrared sensor locates and tracks target	x			x	15#	30W	100 mi 500 mi	0.1 degree 5.0 degrees	Primarily automatic	1. Cannot be used if target backgrounded by hot body.
7. • Visual observation thru calibrated window or split image rangefinder	1. Operator observes target and determines approximate range and relative angles	x	x		x	10#		≈1-2 mi	Range 5-10%	Manual	Use limited to terminal rendezvous and docking
8. • TV monitors	1. Operator controls TV camera, observes target on screen, and determines approximate range and angles	x	x		x	30#	250W	50 mi		Primarily Manual	1. With Solar illumination
						35# inc. searchlight	750W inc. searchlight	≈1 mi.		Primarily Manual	2. No Solar illumination
9. • Optical radar	Optical Radar yields range, range rate, and angles for limited ranges	x	x	x	x	80# 30# on target	1000W	50 mi	Range 1% Rate 0.5%		1. Corner reflector required on vehicle B 2. Feasibility questionable

(1) All systems require rough pointing information

stated in Appendix A. Other methods were considered and eliminated because of excessive power or weight requirements, questionable feasibility by 1965, conflict with the operational concepts enumerated in subsection C-3, or other deficiencies. Included in this group are brute-force search radar, integrated altimeter signature, complete ground tracking, etc.

(3) Determination of Altitude and the Vertical

The two most promising candidates for measurement of both altitude and vertical at CLO altitudes are the horizon tracker and the radar (with a conical scan capability for measurement of the vertical). The horizon tracker works best at altitudes of a few hundred nm and above. At the lower altitudes, its accuracy degrades because of variations in the height of the earth's atmosphere, and it becomes unuseable below altitudes of the order of 200 nm. Radar, on the other hand, works well at low altitudes. Measurement of vertical by conically scanning is useable to altitudes of a few hundred nm, and range (altitude) measurements are feasible with low power (100 watts peak) to a few thousand nm, if some integration is permitted, and another means is provided to lock the radar on initially at the extreme ranges (see Appendix D).

The accuracy requirements of Appendix A for altitude are sufficiently severe so that a radar altimeter is preferred. In general, whenever a radar is included in a candidate for any function, line-of-sight, altitude, or position, radar will be used for both altitude and vertically measurements in that candidate. When such a candidate is selected to be the recommended system, a horizon tracker may be included later as a redundant mode.

(4) Determination of Line of Sight for Rendezvous

Table C-4 lists the candidate systems for line-of-sight determination, briefly describes the operation of each, and compares some of the characteristics upon which system selection is based. The candidates can be divided into four classifications; namely those with manual telescopes (candidates No. 1 and 2), those using cooperative radar (candidates No. 3 and 4), those with non-cooperative radar, (candidate No. 5), and all others. This last classification is primarily for short ranges and includes infrared (candidate No. 6), visual observation (candidate No. 7), TV monitors (candidate No. 8), and optical doppler (candidate No. 9).

The systems which fall into the first classification have several major drawbacks:

- A relatively large amount of power is required aboard the target (payload) in order for the light beacon to be acquired at long ranges.
- The probability of acquisition is low if the target is backgrounded by a light body such as the earth or sun.
- Determination of range by either interferometric means (candidate system 1) or stadiametric means (candidate system 2) is relatively complex and limited in range.

As seen from table C-4, the non-cooperative radar system (candidate No. 5) has a severe power and weight penalty imposed upon it as compared with the cooperative radar systems. Another factor is the addition of corner reflectors aboard the target to obtain the radar cross

section which will insure a return. Figure C-2 shows the variations in required transmitter average power as a function of range for different integration times. Also shown, for comparison purposes, is a plot of required transmitter average power for Beacon operation (Cooperative radar).

The infrared system has two basic limitations which preclude its use as a primary means of determining line-of-sight:

- The target cannot be acquired if it is backgrounded by a hot body such as the sun.
- Range cannot be determined by infrared means.

The best method of determining line-of-sight appears to be a cooperative radar system for long ranges (of the order of 300 nm) supplemented by another system for the terminal rendezvous (less than 1000 feet) and docking operations. Included in the latter classification are visual observation (candidate No. 7), TV monitor (candidate No. 8), and optical radar (candidate No. 9).

(5) Attitude Determination

It is assumed that the OLF frame of reference will be defined by a geocentric coordinate system in which the radial direction to the earth is established using either a horizon seeker or a conically scanning radar that yields altitude information as well. The candidate systems for determining OLF attitude are essentially those described for the payload with one additional variation; namely the combinations obtained by considering the conical scan radar in addition to the horizon tracker. Another candidate

system that appears attractive for the manned OLF includes a manual star tracker with its optics mounted on a stabilized platform. This method has already been mentioned as part of candidate system 3 for orbit determination, and is described in Appendix D.

The following is a list of the candidate systems for OLF attitude determination, all of which have been described elsewhere in this Appendix.

1. Horizon tracker, stable platform, Sun tracker
2. Automatic star tracker (map matching) gyros
3. Horizon tracker, passive sun seeker, gyros
4. Horizon tracker, star tracker, passive sun seeker
5. Horizon tracker, sun tracker, 1 gyro
6. Horizon tracker, gyrocompassing
7. Horizon tracker, stable platform, manual star tracker

8-13 Same as 1 and 3-7 except that conical scan radar is substituted for horizon tracker.

Candidate system No. 2 yields inertial attitutde only, so that orbital information must be added to obtain OLF attitude in geocentric coordinates. The components of candidate system No. 6 and No. 12 will establish the vertical and the direction of the orbital plane. Orbital information is needed to fully define vehicle attitude as a function of time.

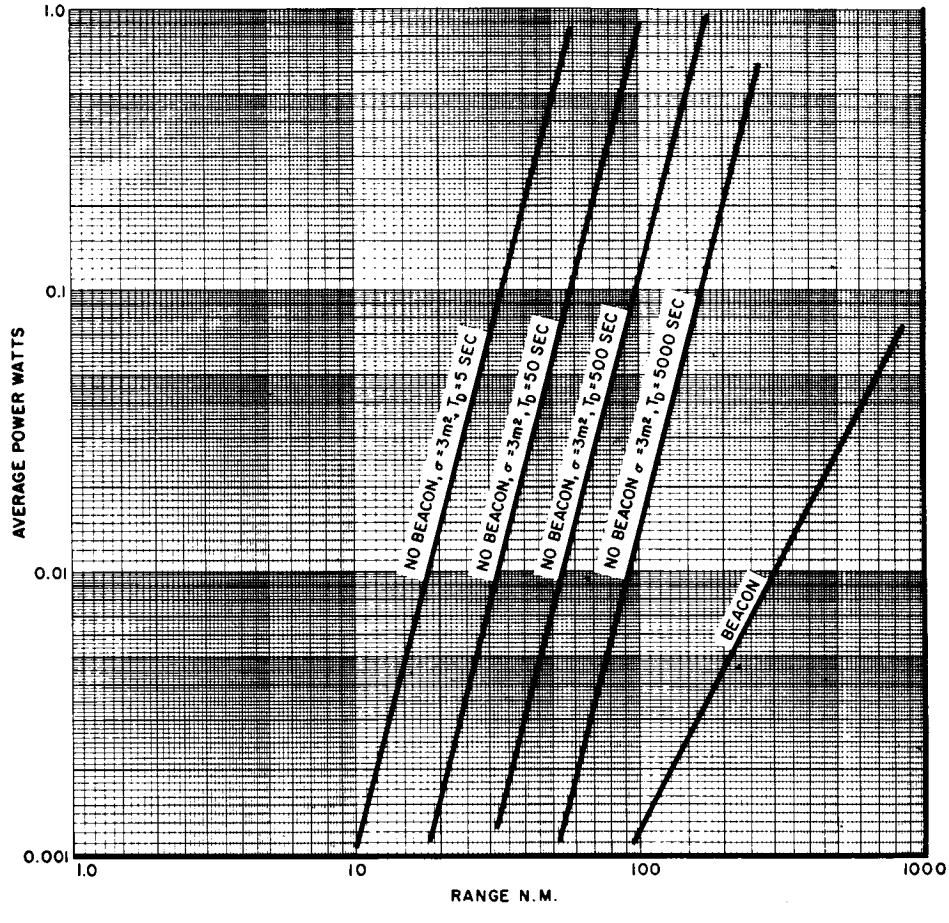


Figure C-2. Transmitter Average Power versus Range for Varying Integrations (Non-Cooperative Case) and Beacon On

(6) Generation of ΔV's. Accelerometers are the obvious choice for the measurement and control of ΔV's for station keeping and other maneuvers. Three mutually orthogonal accelerometers are required, and could be either strapdown or mounted on a stable platform.

(7) Computer and Display

The display is dictated by the final outputs (attitude, line-of-sight, etc.) of the overall guidance system, and will be essentially the same for all of the candidate systems. Display considerations are discussed, in detail in Section IV.

Each of the candidate guidance systems will, include a computer, the choice of which is discussed in Volume V of this study. The computing functions required for the recommended system influence the size of the computer, and are included as part of the system description in Section 3.

(8) Candidate Guidance Systems for OLF

Table C-5 lists and gives the characteristics of six candidate systems which can be used to perform the overall OLF guidance functions. These systems were synthesized by adding to each of the candidate systems for orbit determination, the additional equipment required to determine line-of-sight, attitude, and ΔV's. Where possible, equipment which contributed to more than one guidance function was chosen for the overall system. For example, a radar altimeter meets the accuracy requirements better than a horizon tracker; also radar is the recommended way to

determine line-of-sight to a range of 300 nm. Therefore, in candidate 5, a radar is used for altitude, vertical, rendezvous target acquisition and angle, range and range rate to target.

Some other factors to be noted regarding the selection of the six candidate systems are as follows:

- Candidate systems No's 1, 2, and 3 for orbit determination require only the addition of accelerometers for ΔV to complete the overall guidance system. With only minor modifications, the radar altimeter also serves as a beacon interrogator for line-of-sight.
- The manual star tracker mounted on an inertial platform was selected as the attitude-determining portion of candidate systems No's 4 and 5 because its reliability is higher than either the automatic star or sun tracker.
- Candidate system No. 6 was included in order to provide a comparison of a non-radar system with the 5 other radar-equipped systems.

The components listed in table C-5, under the column headed "CANDIDATES", are the equipment used to perform the guidance functions in the primary mode of operation. The numbers in the "RELIABILITY" column represent the probability of successfully accomplishing these functions over a 20 hour period.

(9) Selection of the Recommended System

A comparison of the candidate systems in table C-5 indicates that although the all-optical system (candidate No. 6) has the lowest weight and power requirements, it must

TABLE C-5. CANDIDATE GUIDANCE SYSTEMS FOR OLF

Candidates(1)	Operation	Reliability	Weight (lb)	Power (Watt)	Automaticity	Inherent Redundancy	Limitations	Remarks
1. • Horizon seeker • 3 gyro stable platform • Accelerometers • Automatic star tracker • Pulse radar with conical scan • Digital computer	1. Horizon seeker gives local vertical, radar gives altitude. Star tracker on stable platform gives 2 lines in space for attitude and orbit computation, which is smoothed by gyros and orbital mechanics. 2. Line-of-Sight to payload is determined using radar and beacon. 3. ΔV's measured by accelerometers.	Excluding Computer 0.8838	81	196	Primarily automatic, with manual monitoring.	Pulse radar with conical scan can be used to determine vertical, if given approximate (10°) pointing information. Horizon seeker can be used for rough altitude data.		1. Beacon required on payload-6#, 20 Watt.
2. • Horizon seeker • 3 gyro stable platform • Accelerometers • Sun tracker • Pulse radar with conical scan • Digital computer	1. Horizon seeker gives local vertical, radar gives altitude. Sun tracker on stable platform gives one line in space for attitude. Gyro compassing gives a second line for orbit. 2. } Same as 2, 3 of candidate (1). 3. }	0.9016	76	185	Primarily automatic.	Same as for candidate (1).	1. Sun not available for tracking about 30% of the time	1. Beacon required on payload. 2. Additional computer capability required for gyrocompassing and sun position as compared with candidate (1).
3. • Horizon seeker • 3 gyro stable platform • Accelerometers • Manual star tracker • Pulse radar with conical scan • Digital computer	1. Same as 1 of candidate (1) except star sightings made with manual telescope. 2. 3. Same as 2, 3 of candidate (1).	0.9032	82	195	Operator required to recognize and track stars for orbit and attitude determination	Same as for candidate (1).		1. Beacon required on payload 2. Wider choice of stars for tracking than in (1)
4. • Pulse radar with conical scan and ground ranging • Ground beacons (3) • 3 gyro stable platform • Manual star tracker • Accelerometers • Digital computer	1. Position determined by ranging to 3 ground beacons at known locations. Series of position fixed smoothed to determine orbit. 2. Computer determines vertical from 3 point ranging and orbit data. 3. Manual Star tracker gives lines in space for attitude. 4. Same as 2, 3 of candidate (1).	0.8913	98	228	Operator required to track stars for attitude	Conical scan radar can be used for vertical	1. Ground beacons not presently available	1. Weight and power penalty for radar ground ranging capability. 2. This radar would require 3 ground beacons within 800 mile circle (max.) for a single position fix. 3. Beacon required on payload.

TABLE C-5. CANDIDATE GUIDANCE SYSTEMS FOR OLF (Cont)

Candidates(1)	Operation	Reliability	Weight (lb)	Power (Watt)	Automaticity	Inherent Redundancy	Limitations	Remarks
5. • Pulse radar with conical scan and ground ranging • Ground beacons (2) • 3 gyro stable platform • Manual star tracker • Accelerometers • Digital computer	1. Position determined by ranging to 2 ground beacons + altitude. Series of position fixes smoothed to compute orbit. 2. Vertical from conical scan radar. 3. Tracker gives lines in space for altitude computation. 4. Same as 2, 3 of candidate (1).	0.8913	98	228	Same as (4)		1. Same as (4). 2. Radar time-shared for ground ranging, line-of-sight, and vertical/altitude	1. Same as candidate (4). 2. Radar requires 2 ground beacons within 800 miles (max.) of each other for a single position fix. 3. Beacon required on payload.
6. • Horizon seeker • 3 gyro stable platform • Manual star tracker • Accelerometers • Periscope with stadiametric ranging. • Digital computer	1. Horizon seeker gives vertical and rough altitude. Data smoothed by gyros and celestial mechanics for orbit determination. Star tracker yields attitude. 2. Periscope with stadiametric ranging used for line of sight. Differentiate range for range rate in computer. 3. ΔV's measured by accelerometers	0.921	56 30*	84 25*	Operator required to track stars for orbit and attitude and track payload for line-of-sight	Manual periscope could be used for star tracking, vertical, and altitude determination functions but with reduced accuracy and additional computer requirements	1. Stadiametric ranging limited to ≈5-10 miles. 2. Target cannot be tracked if backgrounded by Sun.	1. Requires light source on Payload wt = 4 lb, Pwr = 100W for 50 mile range. 2. *Estimated weight and power for servo-gimballed periscope for line-of-sight function only.

[illegible] SATELLITE GUIDANCE SYSTEMS (1 to 10 lbs)

Candidates	Operation	Accuracy (n.m.)	Weight (lb)	Power (watts)	Reliability	Inherent Redundancy	Limitations	Remarks
3. ● [illegible] radar with stadiametric and ground ranging ● Ground beacons (2) ● 3 gyro stable platform ● Manual star tracker ● Accelerometers ● Digital computer	1. Position determined by ranging to ground beacons and bearing to star (sun). Series of position fixes determined by orbit. 2. Vertical from combined data radar. 3. Tracker gives lines in space for attitude computation. 4. Same as No. 3 of Candidate (2).	[illegible]	[illegible]	[illegible]	Same as (2)		1. Same as (4). 2. Radar time-shared for ground ranging, line-of-sight, and vertical/altitude	1. Same as candidate (4). 2. Radar requires 2 ground beacons within 800 miles (max.) of each other for a single position fix. 3. Beacon required on payload.
4. ● Horizon Seeker ● 3 gyro stable platform ● Manual star tracker ● Accelerometers ● Periscope with stadiametric ranging. ● Digital computer	1. Horizon seeker gives vertical and rough altitude. Data smoothed by gyros and celestial mechanics for orbit determination. Star tracker yields attitude. 2. Periscope with stadiametric ranging used for line of sight. Differentiate range for velocity in computer. 3. Data monitored by ground [illegible]	[illegible]	25 2.5	50 15	Accuracy required to track stars for orbit and attitude and track payload for line-of-sight	Manual periscope could be used for star tracking vertical. Sun altitude determination function but with reduced accuracy and additional computer requirements	1. Stadiametric ranging limited to ≈5-10 miles. 2. Target cannot be tracked if backgrounded by Sun.	1. Requires light source on Payload wt = 4 lb, Pwr = 100W for 50 mile range. 2. *Estimated weight and power for servo-gimballed periscope for line-of-sight function only:

be eliminated on the basis of insufficient range for line-of-sight determination. It is interesting to note that, contrary to what might be expected, this system does not have a large reliability advantage compared with the radar-equipped systems.

Candidates No. 4 and 5 utilize radar ranging to fixed ground beacons for orbit determination. This has the advantages of being relatively simple so far as the demands on the operator and computer are concerned. However, the weight and power requirements are greater than for any of the other systems. This combined with the uncertainty of having the required beacons, causes rejection these systems.

The remaining candidates are similar in all respects with the exception of the method of determining orbit. All other considerations being equal, candidate No. 1 is rejected on the basis of being the least reliable. Candidate No. 2 has a number of limitations which do not exist in candiate No. 3. To completely determine the OLF orbit candidate No. 2 must gyro compass to establish the orbital plane in addition to determining a line to the sun. The sun tracker has two additional minor limitations in that the sun is not available for tracking for approximately 30 percent of the orbital trip, (assuming inclinations of about 30 degrees), and the relative sun position must be recomputed periodically. The remaining candidate (No. 3) has been chosen as the recommended system for the OLF. Since the reliability of candidate No. 1 is only slightly lower than candidate No. 2's and since the limitations of candidate No. 2 can be easily overcome, it is recommended that both systems be considered as runner-up systems.

(10) Redundancy With Full Performance Capability

The recommended guidance system is candidate No. 3, which in addition to the displays and manual periscope, contains the following elements:

- Multipurpose radar (described in Appendix D) for altitude vertical and line-of-sight.
- Manual star tracker (described in Appendix D) which is mounted on the platform.
- Inertial platform with three gyros and three accelerometers.
- Digital computer.
- Radar beacon on payload.*

This is the basic system, or primary mode of operation. Two levels of redundancy have been added. This subsection describes the redundancy added to each of the listed elements, to give the entire OLF guidance system a reliability of approximately 0.99 with full performance capability. The following paragraph (11) discusses further backups, usually manual, for each function, which gives reduced performance and which would be used only in the one percent of cases in which a failure occurs; including these, the total reliability is far beyond 99 percent.

These redundancies are summaried in table 3-5 in Section III.

- - - - - - -

* The reliability of the radar beacon on the Payload affects the probability of completing a rendezvous mission which is a function of the OLF, therefore, it is included in the OLF guidance system.

Using the estimated reliability for each of these components based on a 20 hour mission (5 hours for the radar beacon and manual star tracker) which are given in Appendix B, the probability of success is 0.9032. In order to increase this to 0.99, there must be added redundancy. The reliabilities of the individual components (for a 20 hour mission) are:

- 0.9498 - Platform (with gyros and accelerometers)
- 0.9608 - Multipurpose radar (with con. scan)
- 9.995 - Radar beacon on P/L (5 hour mission time)
- 0.9967 - Manual star tracker (5 hour mission time)
- 0.998 - Digital computer

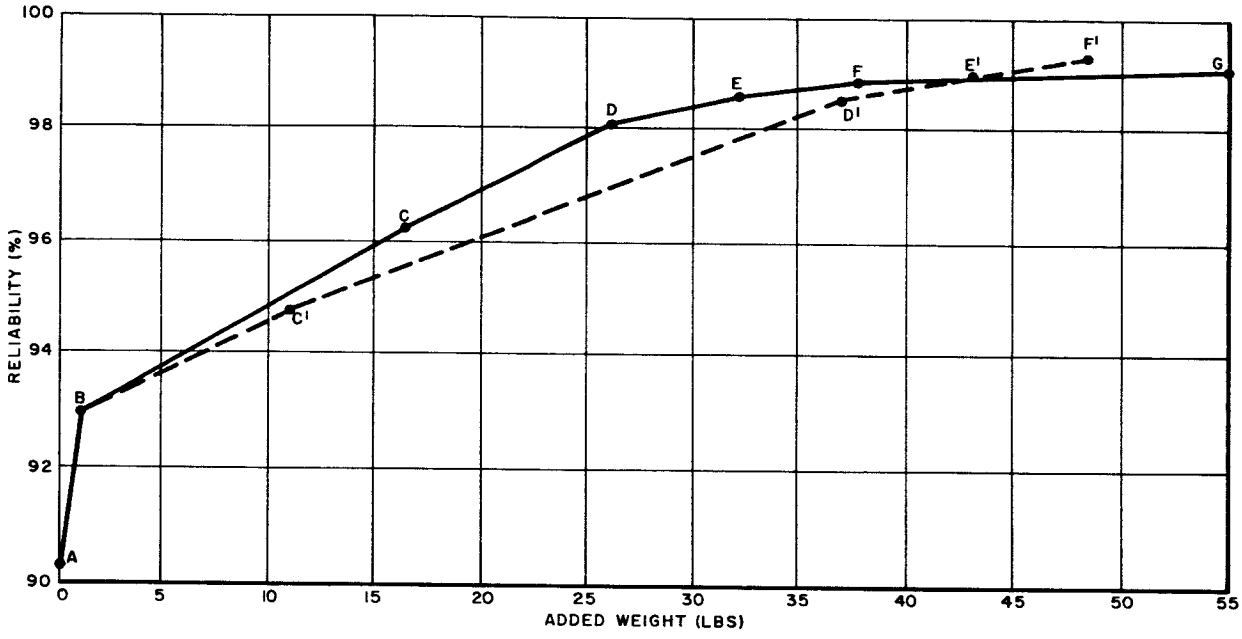
The most critical component is the platform with the three gyros. To increase its reliability, the platform electronics are backed-up and a fourth gyro (capable of replacing anyone of the three gyros) is added. This redundancy increases the platform reliability to 0.9974 and the probability of success of the overall system to 0.9477. The next critical component is the pulse radar. On the basis of adding redundancy to all parts of the radar except the antenna, a radar reliability of 0.9936 was calculated. With both of these components (platform and radar) backed up as stated, the probability of success is increased to 0.9810. By adding complete redundancy to the beacon on the payload, the reliability of success to 0.9859. If redundant servo drives and data pickoffs are added to the manual star tracker the system probability of success increases to 0.9893. Figure C-3 is a curve of reliability vs. weight, as redundancy is added to the basic system, with performance in the

redundant modes equal to the primary mode. It indicates that the attainment of the reliability goal of 0.990 for a 20 hour mission would require considerable additional weight or two antennas. The recommended system, therefore, is that which is estimated to have a reliability of 0.9893 with the amount of redundancy specified for the radar, inertial platform, beacon and manual star tracker.

(11) Further Redundancy With Reduced Capability

To provide for the approximately 1 percent of the missions in which the primary or equivalent redundant modes may be expected to fail, additional backups are added so that each function can still be accomplished, usually with reduced performance. These additional backups are:

- A manual periscope with a simple provision for estimating range to the object sighted; this will provide a backup for acquisition of target, angle, and range, in case of failure of these functions of the multipurpose radar.
- Manual positioning of the star tracker telescope, and manual readout of data will be available in case of failure of the electronics.
- Essential computations and control functions will be performed manually in case of failure of the digital computer or control elements.
- IR horizontal tracker for altitude and vertical, in case of failure of these function of the multipurpose radar.
- Altitude control system gyros and accelerometers will be used in case of failure of the corresponding units of the inertial platform.



POINT A - BASIC GUIDANCE SYSTEM
POINT B - POINT A + REDUNDANT GYRO (CAPABLE OF REPLACING ANY ONE OF THREE GYROS)
POINT C - POINT B + REDUNDANT MULTIPURPOSE RADAR (EXCLUDING A REDUNDANT ANTENNA)
POINT D - POINT C + REDUNDANT PLATFORM ELECTRONICS
POINT E - POINT D + REDUNDANT RADAR BEACON ON PAYLOAD
* POINT F - POINT E + REDUNDANT SERVO DRIVES AND DATA PICKOFFS ON MANUAL STAR TRACKER
POINT G - POINT F + REDUNDANT DIGITAL COMPUTER

POINT C'- POINT B + REDUNDANT PLATFORM ELECTRONICS
POINT D'- POINT C' + REDUNDANT MULTIPURPOSE RADAR (INCLUDING A REDUNDANT ANTENNA)
POINT E'- POINT D' + REDUNDANT RADAR BEACON ON PAYLOAD
POINT F'- POINT E' + REDUNDANT SERVO DRIVES AND DATA PICKOFFS ON MANUAL STAR TRACKER

* RECOMMENDED REDUNDANCY FOR THE OLF GUIDANCE SYSTEM

Figure C-3. OLF Guidance System Reliability versus Added Weight for Redundancy (Tradeoff)

C-4. GUIDANCE FOR THE TUG/FERRY VEHICLE

a. Preliminary

The following operational concepts, from among those developed in Section II, are restated here because they are especially applicable to the selection of a Tug guidance system.

- So far as the guidance system is concerned, the Tug and Ferry vehicles are assumed to be interchangeable
- Because the Tug/Ferry will be used to return one or more of the crew to earth, this guidance function must be thoroughly backed up
- During a rendezvous maneuver, the Tug will monitor the line-of-sight to any of the other vehicles (P/L, OLF) involved
- If necessary, the Tug must be able to rendezvous with an unmanned payload at distances up to 300 nm., and command the payload, if it is operable
- In the event no OLF is used, the Tug guidance may be employed to accomplish rendezvous between the Spacecraft and the fuel payload.

b. Tug/Ferry Guidance Functions

The guidance system aboard the Tug/Ferry must be such that it can perform the following functions:

1. The guidance must be capable of evaluating the Tug's orbit with sufficient accuracy to enable it to rendezvous with another vehicle or return to earth.

2. The guidance system must be able to determine the attitude of the Tug and generate signals for attitude stabilization and control.

3. The Tug guidance must be able to determine the line-of-sight to the payload or other vehicle with which it is performing a rendezvous. Included in this function are:

   - Acquisition of the target
   - Determination of the range and range rate to the target
   - Determination of relative angles and angular rates to the target.

4. The Tug must be able to compute the orbit of the payload or other target, from its own orbit and line-of-sight confirmation.

5. The Tug must be able to compute the best transfer path to accomplish rendezvous with the target. It must therefore compute, measure, and control the ΔV program corresponding to such a path.

6. The Tug must be capable of safely re-entering the earth's atmosphere. The guidance system must therefore:

   - Line up the Tug with respect to attitude, position, and velocity prior to re-entry

- Compute and execute a deboost ΔV program to safely pass into the re-entry corridor
- Measure and control attitude, velocity, and altitude to remain within the re-entry corridor
- Provide the necessary information to permit navigation to the desired landing region.

Implied in all of the functions listed above is sufficient computer capability to perform these functions, as well as an integrated display to illustrate their end results.

c. System Selection

The required functions for the Tug/Ferry include all those required for guidance of an OLF with three modifications:

1. The Tug/Ferry must re-enter the atmosphere.
2. The Tug/Ferry must measure the line-of-sight and execute rendezvous while it is moving.
3. The Tug/Ferry must compute and generate much larger DV's than the OLF.

The general requirements, ranges over which the line-of-sight is to be measured, accuracies, etc. - are not very different from corresponding requirements for the OLF. To meet all requirements except earth re-entry, therefore, the same candidates would be established, and essentially the same system selected as for the OLF, in subsection C-3.

Insofar as the primary mode of operation is concerned, this system can accomplish the earth re-entry function without the addition of any major system elements. The only changes involved would be to the computer and display functions, and to some interconnections among the various system elements. The recommended guidance system for the Tug/Ferry is therefore the same as that recommended for the OLF with the following changes:

1. More extensive use of radar-inertial mixing in order to automatically compensate for the effects of thrusting. Combining accelerometer data with radar range data will allow a long integration time for searching for the P/L in the event the target beacon is inoperative. A range of the order of 100 nm with 4 minute integration time will be possible assuming a 3 square meter radar target crossection. (It is noted that if inertial mixing, longer pulse widths, and integration delay is not used, this range is reduced to approximately 18 nm. Provision can be made for inserting into the computer information received from the ground, such as P/L range rate, which would also aid in acquiring the target. The vertical accelerometer data would be mixed with radar range data when the radar is used as an altimeter in order to obtain more accurate altitude information. Once an initial altitude is established, the radar antenna could be shed during reentry, if necessary, and altitude measured inertially.)

2. Inclusion of computer functions to allow the Tug to automatically compensate for the effects of thrusting on orbit, attitude, and line-of-sight. The computer would also require the capability of determining and monitoring a reentry program.

3. Modification of the display to accommodate the particular environment of the Tug/Ferry. Although essentially the same guidance quantities as are displayed in the OLF will be displayed here, their presentation will differ because of space limitations and the fact that the Tug may be a one-man operation. In addition, those quantities relating to reentry must be displayed in the Tug/Ferry.

4. Because crew safety is of primary concern, backup guidance elements must be provided, which will allow a safe reentry to earth should any or all of the primary mode elements fail.

d. Description of the Selected System

(1) Basic System

It has been shown that, so far as the primary mode of operation is concerned, the Tug/Ferry guidance system contains essentially the same elements as the OLF guidance system. These elements are listed below:

- Reliable multipurpose radar, with beacon-interrogator and altimeter capabilities. Conical scan allows target angle tracking
- Three gyro stable platform for attitude reference

- Manual telescope mounted on platform for star tracking
- Horizon tracker for determination of local vertical
- Three accelerometers for ΔV measurement
- Digital computer
- Radar beacon on payload
- Displays.

(2) Redundancy

Using the estimated reliability for each of the listed elements, based on a 15 hour mission (5 hours for the radar beacon and manual star tracker), which are given in Appendix B, the probability of success is 0.9245. If we assume the radar beacon on the payload has been backed-up, in order to increase the probability of success of the OLF, the probability of success for the Tug/Ferry guidance system becomes 0.9291. In order to increase this to the desired 0.99, the following redundant elements have been added to the basic system:

- One gyro, which can replace any of the original three
- Platform electronics
- Servo drives and data pickoffs for the manual star tracker
- Radar electronics
- Payload beacon (Also required for OLF reliability).

The addition of these elements, together with the inherent redundancy of the conical scan radar for vertical determination, increases the probability of mission success to 0.992. Figure C-4 indicates that the added redundancy increased the total system weight by about 32 pounds

To provide for the less than 1 percent of the missions in which the primary or equivalent redundant modes may be expected to fail, additional backups are added so that each function (including re-entry) can still be accomplished, usually with reduced performance. These are:

- A manual periscope with a simple provision for estimating range to the object sighted; this will provide a back-up for acquisition of target, angle, and range, in case of failure of these functions of the multipurpose radar
- Manual positioning of the star tracker telescope, and manual readout of data will be available in case of failure of the electronics
- Essential computations and control functions will be performed manually in case of failure of the digital computor or control elements
- I.R horizon tracker for altitude and vertical, in case of failure of these functions of the multipurpose radar
- Attitude control system gyros and accelerometers will be used in case of failure of the corresponding units of the inertial platform.

(3) Block Diagram

Figure 3-4 is a block diagram of the Tug/Ferry guidance system, showing the data flow among the various guidance elements.

e. Radar-Inertial Mixing

As stated previously, the mixing of radar range data and accelerometer data can be extremely useful in accomplishing at least two guidance functions while the vehicle is thrusting. First, by combining vertical accelerometer data with radar altimeter data, dynamically exact altitude data can be obtained to a much higher degree of accuracy than would be possible using a radar altimeter alone. This is especially true during re-entry or other periods of large ΔV change. The concept of radar-inertial mixing in altimeter or line-of-sight systems is treated more extensively in Section D.

Secondly, in the event the target beacon is inoperative, the chances of detecting and tracking the target are almost impossible unless inertial information is added to the radar. As the vehicle thrusts, the accelerometers instantly measure the changes in velocity, which is used to update the velocity that positions the radar. This permits the radar bandwidth to be drastically reduced and allows long integration times which are required for detecting an echo from a small cross sectional target area. If the range between the Tug/Ferry and the target is changing very rapidly, this type of "range gate stabilization" may be useful even when the beacon is operative.

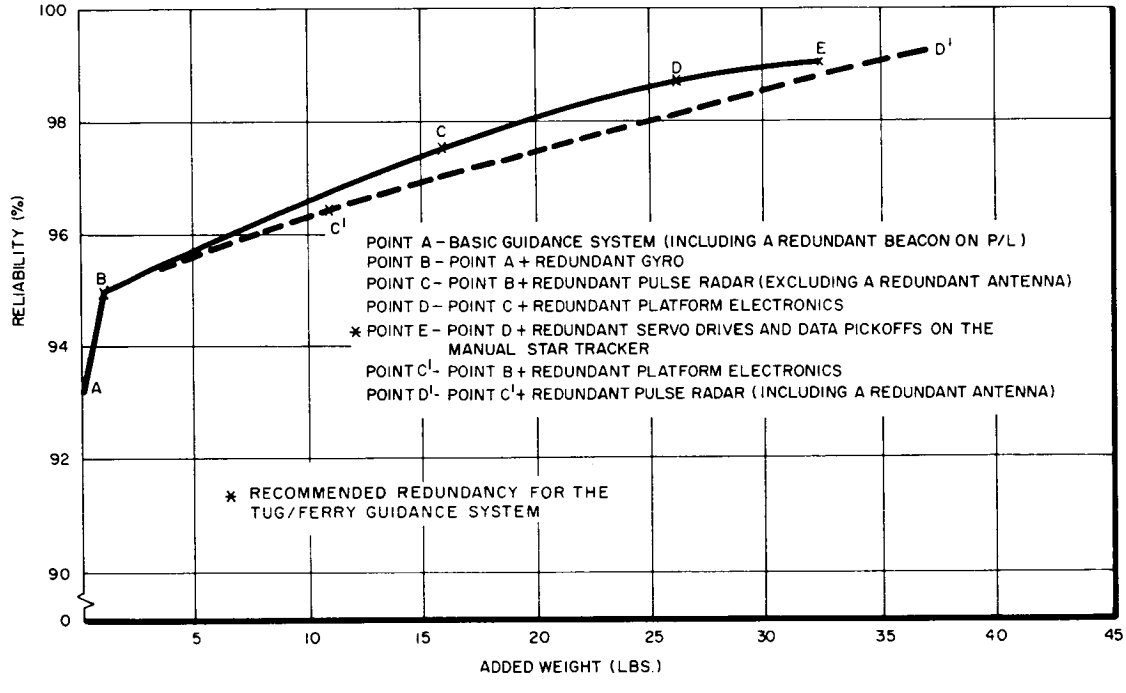


Figure C-4. Tug/Ferry Guidance System Reliability versus Added Weight for Redundancy (Tradeoff)

APPENDIX D

EQUIPMENT CONCEPTS

D-1. GENERAL

The candidate guidance systems which have been considered in this study have stressed proven techniques and guidance elements which are presently available or easy modifications of existing designs. This has been done to assure that the systems which are selected can have proven reliability before The Orbital Launch Operation commences. In general, these guidance elements need no special descriptions. Three of the elements, however, are sufficiently unique to warrant discussion, and are described in this Appendix. The elements are described in sufficient detail to indicate feasibility and to provide a basis for estimates of weight, reliability, etc. in the table of candidates. These three elements are,

- A reliable multipurpose radar for measuring altitude, earth vertical, and range and angle to a rendezvous target. Reliability is the primary consideration in the multipurpose radar design.
- A manual celestial tracker mounted on an inertial platform. This is essentially an automatic star tracker which has been simplified and made extremely reliable by replacing the detector and signal processing electronics by the astronaut.
- An accurate means of orbit determination by means of a radar aboard the spacecraft which interrogates accurately located beacons on the ground.

The OLO guidance system studies show that the first two of these units are required in the guidance systems of the lunar spacecraft, the tug/ferry and the OLF. The third, while not an element of a selected system, is a significant candidate.

D-2. RELIABLE MULTIPURPOSE RADAR FOR DETERMINING ALTITUDE, VERTICAL, and LINE-OF-SIGHT

These paragraphs describe a light-weight low-powered multipurpose radar for use as a primary OLF sensor. Its functions are:

- Rendezvous line-of-sight search operation.
- Acquisition and track of beacon equipped vehicles
- Altimetry and verticality determination.

The application of the radar for the lunar spacecraft is discussed in Paragraph D-3.

The chief objective in establishing the radar configuration is reliability and simplicity within the framework of acceptable physical characteristics and performance. To this end, the radar employs, mainly, standard proven techniques, low transmitter power for all functions, and a minimal number of components to perform the desired sensing tasks. The estimated MTBF is 500 hours, as explained in Appendix B, Paragraph B-4.

a. Preliminary Design Characteristics

The preliminary design characteristics are listed in this paragraph. These represent estimates based on initial definition of the sensor requirements. Unless otherwise indicated, characteristics are for all required radar functions.

(1) Type

- Interrogator/Beacon - Rendezvous
- Interrogator only - Altimeter and Verticality

(2) Frequency

- Nominally 10 kmc
- Interrogator and Beacon Frequencies Offset

(3) Pulse Duration

- 1 microsecond

(4) Modulation

- Pulse code - (lends itself to multi-beacon interrogation and addition of command/data link on single frequency)

(5) PRF - 200 pulses per second

(6) Beamwidth

- 2.5 degrees

(7) Beam Modulation

- Conical scan for angle track and verticality determination
- Squint angle is approximately one beamwidth

(8) Detection Probability

- 99%, negligible delay time with beacon, minimal smoothing for altimetry and verticality

(9) Accuracy

- Range (altitude) - <0.1%
- Range Rate - <2%
- Angle - 3 miles

(10) Range Limits

- 300 nautical miles
- 1000 feet minimum range

(11) Physical Data - Interrogator

- Weight - 26 lbs.
- Space - 1.0 $ft.^3$ (antenna swept volume is approximately 2.5 $ft.^3$)
- Power Drain - 80 watts (ac)
  30 watts (dc)

(12) Transmitter Power

(a) Interrogator

- 50 watts peak
- 10 milliwatts average

(b) Beacon

- 50 watts peak
- 10 milliwatts average

(13) Reliability

Interrogator - 500 hrs. MTBF
Beacon 1000 hrs. MTBF

(14) Gimbal Limits

- ± 45° both axes
- - 100° vertical axis for altitude and verticality modes.

Regarding the selection of a 10 kmc operating frequency, this was based largely on an estimate of acceptable antenna diameter (30 inches or less) to achieve desired performance. However, if larger antennas can be accommodated, the operating frequency is reduced. The transition point where there is an appreciable increase in reliability occurs at approximately 4 kmc. These frequencies permit the use of more reliable transmitting tubes than klystrons, which are currently the limiting reliability item. In reducing the operating frequency to 4 kmc (and accepting the change in antenna size, particularly) the reliability figure can be improved by at least a factor of two, in a radar designed at the present time, beyond the estimated 500 hours.

b. Functions

The rendezvous radar design configuration readily lends itself to altimetry and verticality determination. A brief discussion of each of these functions and the manner in which the radar is employed follows.

(1) Rendezvous

It is required to search, acquire and track injected payloads and other rendezvousing vehicles out to ranges of 300 miles; ability to acquire and lock-on will depend mainly on accuracy of initial location data to the radar.

For design purposes, a 15 to 20 percent accuracy in initial position information is assumed and can be easily obtained. Further, rates between vehicles are expected to be small when the radar is in use, thereby minimizing demands for high angle slew and/or range rates. The pertinent features relating to the rendezvous function are as follows:

- Search is accomplished in angle by driving the reflector in a spiral program. Regarding range search, it is expected the entire range sweep can be strobed without having to gate only small segments. It is assumed that the OLF and the antenna have previously been grossly oriented.
- Conical scan is initially proposed as the technique for automatic angle track. The reflector remains stationary except for the search mode and for small perturbations during track. A nutating feed, specifically designed for operation in a space environment, modulates the beam. Conical scan provides the simplest and most reliable means to angle track.
- Detection time is negligible within the constraints previously defined. Availability of a repeater beacon aboard the rendezvousing vehicles minimizes transmitter power requirements. This permits low power transmitter operation at a fraction of rated value, thereby assuring reliable operation.
- Maximum effort has been made to retain simplicity in configuring this radar. Only absolutely necessary features are included. In this regard, it is planned to use available information in the

range tracking unit to derive range rate. Accuracy and reliability of this data will depend on signal strength and closing rates.

(2) Altimetry

The rendezvous radar inherently possesses the capability to sense altitude and altitude rate. In this regard, the following factors relating to altimetry design are included:

- Pulsed modulation is desired over FM-CW. This is true due to the high operating altitude and need for unambiguous range information, which, in turn, requires a low modulating frequency. In FM-CW, a high modulation rate is desired to avoid overlapping of signal return spectral components arising from different parts of the beamwidth.
- Considering transmitter power requirements and desired range resolution, it is desirable to have the altimeter beam width limited at the operating altitude. That is, the entire ground area subtended by the antenna beam should contribute to the signal received. This, in turn, implies that the transmitted pulse duration satisfy the following inequality in the case of beam axis normal to the terrain:

$$\tau \geq \frac{2h}{c}\left[\sec\frac{\theta}{2} - 1\right]$$

where:

τ = transmitter pulse duration
h = altitude

c = propagation velocity

θ = beamwidth.

In addition, required peak transmitter power, for small incident angles and for over water and hard ground terrain, is on the order of 20 to 40 watts. Receiver integration requirements are minimal. Actually, receiver signal strength for lock-on is sufficient without any integration. However, a nominal delay may be required to achieve desired accuracy.

Requirements peculiar to the altimetry function are

- Added gimballing swing in vertical to permit main reflector to be pointed downward
- Need for moderately accurate verticality data to keep incidence angle small.

(3) Verticality

Use of the main reflector in the vertical position can be extended to sense verticality when using conical scan. This rendezvous angle track feature can be applied to modulate altitude video return at a rate equal to the scan rate. It is noted that for this application the scanner generates a time-range modulation while for angle tracking the scanner generates an amplitude modulated signal.

Sperry has conducted flight tests using this principle with the APN-59 in conjunction with the KC-135 program. Manual observation of an elliptical "doughnut" pattern on the PPI permitted calculation of vertical error within 0.1 degree of the antenna plate quotation by

measuring the eccentricity of the resulting ellipse. Figure D-1 illustrates the conical scan geometry.

Figure D 2A illustrates a video time-amplitude plot and figure D-2B a verticality functional diagram. Time (t_o) marks the center of the antenna beam return. If the scan boresight is along the vertical, t_o will not change as a function of beam scan (discounting the change as a function of altitude). If a verticality error does exist, (t_o) will shift at the scan frequency and the excursion will be a function of error and squint angle.

If the (t_o) position is now bracketed by an early and late gate, and if these gate positions are slaved to an altitude reference which does not respond to the scan rate, then this small altitude change could be detected and converted into altitude errors.

It is noted the circuitry required is basically the same as used for the rendezvous application.

Constraints on the basic radar design are primarily the following:

- The verticality scan must be within ± 10 degrees of the actual vertical to assure that attenuation of back scatter is minimal. This, therefore, requires that a gross verticality reference be available and that the squint angle must be held within this limit.
- The optimum squint angle off vertical at which the beam is scanned is a function of altitude. Longer angles reduce required receiver discrimination but also degrade signal return. The converse is true for smaller angles.

A brief analysis showed that using a squint angle of 3 degrees provides an altitude modulation sufficient to sense better than 0.2-degree verticality errors.

Accordingly, it is expected that the basic rendezvous radar will perform the verticality function subject mainly to the conditions mentioned above.

c. Major Subsystem Elements

Figure D-3 shows a basic block diagram of the OLF radar. A detailed function flow diagram showing the relationship of the OLF radar elements is shown in figure D-4. Refer to these figures during the discussion of major subsystem elements.

(1) Antenna

The antenna consists of a base section, a reflector and feed. The base is comprised of a set of two gimbals, horizontal and vertical drive motors, position synchros, and resolver pick-off. The reflector is a simple paraboloid with diameter of approximately 30 inches. This provides an antenna gain at 10 kmc of 35 db, with a beamwidth of 2.5 degrees. The feed unit must be designed for high reliability in a space borne environment. Accordingly, a scheme which uses a modulated magnetic field as the prime mover of the mechanical rotating element has attractive potential.

(2) Receiver - Transmitter

Two types of transmitters were considered. The first is a conventional type using a klystron oscillator pulsed by a modulator to deliver R.F. energy to the transmitter. A separate klystron oscillator is used as the local oscillator for the superheterodyne receiver. This configuration results in a compact package but requires automatic frequency control for both beacon and radar (altimetry and verticality) modes. Further, the center frequency must be shifted for both mode operations.

The second system employs a klystron oscillator which is mixed with a 30 megacycle oscillator to produce two side bands along with the 10 kmc center frequency. The upper side band is used to drive another klystron amplifier which is modulated to produce the transmitted signal. Taps are available to use either the upper side band or center frequency signal as the mixer input depending on the mode used (that is beacon or radar). Beacon frequency is selected such that with either mode, a 30 megacycle IF signal will result. The latter system was chosen mainly because of improved reliability over the former method.

The basic transmitter reference is slaved to a highly stable reference cavity to assure frequency stability during beacon operation.

A system pulse repetition frequency of 200 pps was chosen to avoid any range ambiguities and because it provided an adequate information rate and minimized the average power requirements of the transmitter.

The received section is a standard superheterodyne type. Mixer output is tentatively chosen at

30 megacycles. Actually, this may vary depending on results of further analysis in choosing an optimum IF frequency. The 30 megacycles is obtainable in the beacon (rendezvous) mode and in the altimetry and verticality modes (without beacon) through use of the side band selector mentioned plus proper selection of beacon frequency. Bandwidth of the IF stage is 4 megacycles to assure preservation of the 1 microsecond pulse shape for range accuracy and to allow for nominal beacon drift. It is noted that for no beacon operation (altimetry, verticality) this bandwidth can be reduced to 2 megacycles since radar transmitter drift will not affect IF frequency.

An automatic gain control loop is included to suppress signal level variations over the large dynamic range expected for the rendezvous operation.

The modulator can be a solid state unit. An input trigger is received from the master signal reference which controls the basic timing of transmission and ranging. Again, the pulse formation task should not be difficult because of the type coding and pulse width required.

(3) Ranging Unit

This unit is composed of conventional automatic tracking circuitry, an accurate range counter and clock reference, beacon delay computer, circuits to extract range rate, a search mode sweep circuit and a smoothing altimeter circuit for providing a reference during the "verticality" mode.

Regarding range tracking, video output is fed to two coincidence gates. These gates are enabled serially by two pulse generators. Output from these gates feed two summers, one sums the output to a boxcar detector in the angle track and verticality unit. The other summer subtracts the gated outputs and the difference signal feeds an integrator (this signal being equivalent to range rate). Integrator output (derived range) is applied to a comparator. The comparator also receives an accurate counted range reading. Comparator output occurs when both inputs are equal. It triggers two pulse generators which, in turn, enable the input coincidence gates.

This is a standard technique, the major consideration being accuracy of the range reference.

Beacon delay compensation is again a conventional task and its complexity depends mainly on range accuracy requirements.

During initial search, a sweep circuit is provided to strobe the entire range. Since clutter should be minimal during rendezvous search, this presents no problem.

To permit discrimination of small range perturbations at the conical scan rate during verticality, the range output is smoothed. A main requirement is that the time constant be sufficiently greater than the conical scan frequency.

(4) Angle Track and Verticality

This unit consists of an automatic angle track loop for the two axis gimballed antenna, an angle search mode drive, and a conical scan drive. The angle track circuitry is also used for automatic verticality computation.

In the angle track loop, the boxcar detector sums and stores video return amplitude modulated at the conical scan rate.

This output feeds two phase detectors which provide outputs referenced to the scan orientation. During angle track, this output drives the gimbal servos to properly position the antenna. During verticality mode, this output is fed to the vehicle attitude control system which then closes the loop through vehicle dynamics.

The angle search mode consists of an instrument servo driving a synchro. This synchro is provided with a ramp excitation. A sine and cosine pick-off is located at inputs to the gimbal drives. This will induce a steadily increasing spiral of coverage. When beacon return is detected this mode is automatically (or manually) cut off.

Regarding the conical scan drive, this potentially represents the most formidable design task. The objective is to nutate the feed using a highly reliable mechanical technique in a space environment. One attractive approach considers the use of an induction motor principle, that is, rotating a floating disk which is contained in a nonreflecting chamber.

(5) Beacon

A beacon will be mounted on each spacecraft manned or unmanned. Beacon design will reflect the same ground rules applied to the radar previously discussed. It has the simple function of sensing the radar interrogation, detecting and decoding the correct address,

coding the reply pulse train and transmitting back a signal of sufficient power to the radar.

Considerations in beacon design include, frequency selection, frequency stabilization, receiver sensitivity, required transmitter power, in addition to the fundamental considerations of high reliability and physical characteristics.

In this regard, the following is included: (Refer figure D-5).

- Beacon transmitter frequency is selected to be offset from the interrogator transmitter by 30 megacycles. This avoids potential interference problems
- Frequency stabilization is crucially important since the frequency references in both the beacon and interrogator will be independent. Widening IF bandwidth in both receivers to allow for drift is limited by desired sensitivity. Accordingly, it is required to lock both sources to extremely accurate reference cavities. System frequencies are chosen such that both cavities are designed to the same frequency. It is expected that a frequency stabilization of ± 1 megacycle for each reference can be readily attainable. This, of course, assumes that environmental requirements are held to reasonable limits.
- A high receiver sensitivity is required in keeping with the objective of low transmitting power. This necessitates a superheterodyne

receiver and minimum IF bandwidths. Use of multiple pulse interrogation code groups improves sensitivity as well as affording protection against noise or unwanted interrogation. Regarding IF bandwidth, emphasis is placed on designing rock solid frequency sources to maintain narrow bandwidths.

- Transmitter power is not a critical parameter as was the case for the interrogator transmitter. Latitude can then be taken in reducing transmitter power and requiring some interrogator receiver integration to achieve the desired signal-to-noise.

D-3. INERTIAL-RADAR MIXING FOR LUNAR SPACECRAFT TERMINAL APPROACH

The basic radar configuration discussed in subsection D-2 can also be employed as a primary range and range rate sensor aboard the lunar spacecraft during the approach and terminal phases of lunar flight. Using a selenocentric reference system and employing a vertically mounted accelerometer, the radar range can be extended out to 4000 nm. This is accomplished by drastically reducing the bandwidth of the radar, employing long pulse integration times and mixing the radar data with the inertially derived accelerometer data. The reduction in radar bandwidth is made possible by the accelerometer, whose function is to measure vehicle thrust and quickly update the radar range gates to prevent loss of the radar pulse returns.

Radar lock-on at 4000 nm is accomplished by slaving the radar range gates to either horizon tracker or manual optics derived data which is the primary mode of guidance operation at ranges beyond 4000 nm. At altitudes below 250 nm, the conical scan feature of the radar is initiated to determine the lunar vertical. In this mode of operation, the verticality function is time shared with the altimetry function. The radar derived data would be used for the lunar landing portion of the mission down to altitudes of 1500 feet, below which, the vehicle would rely on inertial data. This radar system along with the selected manual star tracker is sufficient to provide all of the quantities necessary to accomplish a safe lunar landing at a preselected landing site.

The modifications to the OLF radar configuration to achieve the previously mentioned long range characteristics are primarily as follows:

- Provision for generating low repetition rates. Basic radar uses a PRF of 200 pulses per sec. Minimal ambiguity for long ranges requires a PRF of 20 pps.
- Addition to the modulator pulse forming circuits to permit generation of long pulse widths (that is 20-30 microseconds). This can be accomplished with the same RF generator used in the basic radar. Average power limits are not materially affected since prf has conveniently been reduced.
- Intermediate frequency bandwidth must be reduced corresponding to the increase in pulse width. This simply means the addition of a high Q circuit in the IF, or equivalent.
- Range strobing circuits for search within the grossly gated range segment are defined externally.

Typical characteristics of the spacecraft radar system for a 4000 nm range are as follows:

- Basic OLF Radar (Refer to subsection D-2)

Antenna	30 inch
Wavelength	3.1 cm (10 kmc)
Gain	35 db
Beamwidth	2.5 degree
Transmitter	
Peak Power	50 watts

- Lunar Radar Additions

Receiver	
Bandwidth	50 to 100 kc
Transmitter	
Pulse width	20 to 30 usec
PRF	20 pps
Average power	0.03 watts

Assuming a conservative value for the lunar reflectivity of 0.01, the required signal-to-noise ratio at 4000 nm is -4db. Accordingly, to achieve a detection probability of 0.99 with a 1 percent false alarm rate, approximately 500 pulses must be integrated. Therefore, depending on the accuracy of the initial condition data, a minimum integration time must be allowed to achieve the desired detection probability. For example, if the range designate data is equivalent to 5 range segments, the search mode could take $\frac{5 \times 500}{20\ (PRF)}$ = 125 seconds.

For a more complete description of this long range low transmitter powered radar scheme, reference is made to a Sperry technical paper.[5]

It is noted that the radar design in the referenced paper operates to 5000 nm. The reason for the difference between this and the range cited for the subject radar is in antenna gain and beamwidth. A 48 inch inflatable antenna dish

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[5] "LASSO, Landing and Approach System for Space Vehicles," to be given in February 1962 at the Western Military Electronics Symposium.

was selected for the LASSO radar while a 30 inch dish is proposed for the spacecraft application.

An inertial-radar mixing mode is also used in the Tug/Ferry guidance system recommended in this study, as an alternate in case of failure of the beacon on the payload or other target with which the Tug is completing a rendezvous. If the beacon fails, the Tug would switch to a long-pulse, long integration time, inertial-radar mixing mode and thereby obtain radar range and range rate data to much greater distances than would be possible by brute force radar ranging without a beacon.

D-4. MANUAL CELESTIAL TRACKER

This subsection describes one concept of a celestial tracker mounted on an inertial platform wherein the electronic sensing equipment which is normally part of an automatic tracker is replaced by the human eye. This is done in order to obtain a noticeable increase in reliability. Important features of the concept are described.

- Manual overrides in case of malfunction of the telescope servo drive or angle transducer. The tracker is thereby given the reliability inherent in properly designed mechanical optical devices.
- While a number of mirrors and prisms are required to transmit the image from the telescope to the astronauts eye, these require only moderate precision of alignment.
- Wide angle coverage for locating the star in the star field is provided.

- The images of the azimuth and elevation verniers are transmitted along the same optical path as the star image, giving a simple and very accurate backup for the electrical transducers.
- The telescope is positioned by signals from the guidance computer, just as is an automatic star tracker. The operator's function is to correct any error in the computed star-position.
- The existing optical path can be utilized to transmit platform gimbal angles. The increase in complexity would be slight.

a. Description of Unit

Figures D-6 and D-7 show the optical paths. Light from a celestial target is collected by a folded optical system and imaged on an illuminated reticle. Precision alignment is required only in this portion of the complete optical system. From this point on, the only requirement is that the image of the object on the reticle be transmitted to the operator, and this operation requires only moderate precision.

A total of 11 prisms are required to fold the optical path through the gimbals and bearings of the inertial platform, much in the same manner as images are transmitted through theodolite gimbals and bearings. Figure D-6 illustrates the optical path through the tracker and platform gimbals. For clarity, not all collimating lenses have been shown in this diagram. For operator convenience, the image may be rotated to compensate for gimbal motion.

A feature of this design is the viewing, through the same optical path used for the celestial field, of the

tracker azimuth and elevation verniers. The presentation as seen by the observer is shown in figure D-8. This information can be used as a backup to the normal tracker precision transducers. The designer also has the option of introducing a non-precision transducer (for servo operation) and using the optical vernier as his prime mode of precision tracker positioning. Another alternative could be the elimination of the position servos, utilizing instead a motor gear train for manual slewing and positioning of the tracker.

Another feature of this design is the simultaneous viewing of the platform gimbal angles as illustrated in figure D-8. The only additional components to be added are, for vehicle yaw, an additional divided circle adjacent to the tracker azimuth circle, and a divided circle, light source and lens for the pitch axis. The roll axis, being accessible outside the case, would be viewed externally by a separate vernier assembly.

Where the operator must reorient the guidance system after a shut down, a wide angle field of view will be needed. This is provided by simply allowing the operator to remotely (or by hand) choose, for transmission through the optical path, the field of view from a small mirror attached to the telescope head instead of the normal field.

Although the external viewing window and the internal optics would be chosen for their resistance to radiation, it would be desirable to provide an external door or shield to protect the equipment when not in use.

b. Wide Angle Coverage

The narrow angle coverage of the celestial tracker does not allow a star search to be made by the operator. The ability to search is a basic requirement whenever the entire equipment is first turned on to perform a malfunction check. Therefore, a wide angle device is necessary for the operator to recognize and choose a star to be tracked. The previous paragraphs have indicated one manner in which this may be done by adding a second viewer next to the telescope. In case this should prove too cumbersome, the periscope normally available for the operator to see out of the vehicle can be employed. In this mode, the periscope is slaved by the computer so that its center cross-hairs are close to where the star tracker is looking. This method is not as reliable as having the wide angle devices on a platform, but it is backed-up by the ability of the operator to read the gimbal angles optimally and probably by a similar facility for the periscope angles. Here, the accuracy requirement is relatively coarse inasmuch as the star must be located only well enough to be in the narrow angle tracker's field of view.

c. Reliability

The reliability problems associated with automatic star trackers have been avoided in the subject tracker. By eliminating the detector, whether of the photomultiplier, solid state or electronic scanning types, a significant improvement in reliability is achieved. Even more important, perhaps, is the elimination of the scanning, signal processing and logic circuits associated with the detection and

error sensing processes. The only remaining areas of potential unreliability, the positioning and angular readout functions, are adequately backed up as suggested in the paragraph a. of this previous subsection. Should the position servo amplifier fail, the operator can slew the motor directly to position the tracker gear train. Should the motor fail, the operator can position the tracker gear train by hand through an access port.

d. Accuracy

The accuracy required from this tracker is of the order of 25 seconds RMS. Experience with similar human tasks in determining angles using submarine periscopes indicates that inaccuracies due to human error can be kept below three seconds of arc.

D-5. BEACON NAVIGATION

These following paragraphs describe a highly accurate means of orbit determination which makes use of a radar on a spacecraft which interrogates and measures the range to accurately-located ground beacons to determine the space vehicle's position. This system would use a chain of highly accurate bench marks being set up by the U.S. Air Force to implement an accurate aerial survey tieing strategic land masses of the world together. This type of system requires added weight and power consumption for the vehicle radar compared to the multipurpose altitude, vertical, line of sight radar described in subsection D-2, but allows an extremely accurate orbit determination in a short period of time. Individual positions can be determined to about 100 feet, if necessary.

Since the beacons are not, in general, equally distant from the vehicle, the returns occur at different times and the picture in figure D-9 is no longer accurate. The new situation is shown in one plane in figure D-10. Here, the vehicle nominal orbit (extrapolated from previous information) is shown crossed by two lines-of-position, LOP_1 at time t_1 and LOP_2 at time t_2. Without losing generality, if t_1 is assumed to be earlier, LOP_1 is translated to LOP_1' by the distance along the nominal or it and the intersection at t_2 is found to be the point X in the figure. The use of a fairly accurate nominal trajectory also helps the radar inasmuch as it allows the use of a smaller range gate and provides antenna positioning data.

c. Data Smoothing and Orbit Determination

In this system, data is obtained using a large number of points in the orbit. The data is smoothed to determine the orbit by analytically obtaining the least mean square fit of trajectories allowed by orbital mechanics. This orbit can then be used over large parts of the flight during which time no fixes can be made.

d. Location of Beacons

Since bench marks will be located along the Atlantic Missile Range, which is close to the orbital plane during part of the day, installation of compatible repeater beacons can provide enough data to uniquely determine an orbit. Installation of beacons at other bench marks in the Pacific, such as Hawaii and in the Marshall Islands will further strengthen knowledge of the orbit.

a. Geometrical Picture

The geometrical picture showing how the beacons are employed is given in figure D-9. The vehicle determines its ranges R_1 and R_2 to beacons B_1 and B_2, respectively. Thus, the vehicle is on spheres of position about B_1 and B_2 and the intersection of these spheres is a circular line-of-position. This data can be combined with data from a third beacon to arrive at one point representing the vehicle's position; or instead, can be combined with altitude data to arrive at two points representing possible positions. The latter case is shown in the figure where the two points are shown at the intersection of the circular line-of-position due to the beacons and the constant altitude spheroid containing the vehicle. One of the two points is automatically discarded based on: incompatibility with the radar antenna pointing angles, with the mechanics of orbits representing possible successive points, with known past data, or with any combination of the three.

The only time that an ambiguity arises is when the circle is nearly tangent to the spheroid. This occurs only for a short period of time and if the data here is used at all, it should be weighted very lightly in any data mixing scheme.

b. Effects of Vehicle Motion

The vehicle's motion between transmitting and receiving the data can be accounted for in the computer. However, if the time is averaged for the two events, the resulting error is negligible for this application.

e. Single Beacon Operation

If it were required, the system would work with one beacon and knowledge of altitude or, in the extreme, with just one beacon. For each case, essentially what is done is to use curve fitting procedures to determine the vehicle's orbit, however, the amount of computation required by the beacon plus altitude data is much greater than with two beacons and knowledge of altitude. The computation required by the third method (one beacon) is even greater and would be almost impossible without the use of a fairly accurate nominal as input data.

Actually, with installation of beacons at these available bench marks, there is no need for such extreme measures as enough pairs or even triplets of stations should be available to interrogate and adequately determine the orbit.

f. Radar Characteristics

Radar characteristics are basically the same as discussed in subsection D-2 for the OLF. Salient functional differences are as follows (See figure D-11).

- High power (10 kw) transmitter is required to accomodate the increased range and wider beam widths needed for ground coverage.
- Two small (6" to 8") reflectors and a microwave switch to time share the output power and beacon replies. When using the radar altimeter with these antennas, the transmitter power must also be time shared with the altimeter antenna.

Typical beam widths are estimated to be 40 degrees for maximum spread and 20 degrees for minimum spread (See figure D-12). This coverage leaves a 200 nm gap (below the vehicle) in 1000 nm, which should satisfactorily blanket most beacon locations.

- A number of ranging units equal to the number of independent range measurements.

The following is a list of pertinent design data. It is noted that this feature can be integrated into the basic rendezvous radar. Primary portions of the rendezvous radar usable with the navigation function are the receiver and transmitter driver. A combined functional block diagram showing the general tie-in is illustrated in figure D-13.

Notable design differences in addition to multi-purpose radar are:

(1) Physical Characteristics

Weight	30-pounds
Space	0.5 ft^3 (Antenna swept volume approximately 1.5 ft^3)
Power drain	40 watts

(2) Beamwidth (Each of 2 antennas)

Wide beam	40 degrees
Narrow beam	20 degrees

(3) Range

Nominally	500 nm

(4) Transmitter Power

Peak power	10 kw
Average power	2 watts

(5) Gimbal Limits

± 5° both axes.

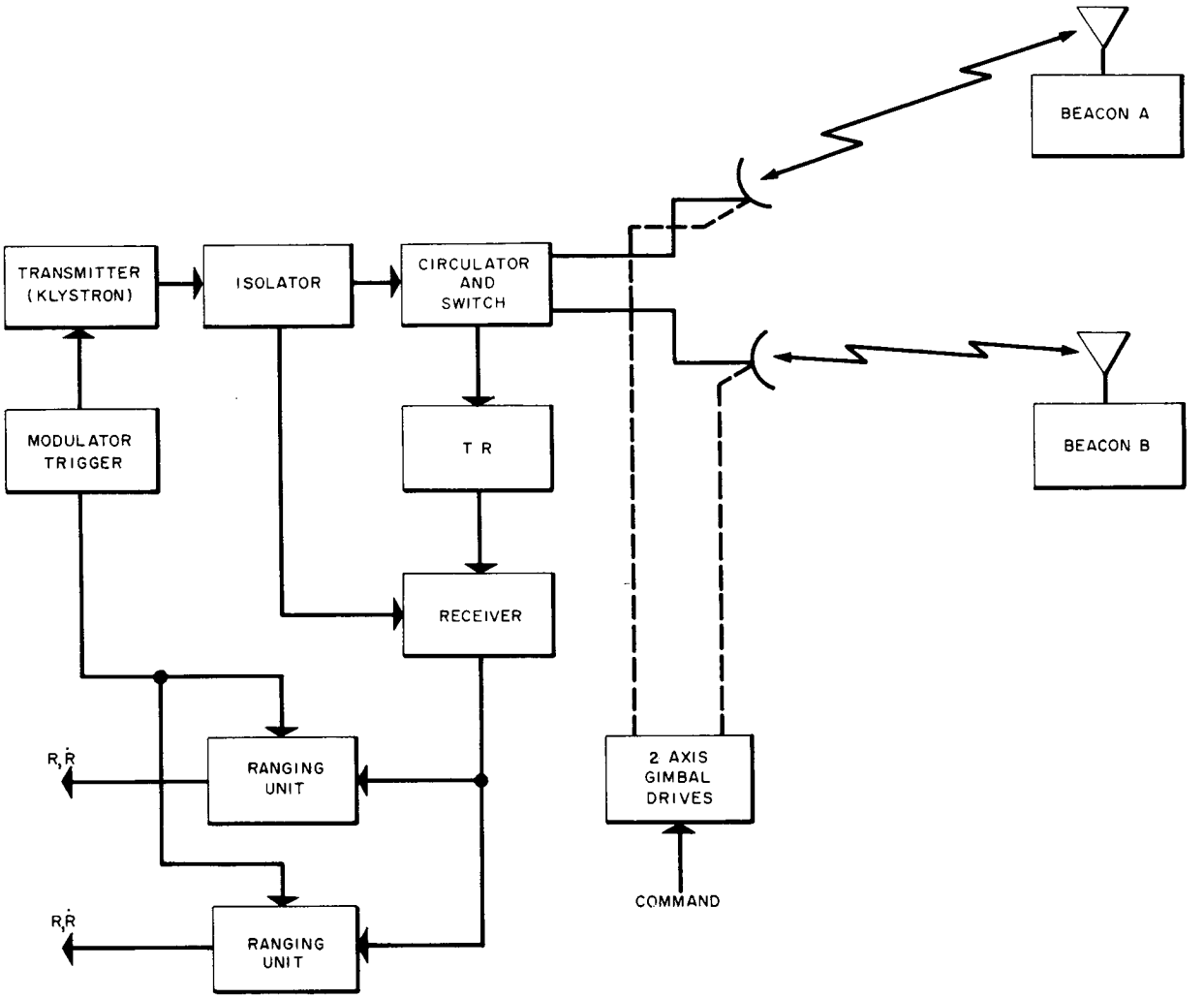


Figure D-11. OLF Navigation Radar

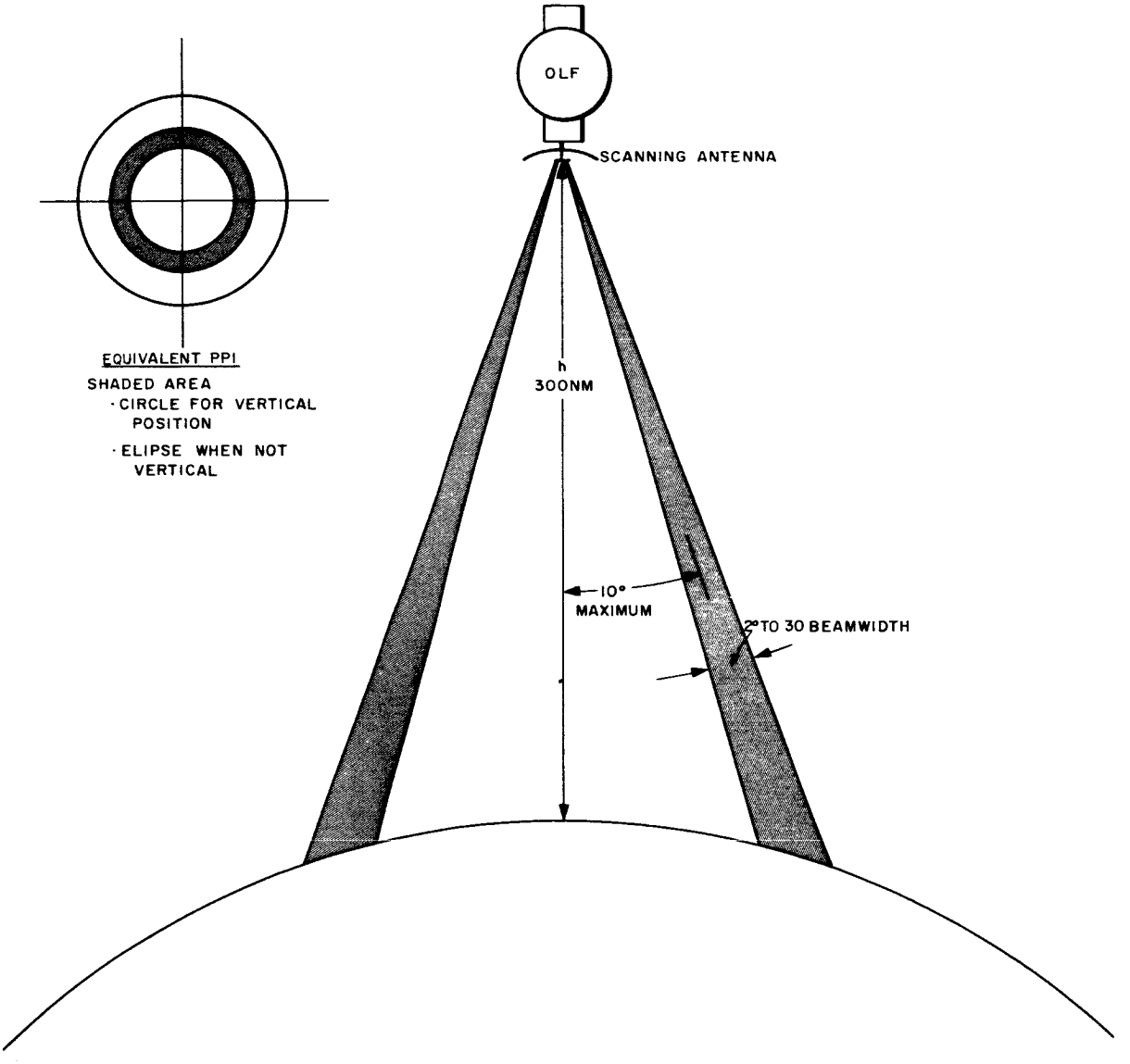
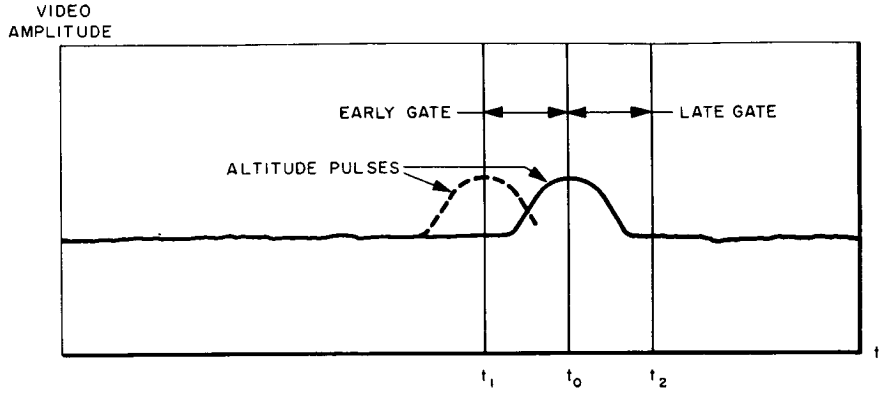
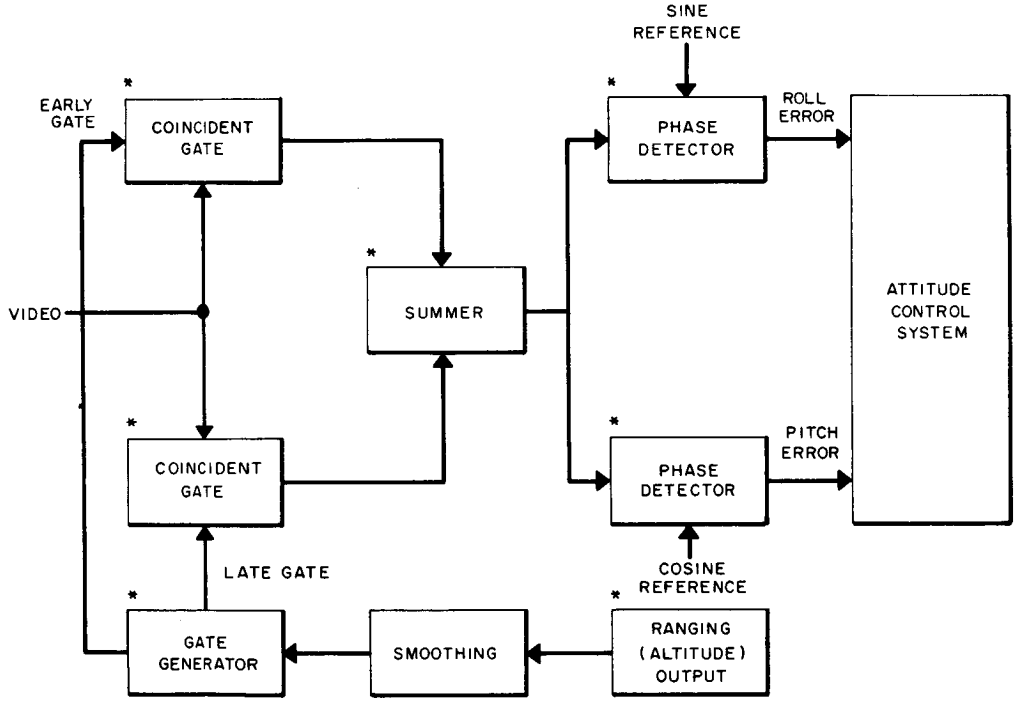


Figure D-1. Conical Scan Geometry for Verticality



A

Figure D-2A. Video Time/Amplitude Plot



B

Figure D-2B. Verticality - Functional Diagram

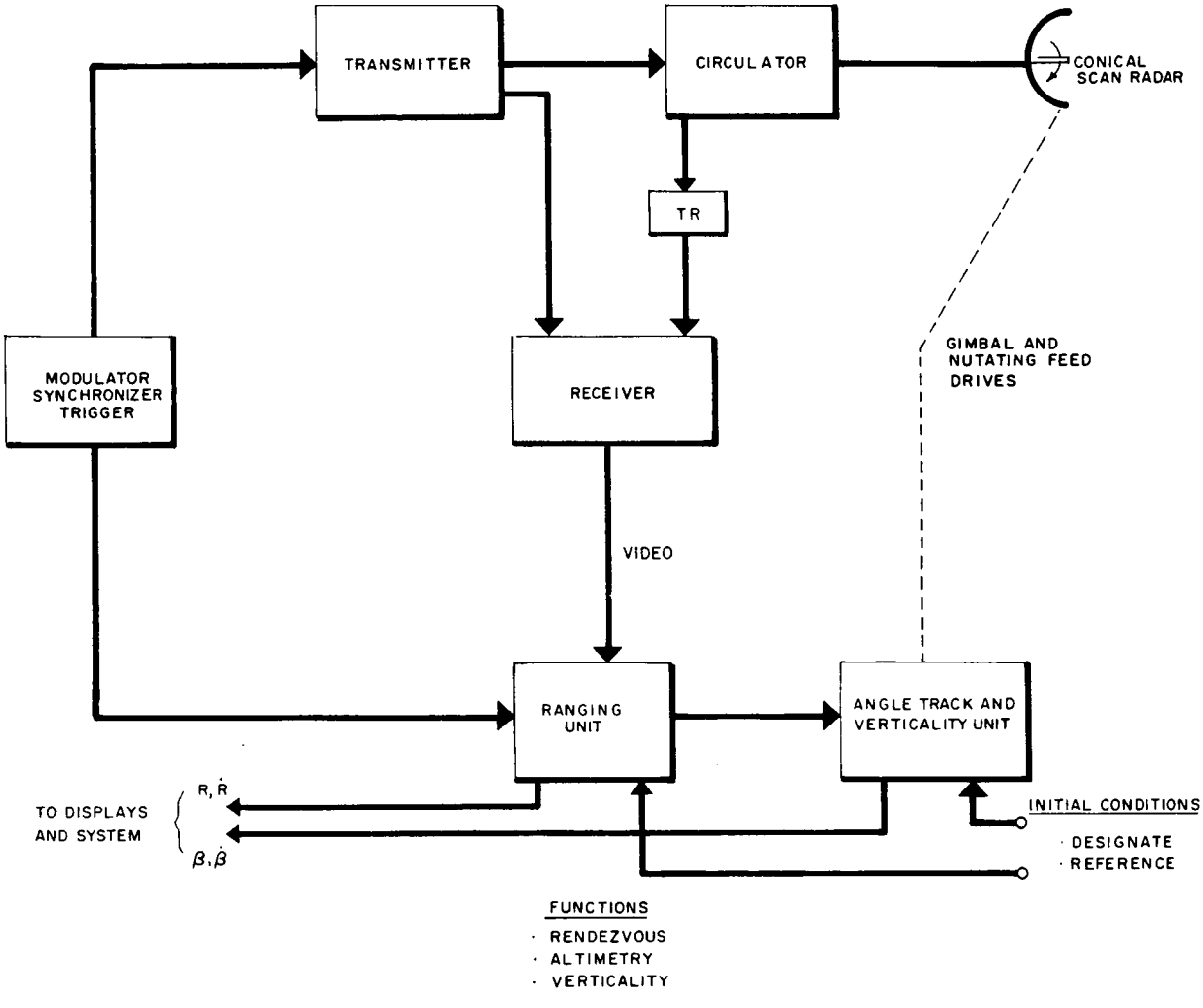


Figure D-3. OLF Radar, Basic Block Diagram

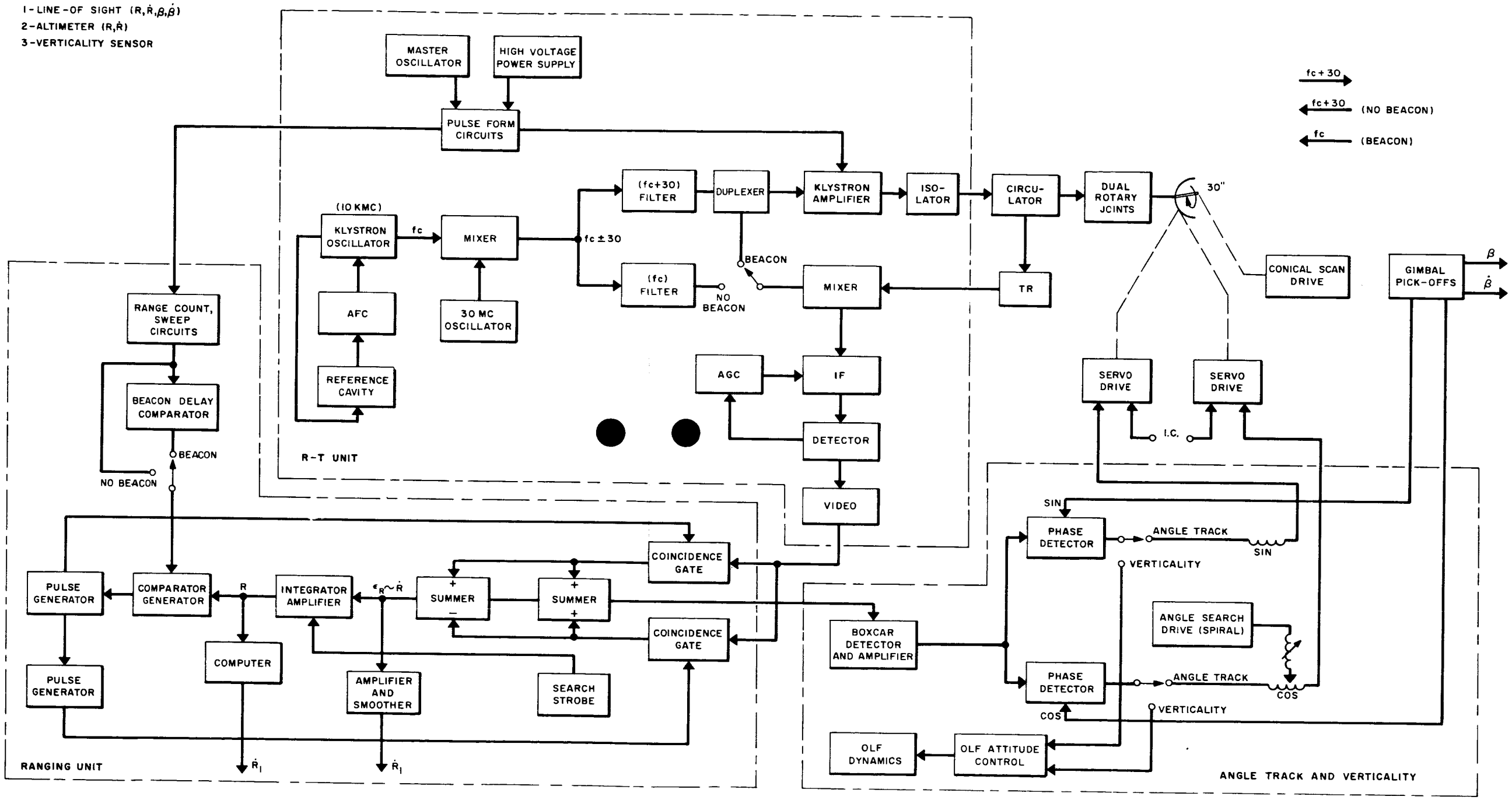


Figure D-4. OLF Radar, Function Flow Diagram

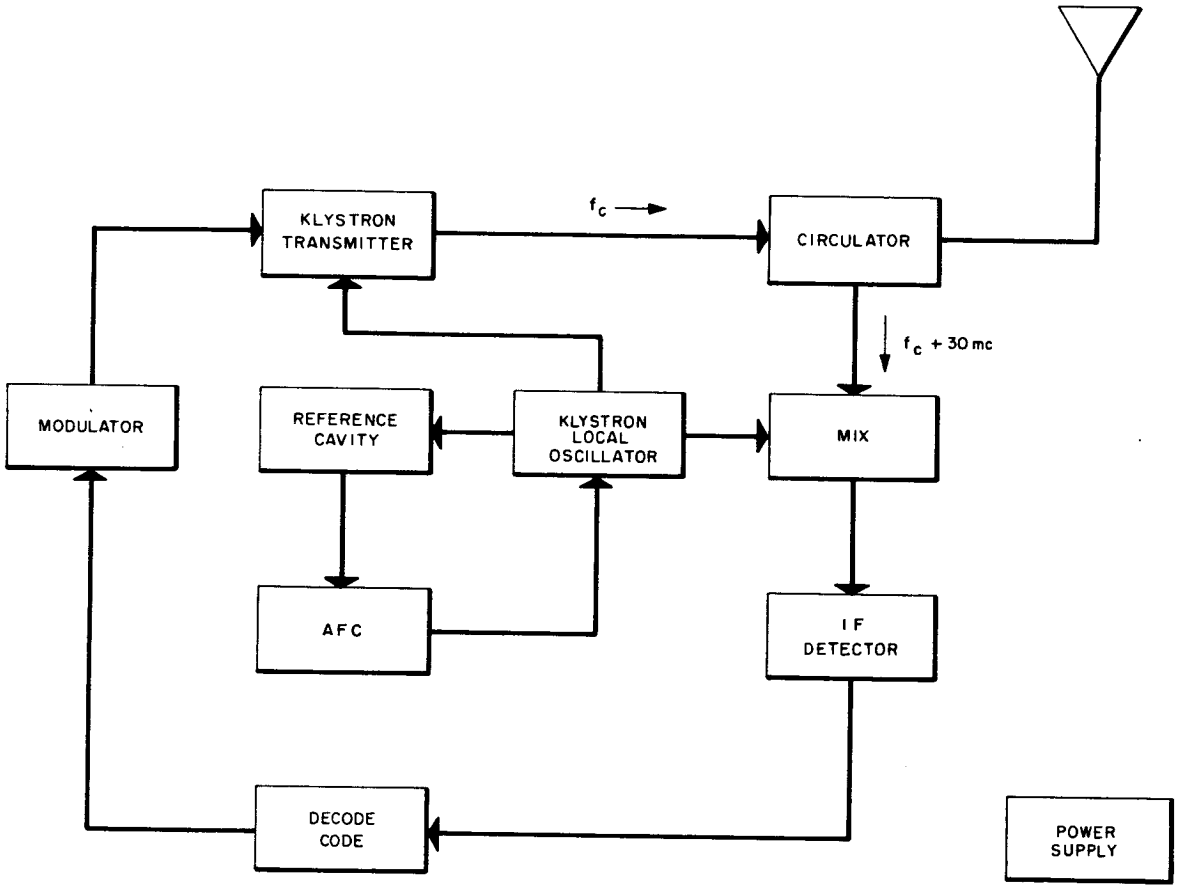


Figure D-5. Beacon, Block Diagram

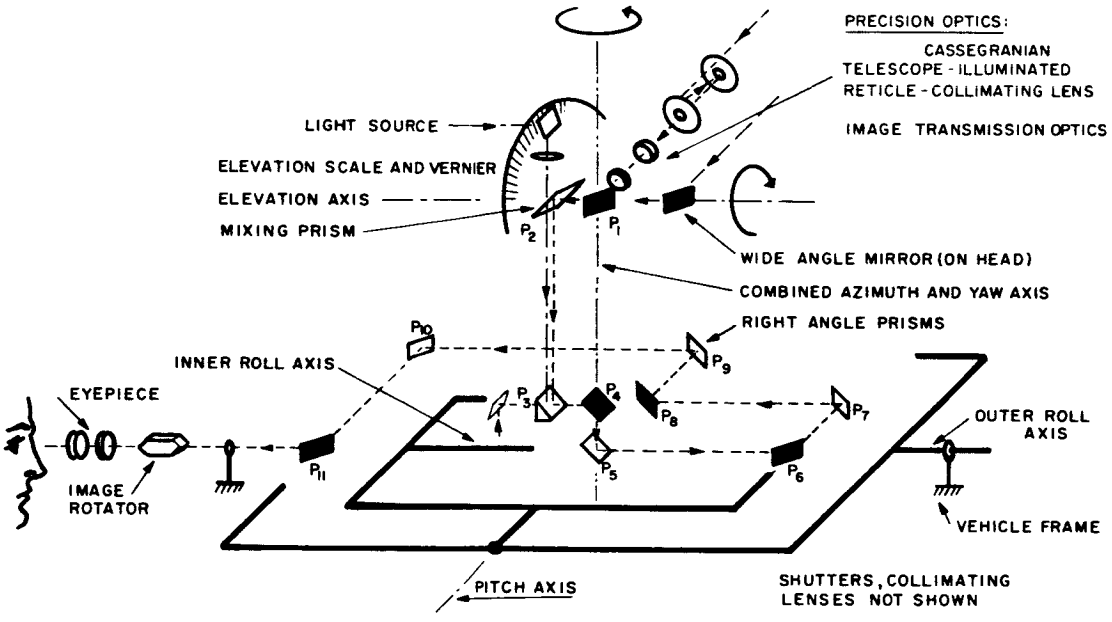


Figure D-6. Optical Paths, Manual Tracker

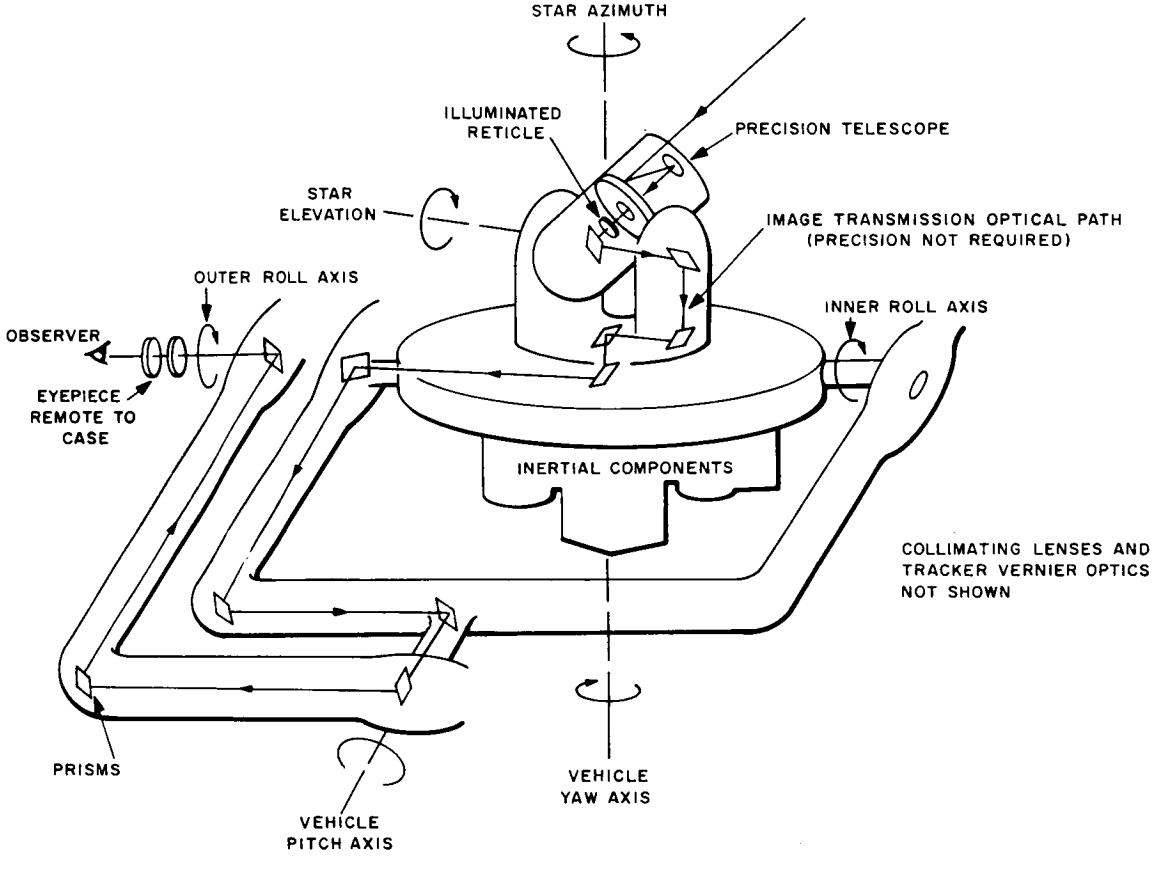


Figure D-7. Manual Tracker, Celestial Field of View Transmission

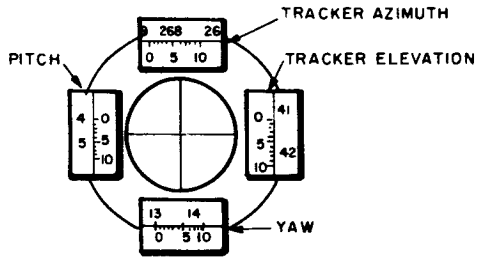


Figure D-8. Field of View of Observer

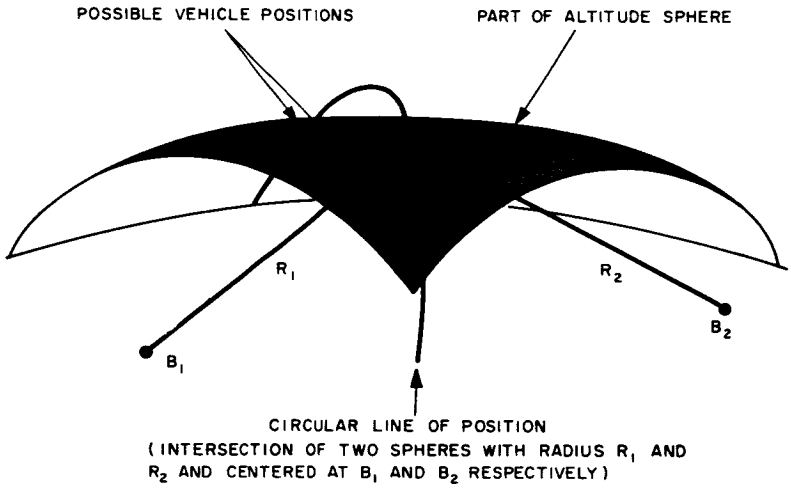


Figure D-9. Ground Beacon Geometry

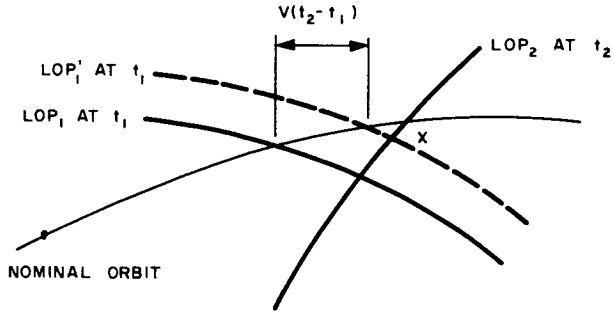
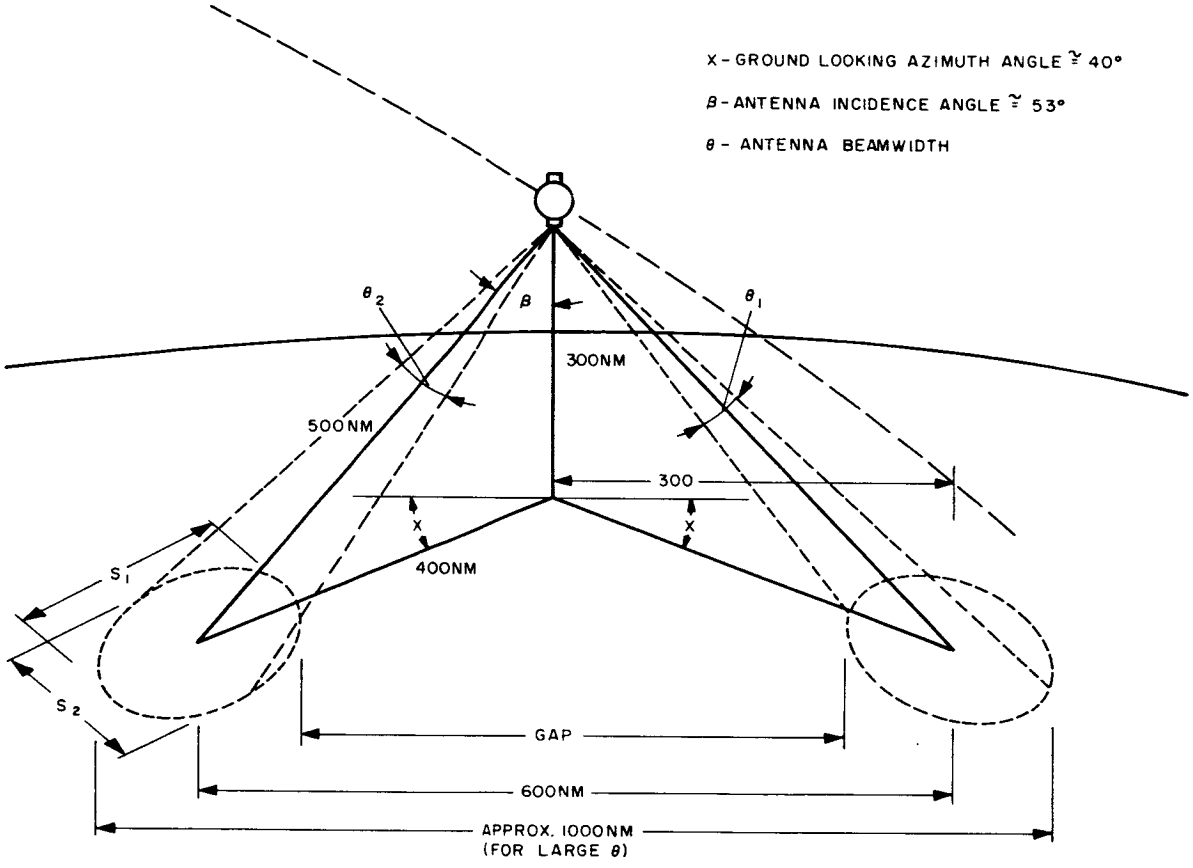


Figure D-10. Effect of Vehicle Motion



θ_1	S_1	θ_2	S_2	GAP(NM)	REQ'D. POWER (AT 500NM)	RADAR* CHARACTERISTICS	
						WT POUNDS	POWER DRAIN WATTS (AC) (DC)
10°	140	5°	70	420	200W	40	80 30
40°	400	20°	200	200	2KW	56	90 35
50°	500	25°	250	100	3KW	56	90 40

*COMBINED MULTIPURPOSE PURPOSE AND NAVIGATION CAPABILITY.

Figure D-12. Ground Beacon Ranging Geometry for Navigation

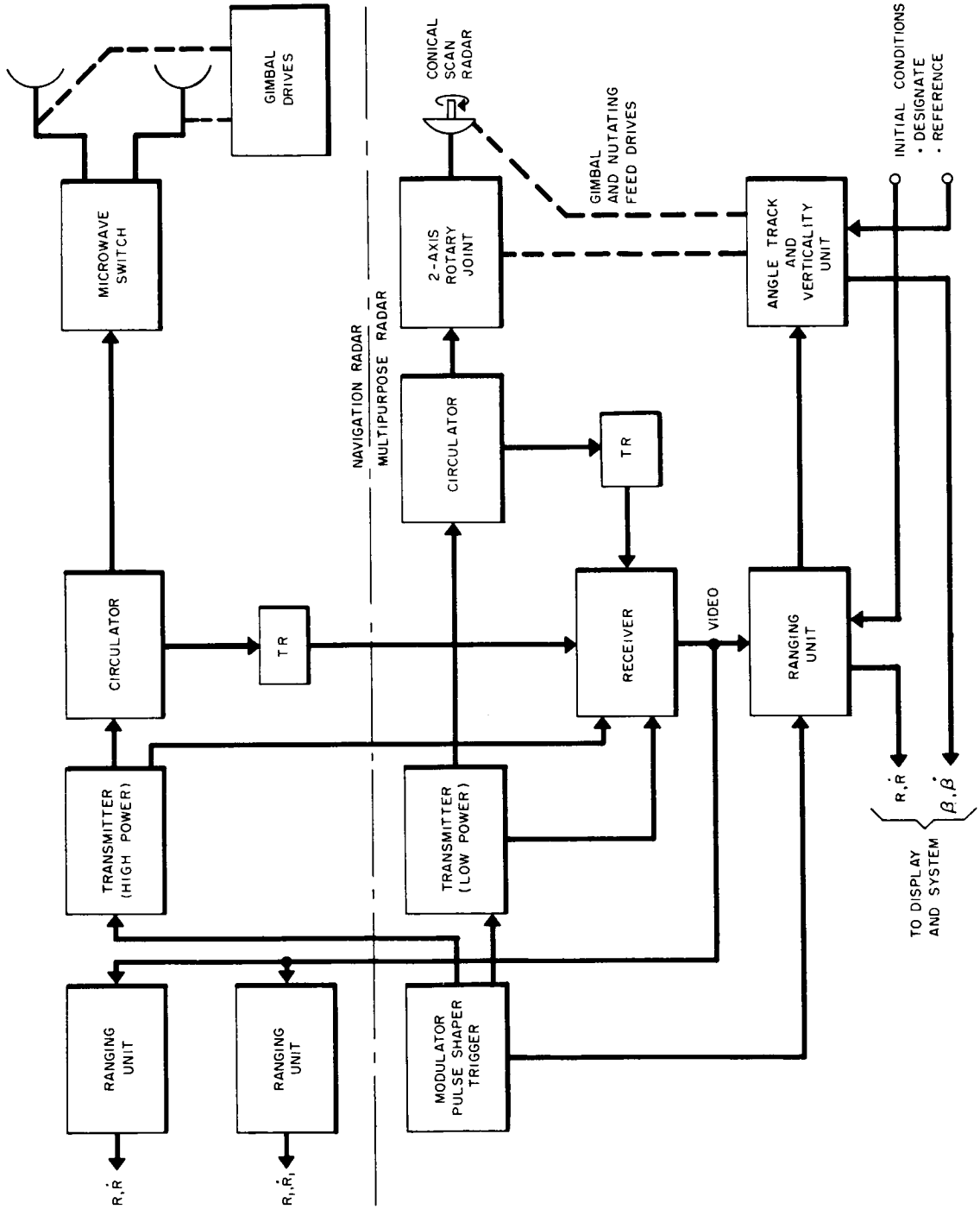


Figure D-13. Integration of Navigation Radar with Multipurpose Radar

N62-11112

ORBITAL LAUNCH OPERATIONS

VOLUME IV

ATTITUDE CONTROL

PREPARED FOR
MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

PREPARED BY: SPERRY GYROSCOPE CO., GREAT NECK, N.Y.
AIR ARMAMENT DIVISION
RESEARCH AND ADVANCED DEVELOPMENT

AIDED BY: VICKERS, INC., DETROIT, MICH.

SPERRY RAND CORPORATION
SPERRY RAND SYSTEMS GROUP
GREAT NECK, NEW YORK

PUB. NO. AB-1210-0004-4

JANUARY 1962

TABLE OF CONTENTS

Section		Page
I	SUMMARY	1-1
II	INTRODUCTION	2-1
III	INITIAL CONSIDERATIONS	3-1
	3-1. Vehicle Characteristics, Missions, and Attitude-Control Requirements	
	a. Orbital Launch Facility	3-1
	b. Unmanned Payload, with no tug	3-3
	c. Orbit-Launched Vehicle, with no tug	3-4
	d. Astrotug	3-4
	e. Unmanned Payload, with tug	3-6
	f. Orbit-Launched Vehicle, with tug	3-6
	3-2. Comparison of Alternate Orbital Launch Operation Configurations	3-7
	a. Permanent Facility/No-Permanent Facility Comparison	3-8
	b. Tug/No Tug Comparison	3-8
IV	RECOMMENDED SYSTEMS AND TRADE OFF CONSIDERATIONS	4-1
	4-1. OLO Vehicle Attitude-Control-System Summary	4-1
	4-2. System Descriptions	4-8
	a. Recommended System for Orbital Launch Facility	4-8
	b. Recommended System for Unmanned Payload Without Tug	4-12

TABLE OF CONTENTS (CONT)

Section		Page
	c. Recommended System for Manned Astrotug	4-17
	d. Recommended System for Unmanned Payload, with Tug Available	4-21
	4-3. Typical Components	4-22
	a. Sensors	4-22
	b. Torque Generators	4-25
	4-4. Reliability	4-28
V	COST AND SCHEDULE	5-1
	5-1. General	5-1
	5-2. Basis for Estimate	5-1
	5-3. Estimated Costs and Schedule	5-4
VI	PROBLEM AREAS AND RECOMMENDED PROGRAMS	6-1
	6-1. Problem Areas	6-1
	a. The Relative-Attitude-Control Problem	6-1
	b. The Coupled Tug/Payload Attitude-Control Problem	6-3
	c. Attitude-Control-System Development Problems	6-3
	6-2. Program Recommendations	6-5
	a. Relative-Attitude-Control Study	6-5
	b. Attitude-Control-System Development	6-6
	6-3. Reliability Program	6-6

LIST OF ILLUSTRATIONS

Figure		After Page
2-1	Alternate Configuration Matrix	2-2
3-1	Orbital Launch Facility Requirements	3-1
3-2	Manned Payload Requirements - No Tug	3-3
3-3	Astrotug Requirements for Payload Petrieval	3-4
3-4	Unmanned Payload Requirements with Tug	3-6
3-5	Orbital Launch Vehicle Requirements	3-6
4-1	Stabilization and Control - Pitch Axis Block Diagram	4-8
4-2	Stabilization and Control - Roll Axis Block Diagram	4-10
4-3	Stabilization and Control - Yaw Axis Block Diagram	4-11
4-4	OLF Cold-Nitrogen System	4-12
4-5	Unmanned Payload Attitude and Vernier-Velocity Control System - No Tug	4-12
4-6	OLF Equipment for Automatic Control of Unmanned Payloads	4-13
4-7	Schematic Arrangement of Lateral Accelerometers for Measuring Angular and Linear Accelerations	4-14
4-8	Minimum Attitude Control System for Manned Astrotug	4-17
4-9	Augmented Attitude and Vernier-Velocity Control System for Manned Astrotug	4-17
4-10	Tug Attitude Control Schematic	4-18
4-11	Unmanned Payload Attitude Control System - with Astrotug	4-21
4-12	Solenoid-Staged System	4-25
4-13	Storable Bipropellant System	4-26
4-14	Storable Bipropellant Reaction Control System	4-26

LIST OF TABLES

Table		Page
3-1	Fuel Requirements for Payload Retrieval, With and Without An Astrotug	3-9
4-1	Orbital Launch Facility Component Weight, Volume, and Power Estimates	4-4
4-2	Unmanned Payload (No Tug) Component Weight, Volume, and Power Estimates	4-5
4-3	Manned Astrotug (Minimum System) Component Weight, Volume, and Power Estimates	4-6
4-4	Unmanned Payload (With Tug) Component Weight, Volume, and Power Estimates	4-7
4-5	Component Failure Rates	4-29
4-6	Single-Axis Attitude-Control-System Reliability	4-29
5-1	Summary of Attitude Control Equipment Costs for All OLO Vehicles in Thousands of Dollars	5-6

SECTION I

SUMMARY

The Sperry Rand Corporation is supporting the stu of Orbital Launch Operations by investigating several Astrionics areas. The subject of this volume is Attitude Control of OLO vehicles, which covers work accomplished by the Air Armament Division of Sperry Gyroscope Company and Vickers, Inc., both Divisions of Sperry Rand Corporation

The principal conclusions of this work are:

a. The attitude-control aspects of the OLO program are within the time scale for state-of-the-art development as conceived by Sperry Rand. No major technological breakthr are necessary to successfully establish OLO operations.

b. Certain follow-on study work is absolutely necessary firmly establish plans for successful OLO operations. In th area of attitude control these are

(1) To establish the relative-attitude sensors and concepts of relative-attitude control required to operate two OLO vehicles in close proximity to one another.

(2) To complete specifications for attitude-control equipment to be used in the initial orbital-launch facility, the orbital-launch spacecraft/vehicle, and unmanned payloads.

Since the above work is well within the capabilities of the Air Armament Division of Sperry Gyroscope Compan

and Vickers, Inc., it is requested that assignment of this work to Sperry Rand for the next funding period be considered.

c. Other recommended follow-on work is as follows. These items, while not placed in the absolutely necessary category, are still considered essential for well-balanced planning, and the proper initiation of work programs required to meet the objectives of the national OLO program on schedule.

(1) To proceed with further development of storable bipropellant attitude controls.

(2) To establish the optimum attitude-control technique for use with a coupled tug/payload combination.

(3) To prepare specifications for the attitude control equipment required in the intermediate orbital-launch facility, the intermediate unmanned payload, and the astro-tug.

These programs are also recommended for Sperry Rand support in the Astronautics portion of the OLO Development Program.

d. Some of the major technical conclusions, of interest to Marshall as an aid in planning the above work, are as follows:

(1) Development of special relative-attitude sensors will be required for use during docking or joining of two vehicles. Otherwise, component requirements are well within the state-of-the-art.

(2) It appears that electrical, as well as mechanical, interconnection of the astro-tug and an incoming payload will be advantageous from the standpoint of fuel economy and equipment weight savings.

(3) When operating unmanned payloads, it appears that the attitude command function can be performed remotely from the orbiting facility. This should allow a significant reduction in the weight of payload attitude-control equipment.

SECTION II

INTRODUCTION

This report presents the results of preliminary study of attitude-control requirements for the vehicles to be used in Orbital Launch Operations (OLO). The study was conducted by the Research and Advanced Development Directorate of the Air Armament Division of Sperry Gyroscope Company with the assistance of Vickers, Inc. The objectives of the study were to synthesize candidate attitude-control systems for each OLO vehicle, to isolate the basic attitude-control problems peculiar to Orbital Launch Operations, and to recommend programs that will resolve the problems and produce equipment designs suitable for the applications.

The Orbital Launch Operation will consist of placing two or more payloads in an earth orbit, combining all or part of the individual payloads into a space vehicle, and launching this vehicle from orbit. This operation can be performed in several ways. The simplest type of Orbital Launch Operation would proceed in the following manner. Unmanned payloads would be boosted into orbit by large, non-manrated boosters and the crew would follow in a smaller, more reliable vehicle. A launch vehicle would be assembled from the several vehicles and injected into a lunar or deep-space trajectory. Nothing of importance would remain in orbit after the launch vehicle departed.

A more sophisticated procedure would be to orbit a large permanent facility, sending the facility crew up after a satisfactory orbit is assured. Several payloads, both

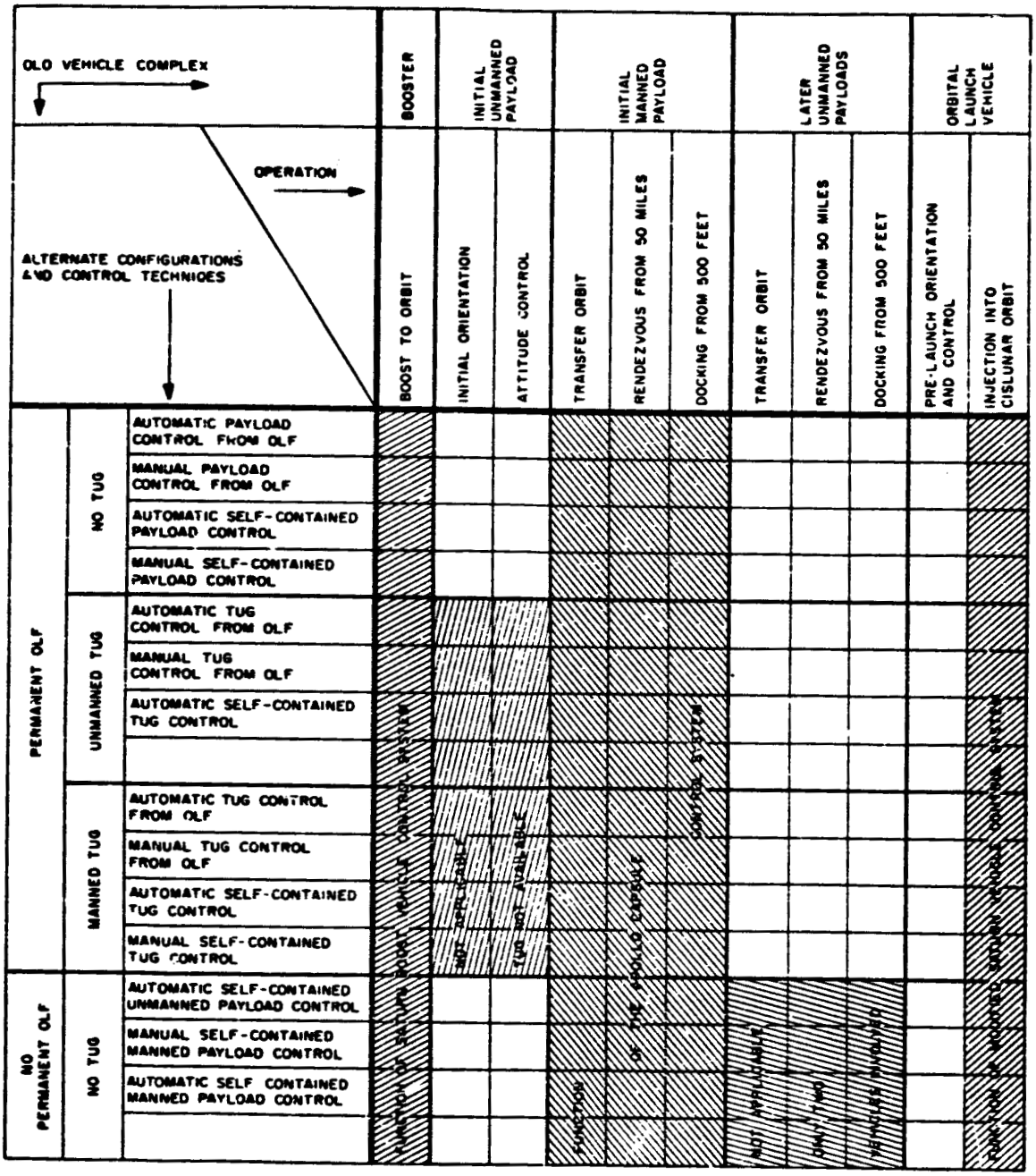


Figure 2-1. Alternate Configuration Matrix

manned and unmanned, would follow, and the facility crew would then assemble and/or fuel a launch vehicle. The permanent facility and its crew would remain on station for later operations. An astrotug might be required for payload retrieval and launch-vehicle orientation in the course of this Orbital Launch Operation.

Since the type of Orbital Launch Operation to be employed has not yet been firmly established, three basic OLO complexes must be considered: systems without a permanent orbiting facility, systems with a permanent facility and no astrotug, and systems with both a permanent facility and an astrotug. An effort was made in this study to compare the overall attitude-control requirements for these three alternate configurations, as well as to synthesize tentative attitude-control systems for the individual vehicles in each complex.

Figure 2-1 shows a matrix of the vehicles required in each of the alternate configurations. The first column of figure 2-1 summarizes the several attitude-control schemes considered for each vehicle. The control methods are manual control of remote vehicles from the permanent facility, automatic control of remote vehicles from the permanent facility, self-contained manual control of manned vehicles, and self-contained automatic control of unmanned vehicles. The applicability of each method varies, of course, with the vehicle and with the phase of operations in which the vehicle is employed.

The remaining columns of figure 2-1 list phases of the Orbital Launch Operation in which attitude control of one or more vehicles is required. As noted in the figure, boost to orbit, rendezvous by manned payloads, and cislunar

injection of a spacecraft will be performed by modified Saturn or Apollo vehicles. Attitude-control systems are already available for these vehicles, and were therefore not considered in this study. For the purposes of this discussion, Orbital Launch Operations begin when the last booster stage separates from an unmanned payload, and end when the main propulsion rocket on the orbital launch vehicle (OLV) is ignited.

The program time schedule made investigation of all control schemes in every vehicle impractical. Accordingly, the checked combinations of figure 2-1 were selected for detailed study. At least one example of every control scheme under consideration is included.

The areas investigated in synthesizing attitude-control systems for these vehicles are

- Manual and/or automatic attitude and vernier-velocity controls
- Stability augmentation
- Reaction-control torquers and vernier-velocity thrusters.

Particular attention was given to the role of the human in performing attitude control, and an effort was made to establish which functions are best suited to manual control. Detailed study of such items as manual override and authority limits for automatic systems, automatic trimming of manual systems, and synchronization of the automatic and manual systems were reserved for follow-on programs. All of these items warrant considerable further study.

In performing tradeoffs, reliability was considered of primary importance and, subject to performance

limitations, was given highest priority. Weight was also given considerable attention, particularly when equipment was to be duplicated in every payload boosted from earth. An attempt was made to centralize common equipments in the permanent facility whenever possible.

The following sections of this report summarize the study and its findings. Section III describes the vehicles, their missions, and their attitude-control requirements. Comparisons of the permanent facility/no permanent facility and tug/no tug configurations are drawn directly from this basic information. Section IV discusses the recommended control systems in detail, and summarizes alternate systems and tradeoff study results. Section V summarizes the program cost and schedule through 1966, while Section VI is devoted to discussion of the program proposed for calendar year 1962.

SECTION III

INITIAL CONSIDERATIONS

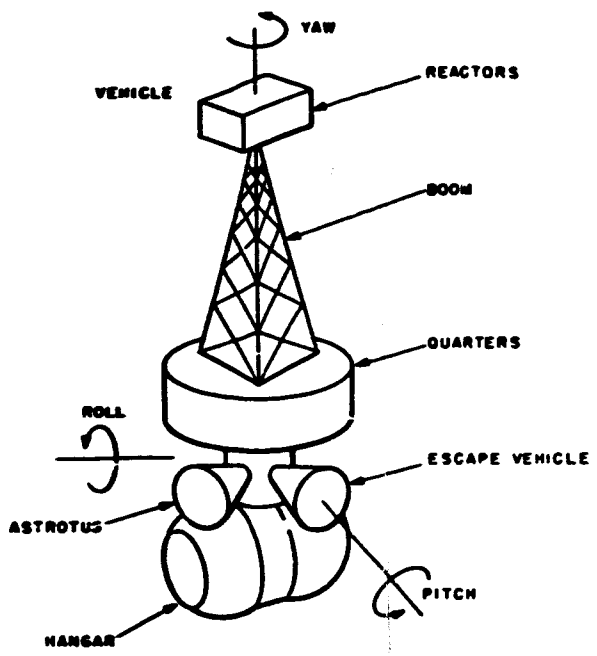
3-1. VEHICLE CHARACTERISTICS, MISSIONS, AND ATTITUDE-CONTROL REQUIREMENTS

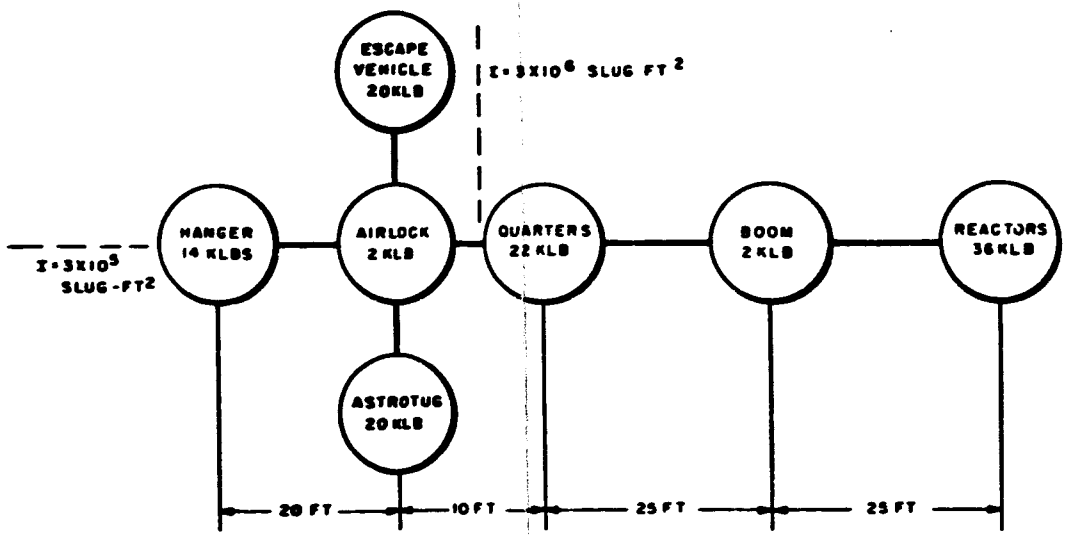
The vehicles described below are a composite of those being considered by Vought Astronautics at the time this study was initiated. The quoted performance specifications are based on documents supplied by Vought Astronautics, extractions from the literature, and data obtained from other space programs underway at Sperry. Accordingly, the vehicle and mission characteristics, and the consequent attitude-control requirements, are of a preliminary nature and will be updated as more definitive information becomes available in follow-on programs.

a. Orbital Launch Facility

A sketch of the Orbital Launch Facility (OLF) and a tabulation of its attitude-control requirements are shown in figure 3-1. The vehicle consists of a hangar, living quarters, an airlock with attached escape vehicle and Astrotug, and a nuclear reactor. The attitude-control functions include initial alignment, coarse attitude-hold, orientation for docking, and fine attitude-hold.

The OLF is considered to be boosted into orbit with its longitudinal axis approximately horizontal. The first function of the attitude-control system is to stop





ATTITUDE CONTROL

REQUIREMENT \ FUNCTION		INITIAL ALIGNMENT		ATTITUDE HOLD		ORIENTATION FOR DOCKING		DOCKING ATTITUDE HOLD	
		PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL
ATTITUDE CONTROL									
DEG.	1. POSITIONING ACCURACY	± 5°	± 5°	± 5°	± 5°	—	—	± .5	± .5
DEG./SEC^2	2. MAX. ACCELERATION	10^{-3}	10^{-2}	—	—	10^{-3}	10^{-2}	—	—
DEG./SEC.	3. THRESHOLD RATE	05	05	05	05	—	—	05	05
FT. LBS.	4. STEADY DISTURBANCES	—	—	.045	—	—	—	5	—
FT. LBS.	5. VARIABLE DISTURBANCES	—	—	045	—	—	—	045	—
FT. LBS.	6. MAX. DEMAND, STEADY	58	52	—	—	52	52	—	—
SEC (REPEATED)	7. DURATION MAX. DEMAND	270	540	—	—	270	540	—	—
FT.-LB-SEC.	8. THRESHOLD IMPULSE	$3X10^3$	$3X10^2$	$3X10^3$	$3X10^2$	—	—	$3X10^3$	$3X10^2$
SEC.	9. MINIMUM PULSE-WIDTH	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5
THRUST CONTROL									
	10 MAX FORE/AFT ΔV	—		—		—		—	
	11 MAX LATERAL ΔV	—		—		—		—	
	12 THRESHOLD ΔV	—		—		—		—	
	13 MAX. DEMAND, STEADY	—		—		—		—	
	14 DURATION MAX DEMAND	—		—		—		—	
	15 THRESHOLD IMPULSE	—		—		—		—	

INITIAL ALIGNMENT
DAMPING IRPM INITIAL TUMBLING RATE : 90 MINUTES
ROTATE 90° IN PITCH : 9 MINUTES IF NO TUMBLING

ATTITUDE HOLD
SLAVE TO LOCAL VERTICAL

ORIENTATION FOR DOCKING
ROTATION OF 360° MAX AROUND LOCAL VERTICAL
±30° DISPLACEMENT FROM LOCAL VERTICAL

DOCKING ATTITUDE HOLD
MAINTAIN SELECTED ATTITUDE FOR 20MIN DOCKING MANEUVER

Figure 3-1. Orbital Launch Facility Requirements

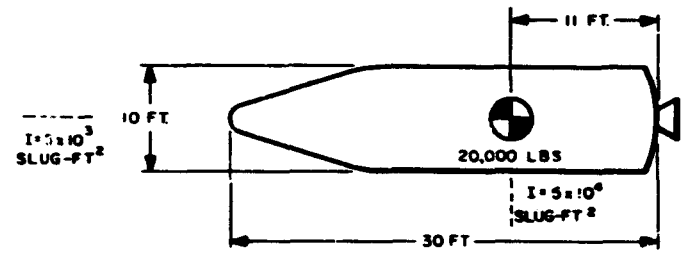
The vehicle is first oriented normal to both the line-of-sight and the inertial angular-velocity vector, after which the main propulsion unit is fired to reduce the angular velocity of the line-of-sight to zero. The vehicle is then re-oriented parallel to the line-of-sight and thrust is applied until the range rate/range schedule is satisfied. These two operations are repeated sequentially throughout the rendezvous maneuver.

The tabulated thrust levels, angular accelerations, and rendezvous maneuver time for the tug alone were extracted from the NASA studies mentioned earlier, as were the initial ranges and relative velocities. It would appear that the initial relative velocity corresponds to an orbital plane misalignment of roughly two degrees. This is quite realistic.

The maneuver time for the tug/payload combination was arbitrarily increased over that for the tug alone by an amount proportional to the payload-to-tug weight ratio. This was done to keep control forces at reasonable levels. Note that the total trip time for the tug is more than two hours. Further time extensions might cause serious operator fatigue.

The attitude angle accuracy, velocity threshold, and angular-rate threshold specified for the docking maneuver were obtained from documents supplied by Vought Astronautics. These represent the conditions required at the end of the docking maneuver. Similarly, the rendezvous-phase specifications correspond to the conditions required by Vought at the initiation of the docking maneuver.

REQUIREMENT \ FUNCTION	PAYLOAD RENDEZVOUS		PAYLOAD COUPLING		PAYLOAD RETURN (NO HELP FROM PAYLOAD)		PAYLOAD RETURN (HELP FROM PAYLOAD)		PAYLOAD DOCKING (NO HELP FROM PAYLOAD)		PAYLOAD DOCKING (HELP FROM PAYLOAD)		
	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	
ATTITUDE CONTROL													
1. POSITIONING ACCURACY	±.5	±.5	±.5	±.5	–	–	BELOW		±.5	±.5	BELOW		DEG.
2. MAXIMUM ACCELERATION	2.5	4.5	.03	.03	.15	3			.03	.03			°/SEC.²
3. THRESHOLD RATE	.05	.05	.05	.05	.05	.05			.05	.05			°/SEC.
4. STEADY DISTURBANCES	–	–	–	–	–	–			–	–			
5. VARIABLE DISTURBANCES	–	–	–	–	–	–			–	–			
6. MAXIMUM DEMAND, STEADY	2200	400	30	3	4000	800			800	80			FT. LBS.
7. DURATION, MAXIMUM DEMAND	6	6	–	–	24	24			–	–			SEC. (REPEATED)
8. THRESHOLD IMPULSE	4.5	4.5	4.5	4.5	1.3×10^3	1.3×10^2			1.3×10^3	1.3×10^2			FT-LB-SEC.
9. MINIMUM PULSE-WIDTH	0.5	0.5	0.5	0.5	0.5	0.5	NOTE		0.5	0.5	NOTE		SEC.
THRUST CONTROL													
10. MAXIMUM FORE/AFT ΔV	900		1		900		SEE		1		SEE		FT./SEC.
11. MAXIMUM LATERAL ΔV	–		1		–				1				FT./SEC.
12. THRESHOLD ΔV	1		.01		1				.01				FT./SEC.
13. MAXIMUM DEMAND, STEADY	4000		60		4000				25				LBS.
14. DURATION MAXIMUM DEMAND	140		10		550				100				SEC. (REPEATED)
15. THRESHOLD IMPULSE	600		6		2.5×10^3				25				LB-SEC.



MAIN ROCKET
TWO THRUST LEVELS
(4000 AND 2000 LBS)
NON-STEERABLE
RE-STARTABLE

PAYLOAD RENDEZVOUS

TRAVEL TO PAYLOAD, 50 MILES MAXIMUM RANGE (NOMINAL 10 MILES AT ALTITUDE, 3 MILES LATERALLY, 3 MILES ALTITUDE ERROR) INITIAL VELOCITY 900FT/SEC MAX. (450 FT/SEC NOMINAL) MANEUVER TIME 20 MIN.

PAYLOAD COUPLING

INITIAL RANGE 500 FT INITIAL VELOCITY 1 FT/SEC.; TIME 20 MIN

PAYLOAD RETURN

AS FOR PAYLOAD RENDEZVOUS, EXCEPT MANEUVER TIME 60 MIN.

PAYLOAD DOCKING

AS FOR PAYLOAD COUPLING

NOTE: MASS AND MOMENT OF INERTIA OF TUG NEGLECTED WITH RESPECT TO PAYLOAD

NOTE: PAYLOAD MAY AUGMENT TUG IN ATTITUDE AND/OR THRUST CONTROL IF SO TUG WILL NOT PROVIDE ALL CONTROL THRUST AND/OR CONTROL TORQUES

NOTE: FOR INITIAL RENDEZVOUS WITH OLF SEE "MANNED PAYLOAD REQUIREMENTS"

NOTE: FOR OLV POSITIONING, SEE "OLV REQUIREMENTS"

Figure 3-3. Astrotug Requirements for Payload Retrieval

ATTITUDE CONTROL

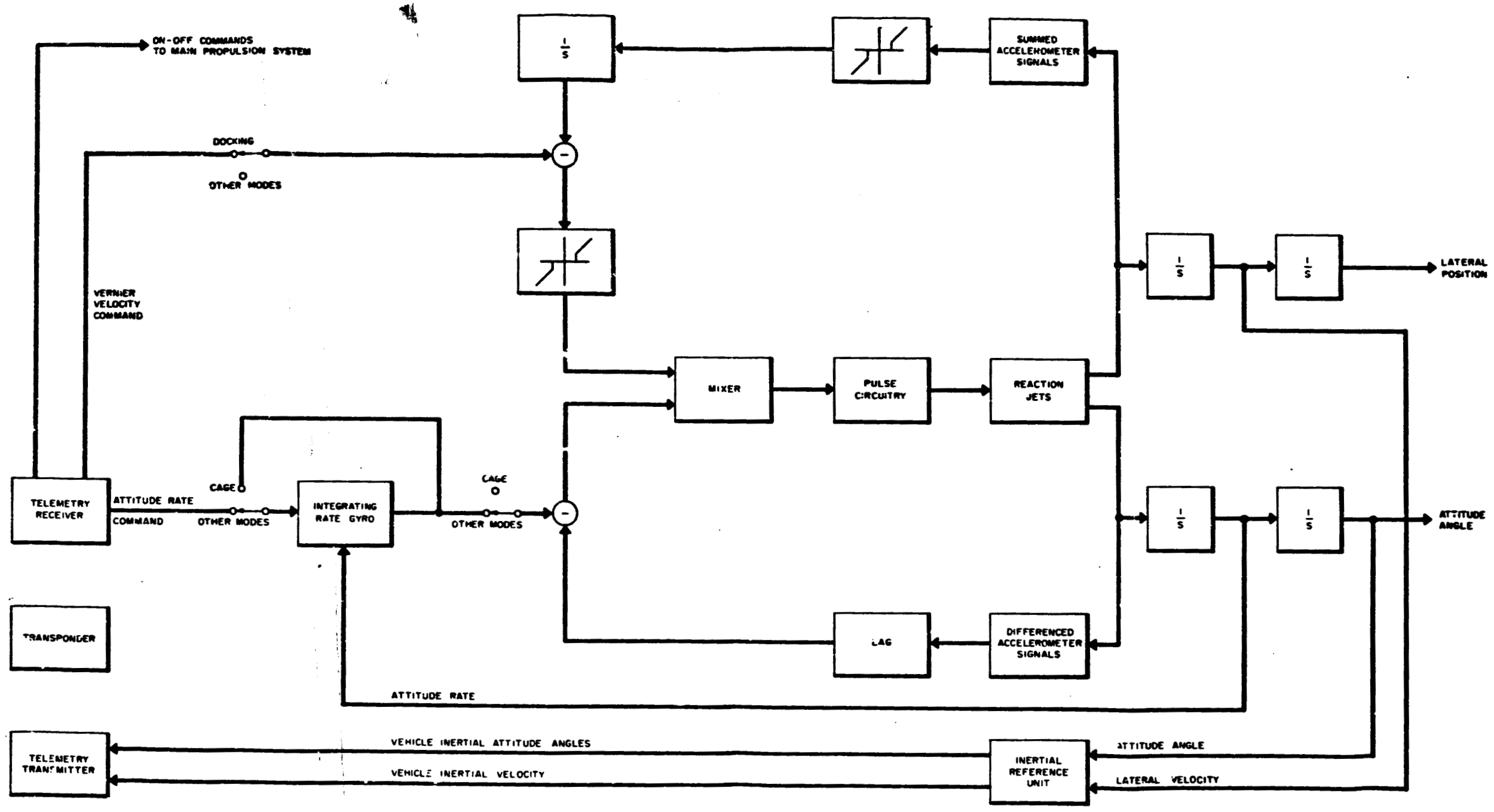


Figure 4-5. Unmanned Payload Attitude and Vernier-Velocity Control System - No Tug

For example, if the torque output is M and the minimum pulse width is T_{min}, then

$MT_{min} \leq$ threshold-impulse specification

$T_{min} \leq$ minimum-pulse-width specification

The specified threshold impulse will deliver the specified threshold rate, while the specified minimum pulse width assures a modulation frequency high enough to have a negligible effect on the dynamic response of the vehicle. Given a value of T_{min} satisfying the second inequality, the first inequality establishes an upper limit on the maximum control torque. A limited trade-off between maximum torque and duration of maximum demand is still possible. It should be noted that the maximum torque output will be required for maximum duration a number of times during any flight, and does not establish an upper limit on fuel requirements.

b. Unmanned Payload, with no tug

The attitude-control requirements for the unmanned payload are summarized in figure 3-2 for the no-tug situation. As shown in the figure, the unmanned payload was assumed to consist of an 30,000-pound cylinder with a non-steerable, re-startable main propulsion unit.

When the vehicle is placed in orbit it may be a considerable distance from the desired rendezvous point, and transfer orbits will probably be used to reduce the initial separation. It was not possible to establish attitude-control requirements for this phase of flight, because neither the initial conditions nor the guidance technique have been defined.

After reaching a point within 50 miles of the permanent facility, the rendezvous flight mode is initiated. Docking maneuvers begin at a range of 500 feet. For simplicity, the astrotug rendezvous technique was also assumed for the unmanned payload. The details of this scheme are discussed below in paragraph d.

Most of the attitude-control-system requirements tabulated in figure 3-2 were obtained in a manner similar to those for the OLF. The exceptions are thrust-level and angular-acceleration requirements, which are the same as those specified for the astrotug/payload combination.

c. Orbit-Launched Vehicle, with no tug

Requirements for the space vehicle will be similar in many respects to those of the unmanned payload. Detailed discussion is reserved for paragraph e.

d. Astrotug

A sketch of the astrotug is shown in figure 3-3 along with a table of attitude-control requirements for payload retrieval. The tug must rendezvous with the incoming payload, couple itself to the payload, return to the permanent facility, and dock the payload.

The rendezvous technique selected for purposes of the attitude-control-system investigation was the simplest possible, and is based on the results of NASA simulator studies. A collision course is established with the objective by reducing the inertial rate of rotation of the line-of-sight to zero. Braking thrust is then applied along the line-of-sight to maintain a precomputed schedule of range rate vs. range.

REQUIREMENT \ FUNCTION	RENDEZVOUS		DOCKING		
	PITCH YAW	ROLL	PITCH YAW	ROLL	
ATTITUDE CONTROL					
1 POSITIONING ACCURACY	±5°	±5°	±5°	±5°	DEG
2 MAXIMUM ACCELERATION	15	.30	.03	03	DEG / SEC^2
3 THRESHOLD RATE	05	05	05	05	DEG. / SEC.
4 STEADY DISTURBANCES	–	–	–	–	
5 VARIABLE DISTURBANCES	–	–	–	–	
6 MAXIMUM DEMAND, STEADY	4×10^3	800	800	80	FT.-LBS
7 DURATION MAXIMUM DEMAND	24	24	–	–	SEC. (REPEATED)
8 THRESHOLD IMPULSE	1.3×10^3	1.3×10^2	1.3×10^3	1.3×10^2	FT.-LB.-SEC.
9 MINIMUM PULSE-WIDTH	.5	5	.5	5	SEC.
THRUST CONTROL					
10 MAXIMUM FORE / AFT ΔV	900		1.0		FT / SEC
11 MAXIMUM LATERAL ΔV	—		1.0		FT / SEC.
12 THRESHOLD ΔV	1.0		.01		FT. / SEC.
13 MAXIMUM DEMAND, STEADY	4000		25		LBS.
14 DURATION MAXIMUM DEMAND	560		100		SEC. (REPEATED)
15. THRESHOLD IMPULSE	2.5×10^3		25		LB.-SEC.

ORBIT TRANSFER

REQUIREMENTS NOT ESTABLISHED

RENDEZVOUS

INITIAL RANGE 50 MILES MAX (NOMINAL 10 MILES AT ALTITUDE 3 MILES LATERALLY, 3 MILES ALTITUDE ERROR)
INITIAL VELOCITY 900 FT./SEC. MAX (NOMINAL 450 FT./SEC.)
MANEUVER TIME 80 MIN.

DOCKING

INITIAL RANGE 500 FT.
INITIAL VELOCITY ±1 FPS
MANEUVER TIME 20 MIN.

VEHICLE

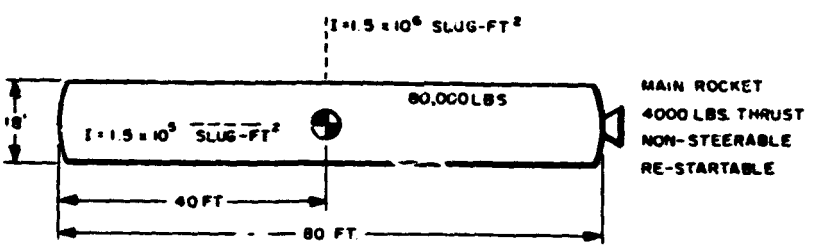


Figure 3-2. Manned Payload Requirements - No Tug

e. Unmanned Payload, with tug

The attitude-control requirements for the unmanned payload are summarized in figure 3-4 for the case when a tug is available. In this case, the payload attitude-control system is required to damp the initial tumbling and, upon command from the tug, hold steady while the tug attaches itself.

Depending upon the capability of the tug, the payload may have to provide all or part of the torque and/or thrust required for the return trip and for docking. The question of whether or not to couple the payload reaction jets into the tug control system after the vehicles are joined has not been resolved.

f. Orbit-Launched Vehicle, with tug

A possible Orbit Launched Vehicle (OLV) configuration is shown in figure 3-[illegible]. It is constructed of a modified Saturn S-IV, two Agena B's, and an Apollo capsule. Prior to orbital launch, an attitude-control system is required to stabilize the vehicle in order to avoid hitting the permanent facility, to control the vehicle while it taxis to the launch site, and to orient the vehicle for launch.

Taxi to the launch site was considered to begin with an initial small velocity impulse followed by a long coasting phase. The vehicle will be reoriented while coasting so that a retro-thrust can be applied when the launch position is reached. The OLV will then be slewed to the proper launch direction as computed and commanded by the OLF.

FUNCTION / REQUIREMENT	INITIAL ALIGNMENT		COUPLING ATTITUDE HOLD		TUG AUGMENTATION ATTITUDE CONTROL		TUG AUGMENTATION THRUST CONTROL		
	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	
ATTITUDE CONTROL									
1 POSITIONING ACCURACY	$\pm 5^{\circ}$	$\pm 5^{\circ}$	$\pm .5$	$\pm .5$	—	—	—	—	DEG.
2. MAX. ACCELERATION	10^{-3}	10^{-2}	—	—	.15	.3	—	—	DEG./SEC.^{2}
3. THRESHOLD RATE	.05	.05	.05	.05	.05	.05	—	—	DEG./SEC.
4. STEADY DISTURBANCES	2.5	—	2.5	—	—	—	—	—	FT.-LBS.
5. VARIABLE DISTURBANCES	.045	—	.045	—	—	—	—	—	FT.-LBS.
6. MAX. DEMAND, STEADY	26	26	—	—	4000	800	—	—	FT.-LBS.
7. DURATION MAX. DEMAND	270	540	—	—	24	24	—	—	SEC.(REPEATED)
8. THRESHOLD IMPULSE	1.3×10^{3}	1.3×10^{2}	1.3×10^{3}	1.3×10^{2}	1.3×10^{3}	1.3×10^{2}	—	—	FT.-LB.-SEC.
9. MINIMUM PULSE-WIDTH	0.5	0.5	0.5	0.5	0.5	0.5	—	—	SEC.
THRUST CONTROL									
10. MAX. FORE/AFT ΔV	—		—		—		900		FT./SEC.
11. MAX. LATERAL ΔV	—		—		—		1		FT./SEC.
12. THRESHOLD ΔV	—		—		—		.01		FT./SEC.
13 MAX. DEMAND, STEADY	—		—		—		4000		LBS.
14. DURATION MAX DEMAND	—		—		—		550		SEC.(REPEATED)
15. THRESHOLD IMPULSE	—		—		—		25		LB.-SEC.

INITIAL ALIGNMENT

DAMPING 1RPM INITIAL ROTATION IN ROLL
MANEUVER TIME 90 MINUTES

COUPLING ATTITUDE HOLD

HOLD SELECTED ATTITUDE FOR 20 MINUTE MANEUVER

TUG AUGMENTATION (ATTITUDE CONTROL)

ACCEPT COMMANDS FROM TUG AND PROVIDE CONTROL TORQUES

TUG AUGMENTATION (THRUST CONTROL)

ACCEPT COMMANDS FROM TUG AND PROVIDE THRUST

NOTE:

TUG DESIGN WILL SPECIFY WHICH AUGMENTING FUNCTIONS ARE REQUIRED, IF ANY

NOTE:

SEE "UNMANNED PAYLOAD REQUIREMENTS" FOR RENDEZVOUS-DOCKING BREAKDOWN

VEHICLE:

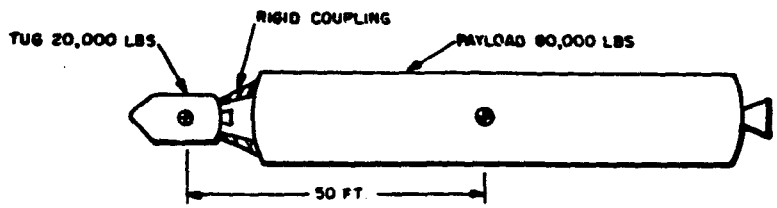


Figure 3-4. Unmanned Payload Requirements with Tug

REQUIREMENT \ FUNCTION	ATTITUDE HOLD NEAR OLF		TAXI TO LAUNCH SITE (NO TUG)		LAUNCH ATTITUDE AND POSITION HOLD		
	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	
ATTITUDE CONTROL							
1. POSITIONING ACCURACY	$\pm 2°$	$\pm 2°$	$\pm 5°$	$\pm 5°$	$\pm 2°$	$\pm 2°$	DEG
2. MAX ACCELERATION	—	—	2×10^{-4}	2×10^{-4}	2×10^{-4}	2×10^{-4}	DEG/SEC^2
3. THRESHOLD RATE	.02	.02	.05	.05	.02	.02	DEG/SEC
4. STEADY DISTURBANCES	33	—	—	—	33	—	FT LB
5. VARIABLE DISTURBANCES	—	—	—	—	—	—	
6. MAX DEMAND, STEADY	—	—	20	2	20	2	FT-LBS
7. DURATION MAX DEMAND	—	—	900	900	900	900	SEC(REPEATED)
8. THRESHOLD IMPULSE	2×10^{3}	2×10^{2}	5×10^{3}	5×10^{2}	2×10^{3}	2×10^{2}	FT-LB-SEC
9. MINIMUM PULSEWIDTH	0.5	0.5	0.5	0.5	0.5	0.5	SEC
THRUST CONTROL							
10. MAX FORE/AFT ΔV	—		4.4		1.0		FT/SEC
11. MAX LATERAL ΔV	—		1.0		1.0		FT/SEC
12. THRESHOLD ΔV	—		1.0		1.0		FT/SEC
13. MAX DEMAND, IMPULSE	—		2×10^{4}		4.7×10^{3}		LB-SEC
14. DURATION MAX DEMAND	—		—		—		
15. THRESHOLD IMPULSE	—		4.7×10^{3}		4.7×10^{3}		LB-SEC

ATTITUDE HOLD NEAR OLF

SLAVE TO OLF ATTITUDE (LOCAL VERTICAL)

TAXI TO LAUNCH SITE

TAXI 3 MILES IN ONE HOUR; HOLD ATTITUDE AND POSITION AS COMMANDED FROM OLF

NOTE.

IF TUG IS USED WITHOUT TUG/OLV CONTROL SYSTEM COUPLING, THE OLV HAS NO TAXI REQUIREMENTS IF OLV ATTITUDE CONTROL SYSTEM IS CONTROLLED FROM TUG, THE ATTITUDE CONTROL FUNCTIONS LISTED ABOVE STILL APPLY BUT THE TUG WILL PROVIDE THRUST FOR TAXI.

THE TUG CANNOT ASSIST THE OLV IN LAUNCH ATTITUDE AND POSITION HOLD.

NOTE.

INJECTION, MIDCOURSE AND LUNAR ATTITUDE AND THRUST VECTOR CONTROL PROVIDED BY SATURN S-IV, AGENA-B, AND APOLLO CONTROL SYSTEM.

VEHICLE.

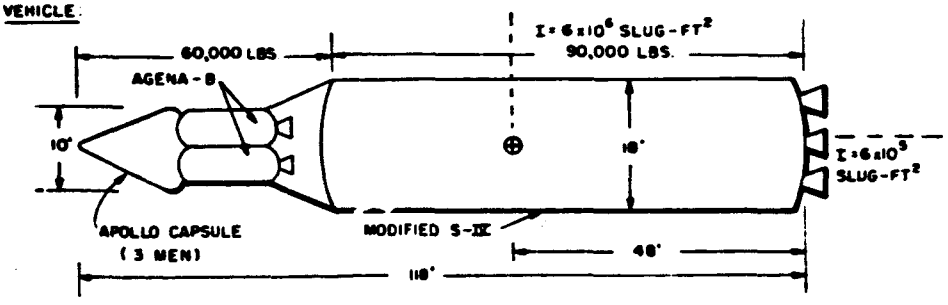


Figure 3-5. Orbital Launch Vehicle Requirements

It does not appear feasible to launch directly from the permanent facility because of the danger to the facility crew and because both vehicles are of roughly the same size. Any impacts may seriously perturb the OLF orbit (and structure). For similar reasons, the tug cannot assist the OLV in the final phases of prelaunch attitude hold.

The Saturn S-IV is presently expected to have a restart capability. In order to provide this capability, a coast attitude-control system will be added so that adequate attitude control will be available when the main engines are off. Such a system has been sized by NASA, utilizing gaseous H_2O_2 as the propellant and swiveling reaction nozzles as torque generators. If available, this Saturn modification should provide sufficient prelaunch attitude control and make an OLO-supplied piggyback unit unnecessary.

3-2. COMPARISON OF ALTERNATE ORBITAL LAUNCH OPERATION CONFIGURATIONS

The information assembled in the preceding section can be used to develop a comparison of the permanent-orbiting-facility/no-permanent-facility and tug/no-tug complexes mentioned earlier. The results of these comparisons, discussed further below, can be summarized very briefly; any attitude-control equipment trade-offs seem relatively unimportant when compared to the major operational variations implied by the alternate configurations.

a. Permanent-Facility/No-Permanent-Facility Comparison

The major difference between the permanent-facility and no-permanent-facility configurations from the standpoint of attitude control appears to be one of equipment duplication in successive operations. If a permanent facility is available, any special equipment required for rendezvous can be reused. If no permanent facility is available, this equipment will presumably be ejected as waste before the space vehicle is injected into its cislunar trajectory. The ejected equipment will be lost and booster fuel will have to be expended in replacing it for later operations. It does appear, incidentally, that special relative-attitude sensors will be required to join any two vehicles.

There appears to be no other major difference between the permanent-facility and no-permanent-facility configurations from the standpoint of attitude control. The former requires another vehicle and thus another attitude-control system, and some minor changes may be made in the vehicle attitude-control systems if a permanent facility is available to allow remote control of auxiliary vehicles. However, the most pressing OLO attitude-control problem - the control of the relative attitude of two vehicles in close proximity to one another - must be solved in either case.

b. Tug/No Tug Comparison

Attitude-control requirements are somewhat sensitive to the presence or absence of an astrotug. For example, rough calculations of the fuel required for torque and thrust, with and without a tug, lead to the tabulation of table 3-1.

TABLE 3-1

FUEL REQUIREMENTS FOR PAYLOAD RETRIEVAL, WITH AND WITHOUT AN ASTROTUG

Phase of Payload Retrieval Operation	Thrust Requirement	Lbs fuel req'd for thrust		Attitude Control required	Lbs. fuel req'd for torque (I_{sp} = 200)		
		Without tug	With tug		Without tug	With tug coupled electrically	With tug not coupled electrically
1. Rendezvous with Payload	ΔV = 900fps (I_{sp} = 300)	0	1870	Three max. demand, max. duration maneuvers	0	20	20
2. Coupling to Payload	Three max. demand, max. duration maneuvers (I_{sp} = 300)	0	15	Three max. demand, max. duration maneuvers	0	negligible	negligible
3. Payload Return	ΔV = 900fps (I_{sp} = 300)	7,330	9200	Three max. demand, max. duration maneuvers	140	280	280
4. Payload Docking	Three max. demand, max. duration maneuvers (I_{sp} = 200)	38	45	Three max. demand, max. duration maneuvers	negligible	negligible	93
5. Totals	-	7,368	11,130	-	140	300	393

The rather striking differences in fuel requirements are due to the following factors:

- The 20,000-pound tug must be taken to the payload and returned to the facility
- The tug, positioned 50 feet off the tug/payload CG, increases the vehicle mass by 25 percent and doubles the moment of inertia
- If the tug cannot operate the payload reaction jets (i.e., there is no electrical coupling) then a lateral vernier thrust applied by the tug must be accompanied by a large compensating torque to maintain a fixed attitude during docking maneuvers.

The use of a tug for the entire rendezvous maneuver is seen to involve a severe weight penalty. If use of the tug is restricted to the docking maneuver, the weight penalties are considerably reduced, as shown below.

Total Docking Fuel Weights

No tug:	38 pounds
20,000-pound tug:	60 pounds if coupled electrically, 153 pounds if not.

If one assumes that the tug is a dual-purpose vehicle which would be in orbit whether used during the docking operation or not, then the use of the tug is justified if 22 pounds of equipment associated with docking the payload can be eliminated in the incoming payload, and the tug and payload can be connected electrically. If the tug must operate without electrical interconnections, then 115 pounds of equipment must be eliminated to justify its use. Thus, if the tug and payload can be interconnected electrically, use of a tug appears promising for docking. If not, the tug seems somewhat marginal.

SECTION IV

RECOMMENDED SYSTEMS AND TRADE OFF CONSIDERATIONS

4-1. OLO VEHICLE ATTITUDE-CONTROL-SYSTEM SUMMARY

A summary of the detailed discussion which follows is presented here. The major elements of the attitude-control systems recommended for the orbital launch facility, the unmanned payload, and the astrotug are presented. Tabulations of power, weight, and volume requirements are included.

The attitude-control system recommended for the Orbital Launch Facility (OLF) is quite conventional. A horizon seeker serves as the pitch-and-roll reference, slaving the vehicle to the local vertical. An integrating rate gyro is used to stabilize the vehicle in yaw. Periodic manual realignment of the yaw axis will probably be required, although automatic alignment is feasible. A capability for manual reorientation prior to docking is provided, along with a fine-attitude-hold mode for use during docking maneuvers. Torques are obtained from a storable bi-propellant reaction-jet control system. Table 4-1 contains summary information pertaining to the OLF attitude control system.

The attitude of an unmanned payload is controlled remotely from the orbiting facility if no tug is available. An inertial reference unit is required in the payload to derive vehicle attitude for transmission to the permanent facility. The most significant attitude-control requirement

for the unmanned payload is the wide range of torques necessary to perform the requisite maneuvers. Orientation of the vehicle for thrusting during rendezvous requires large torques, while the docking procedure will require very small torques. The high torque required for rendezvous places a definite weight advantage with the storable bi-propellant system. Summary data is presented in table 4-2.

The components required for attitude control of the manned astrotug are listed in table 4-3. The system is functionally quite similar to that for the unmanned payload, except that control of the tug is exercised directly, rather than remotely as for the unmanned payload. The recommended method of torque generation in this case uses boil-off hydrogen from the main propulsion tanks as a power source.

Attitude-control requirements for the unmanned payload are listed in table 4-4 for the case when a tug is available. Comparing with table 4-3, it can be seen that the presence of a tug eliminates 55 pounds of equipment in the payload proper. However, as discussed in the preceding section, the overall fuel-weight requirements (tug plus payload) will be much higher than for the payload alone.

The functional requirements presented in section III, and the attitude control system characteristics outlined above, indicate that Orbital Launch Operations will require no major state-of-the-art advances in attitude control. Control-torque requirements, dynamic-response and accuracy specifications, and the range of critical variables, are all within the capabilities of available components.

There are, however, several areas which will require intensive study. These include attitude control of two vehicles when in close proximity to one another, and attitude control of the coupled tug/payload combination. A discussion of these two items is presented in section VI, where the problems are outlined in detail and a program is suggested to solve them. It must be emphasized that this is not a question of advancing the state-of-the-art, but rather one of applying known techniques and equipments to a new problem.

It also appears from tables 4-1 through 4-4 that whenever applicable, storable bipropellant systems offer a significant weight advantage. Thus a developmental program aimed at moving the reliability of the storable bipropellant system would seem in order.

TABLE 4-1

ORBITAL LAUNCH FACILITY
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

Component	Weight (lbs.)	Volume (ft.3)	Power (V.A.)	Remarks
1 Integrating Rate Gyro	2	0.02	13	
1 Three Axis Rate Gyro Assembly	3	0.04	24	
1 Panel Controller	3	0.05	1	
1 Electronic Assembly				
3 Threshold Detector Amplifiers	15	0.25	50	
1 Gyro Torquer Amplifier				
3 Guidance Couplers Power & Switching Components				
Storable Bi-propellant Reaction Jet Control System; Nozzles, Vales, etc.	57	-	-	
Propellant Weight for 30 Days	24.5			
Totals	105	0.31	88	

ASSOCIATED GUIDANCE EQUIPMENT

2 Horizon Seekers			
1 Long Term Yaw Attitude Reference			
1 Telemetry Receiver			Provides capability for remote initiation of docking hold mode for initial rendezvous

TABLE 4-2

UNMANNED PAYLOAD (NO TUG)
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

Component	Weight (lbs.)	Volume ($ft.^3$)	Power (V.A.)	Remarks
3 Integrating Rate Gyros	6	0.06	39	
4 Linear Accelerometers	3	0.03	6	
1 Inertial Attitude Reference Unit	50	1.5	60	
1 Electronic Assembly				
3 Threshold Detector Amplifiers				
3 Gyro Torquer Amplifiers Power & Switching Components	20	0.4	66	
2 Electromechanical Integrators				
Storable Bi-propellant Attitude and Vernier-Velocity Reaction-Jet Control; Nozzles, Valves, etc.	129			
Propellant Weight for Rendezvous & Docking changeable to attitude control	33			
Totals	241	2	171	

ASSOCIATED GUIDANCE EQUIPMENT

1 Telemetry Transmitter Receiver
1 Radar Transponder

TABLE 4-3

MANNED ASTROTUG (MINIMUM SYSTEM)
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

Component	Weight (lbs.)	Volume ($ft.^3$)	Power (V.A.)	Remarks
1 3 Axis Rate Gyro Assembly	3	0.04	24	*Cannot be specified until optimum division of attitude-control functions between tug and payload is established. Quoted weight corresponds to electrically coupled tug/payload case, when the tug is used for both rendezvous and docking
1 Panel Controller	3	0.05	1	
1 Electronic Assembly				
3 Threshold Detector Amplifiers	10	0.20	33	
Power and Switching Components				
Hydrogen Boil-Off Attitude and Vernier-Velocity Reaction-Jet Controls; Nozzles, Valves, etc.	113*			
Propellant Weight chargeable to attitude control	160*	-	-	
Totals	289	0.29	58	
ADDITIONAL EQUIPMENT REQUIRED FOR AUGMENTED SYSTEM				
1 Inertial Attitude Reference Unit	50	1.5	60	
1 Resolving Computer	10	0.20	25	
1 Electronic Assembly				
5 Electromechanical Integrators	6	0.15	25	
Additional Power and Switching Components				
4 Linear Accelerometers	3	0.03	6	
ASSOCIATED GUIDANCE EQUIPMENT				
Tracking Radar Gimbal Pickoffs				

TABLE 4-4

UNMANNED PAYLOAD (WITH TUG)
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

Component	Weight (lbs.)	Volume (ft.3)	Power (V.A.)	Remarks
3 Integration Rate Gyros	6	.06	39	
1 Electronic Assembly 3 Threshold Detector Amplifiers 3 Gyro Torquer Amplifiers Power and Switching Components	15	.25	50	
Attitude Reaction Jet Controls; Nozzles, Valves, etc.	129*	-	-	*Cannot be specified until optimum division of attitude control functions between tug and payload is established. Quoted weight corresponds to electrically coupled tug/payload case, when tug is used for both rendezvous and docking.
Propellant weight for rendezvous and docking changeable to attitude control	33*			
Total	183	.31	89	

4-2. SYSTEM DESCRIPTIONS

a. Recommended System for Orbital Launch Facility

The following description is presented by control axis (pitch, roll and yaw) broken down into operating modes. The system operating modes considered are those presented in Section III.

- Initial Stabilization and Alignment
- Coarse Attitude Hold
- Docking Alignment
- Fine Attitude Hold
- Emergency Manual Control

(1) Pitch Axis

(a) Initial Stabilization and Alignment

The control signal for initial alignment is derived from body-mounted horizon seekers. As shown in figure 4-1 , the error signal, proportional to the angle between the vehicle's pitch axis and the local vertical, is filtered and fed to the ASCS threshold detector. If the attitude error exceeds five degrees, the threshold detector fires the pitch reaction jets, torquing the vehicle towards zero attitude error. The vertical-attitude error is summed with a pitch-rate gyro that provides the damping necessary to control the on-off limit cycle. The type of threshold detector employed (i.e., pulse-width modulator or conventional on-off) will be determined as a result of proposed trade-off studies.

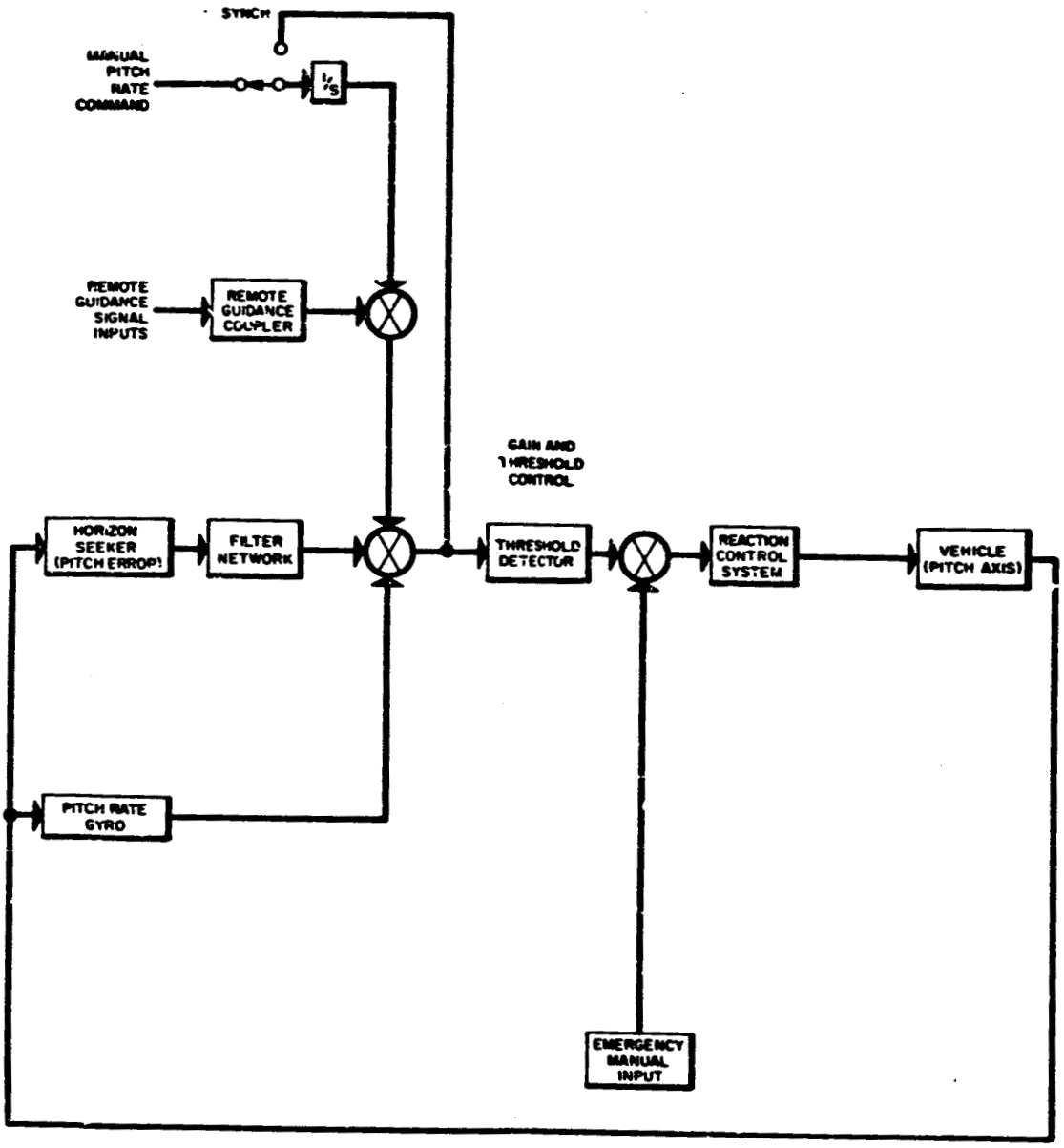


Figure 4-1. Stabilization and Control - Pitch Axis Block Diagram

(b) Coarse Attitude Hold

The system configuration for coarse attitude hold is the same as for the initial alignment. The vehicle's pitch attitude is held to within five degrees of the local vertical. The system's angular rate threshold will be set above the 4-degree-per-minute vehicle pitch rate required to track the local vertical, in order to prevent reaction-control actuation for a satisfactorily stabilized vehicle.

(c) Docking Orientation and Fine Attitude Control

During rendezvous and docking of a payload with the OLF, the OLF's attitude will be held to one degree or less, and the vehicle will be capable of being reoriented, remotely or manually, to facilitate the mating of the OLF to the incoming vehicle.

By remote or local selection, the system's attitude and rate thresholds will be tightened to 0.5 degree and 0.05 degree per second, respectively, to provide a fine-attitude-control mode. The remote selection is required for rendezvous and docking of the initial incoming payload with the unmanned OLF. Remote guidance signals could be coupled to the OLF's reaction-control system to provide a capability for remotely controlling the OLF's pitch attitude for alignment during docking, but this does not appear necessary.

For a manned OLF, a manual controller is provided with a ± 30-degree attitude command capability. The operator inputs are fed through a synchronizing integrator so that the system has control characteristics similar to the VATE control system proposed for the yaw axis.

In addition to the pitch-attitude control, the controller provides fine-attitude-control selection and gain control. A gain selector is provided to compensate for large moment-of-inertia changes that can possibly occur when large payloads are accumulated on the OLF. Manual gain programming is proposed in lieu of a more complex adaptive control.

(d) Emergency Manual Control

An emergency manual control capability is provided. Provision is made for direct coupling of an emergency control to the reaction-jet system to enable manual override for OLF attitude control. The override will be either through direct electrical inputs to the reaction-control-valve torquer or through mechanical override of the torquer output. The fail-safe characteristics of each will be considered prior to final selection.

The reaction-jet controls will be authority limited for safety. Full-on (control hardover) will not disable the men on board the OLF. In the event of a system malfunction, emergency manual control is assumed to stabilize the vehicle and allow replacement of the malfunctioning equipment. Spare replacement equipment is proposed in addition to redundancy in the interests of a less complex and therefore more reliable configuration.

(2) Roll Axis

The roll axis, shown in figure 4-2, is functionally the same as the previously described pitch axis.

(3) Yaw Axis

The yaw axis, shown in figure 4-3, functions in a manner similar to the roll and pitch axes except for its control reference. The control reference for the yaw axis is obtained by periodic manual readjustment. The operator drives the vehicle to the desired direction by means of manual inputs. When the vehicle is stabilized to within the system's deadzone, the control reference is switched to a synchronized integrating rate gyro, (i.e., the gyro follows up on itself until the switching point is reached) and the gyro then becomes the attitude reference. Deviations of the vehicle's attitude are sensed by the integrating rate gyro and when the attitude error exceeds 5 degrees (0.5 degree for fine control) or rate error exceeds the threshold, the reaction jets restore the reference attitude. Full-freedom (360-degree) yaw-attitude reorientation is available to facilitate payload rendezvous and docking.

(4) Torque Generators

Three types of reaction-jet systems were considered for the orbital launch facility. They are cold nitrogen, storable bipropellant, and H_2 -O_2 bipropellant systems. An analysis of the OLF attitude control requirements for the three systems resulted in the following weight breakdown:

Type of System	Propellant Weight (per 30 day period at 0.75 lbs/day) lbs	Tare Weight lbs	System Weight lbs
Cold Nitrogen	24.5	55	79.5
Storable Bipropellant	24.5	57	81.5
H_2 -O_2 Bipropellant	24.5	45	69.5

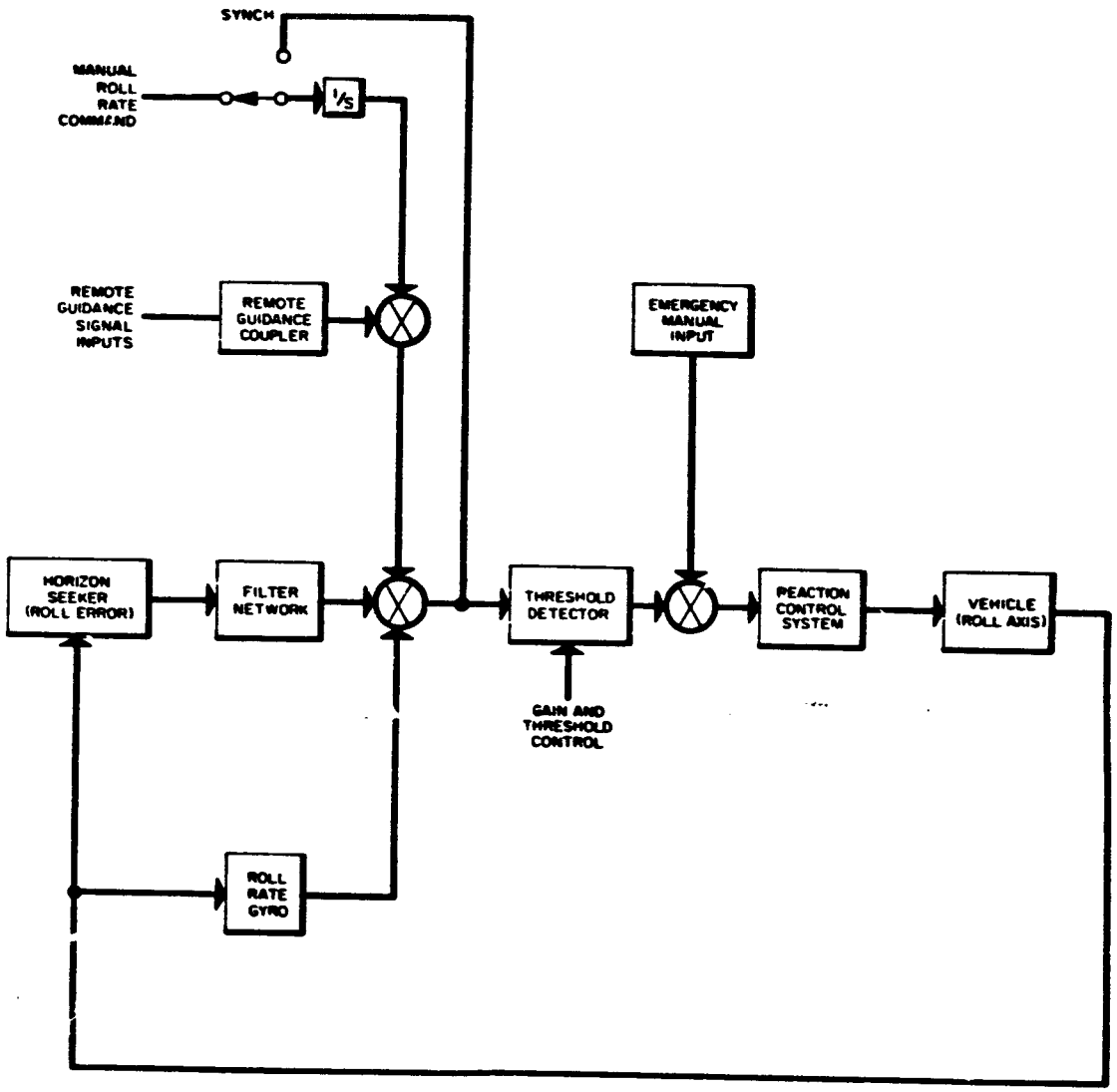


Figure 4-2. Stabilization and Control - Roll Axis Block Diagram

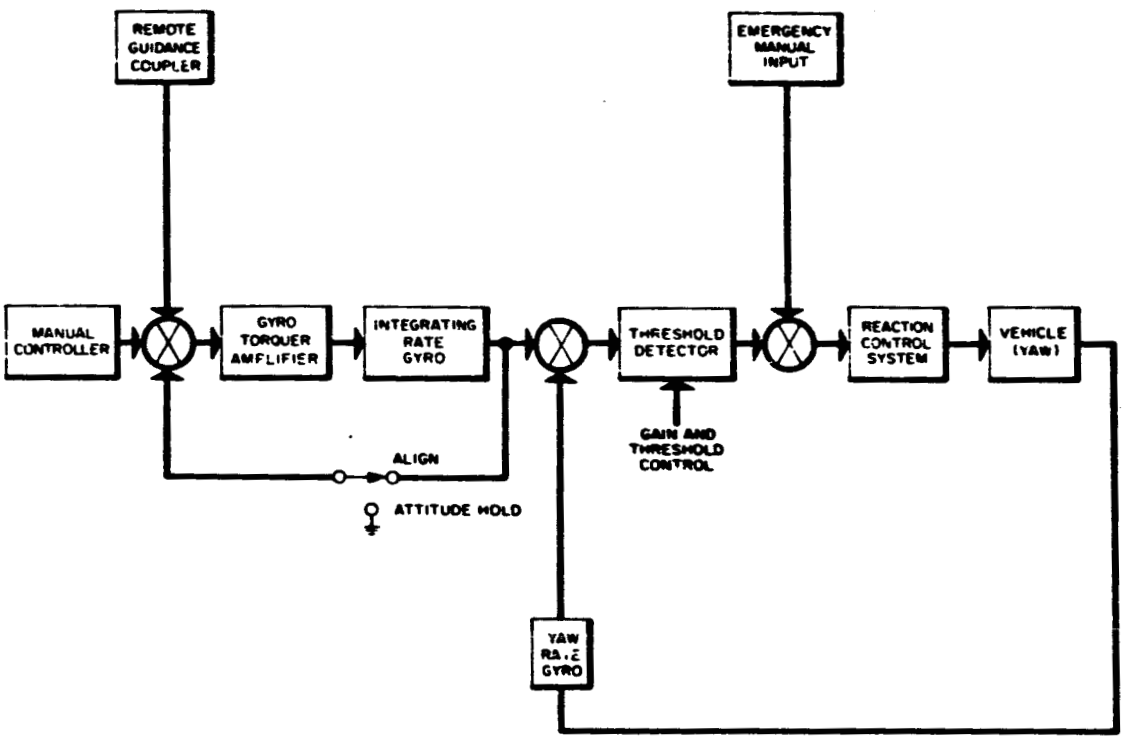


Figure 4-3. Stabilization and Control - Yaw Axis Block Diagram

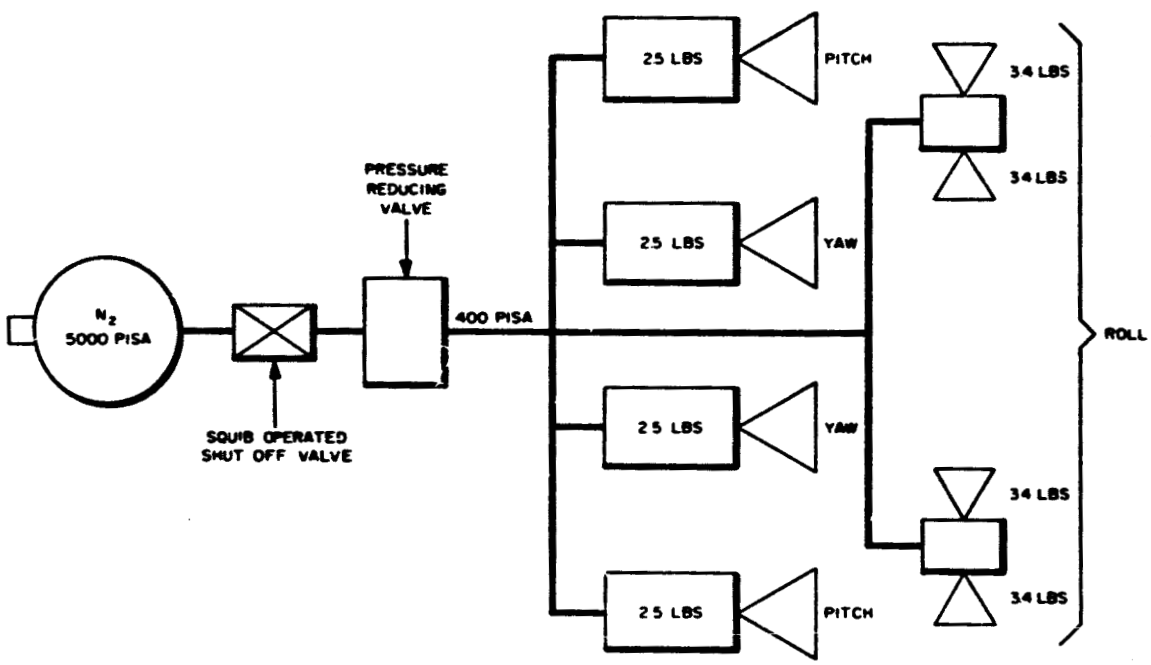


Figure 4-4. OLF Cold-Nitrogen System

The candidate systems were rated on the basis of certain parameters which were weighted by appropriate sensitivity factors. The selection parameters and the corresponding sensitivity factors are:

Reliability	1.0
Weight	1.0
Volume	0.9
Power	0.9
Availability	0.8
State-of-Art development required	0.8
Maintainability	0.8

As a result of this trade-off procedure, the storable bi-propellant system was selected as the recommended system.

b. Recommended System for Unmanned Payload Without Tug

(1) Configuration

One axis of the recommended attitude control system configuration for the unmanned payload is shown in figure 4-5. The attitude-control loop will produce an angular rate proportional to the input signal telemetered from the OLF, and will hold the existing attitude in the absence of an input. The lateral velocity-control loop, used only for docking, will produce a lateral velocity proportional to the input from the OLF and will hold essentially zero velocity in the absence of an input. An inertial reference unit is used to measure the inertial attitude and velocity

of the vehicle for transmission to the OLF. A transponder, which is used to aid radar tracking of the payload from the OLF, is also shown in figure 4-5.

The equipment required in the OLF to derive commands for rendezvous and docking of the payload is shown in figure 4-6. Although the mode of operation shown is automatic with manual override, it could also be completely manual or manual with automatic assists. The selection of a specific control method will be based on how well the human operator can perform the rendezvous and docking maneuvers in terms of time, extra fuel, and the amount of automatic equipment he replaces in the OLF.

If manual control is used, the operator in the OLF will have a display of the payload's position, orientation, and angular rates with respect to both the OLF and inertial space. From these quantities he will determine the proper attitude rate and thrust on-off commands to accomplish a rendezvous. For docking, accurate payload position and orientation with respect to the OLF docking centerline will be displayed, and the operator will determine the proper payload angular rates and lateral velocity commands.

If automatic control is used, the human operator will be replaced by a computer operating on essentially the same input information. A significant difference will be that the computer will allow more efficient (in terms of time and fuel) rendezvous and docking maneuvers.

(2) Operation

Initial orientation of the payload before it is acquired by the OLF is accomplished by having the IRG follow up on itself (electrical-caging mode) until just prior to booster separation. The payload will then hold the attitude existing at separation.

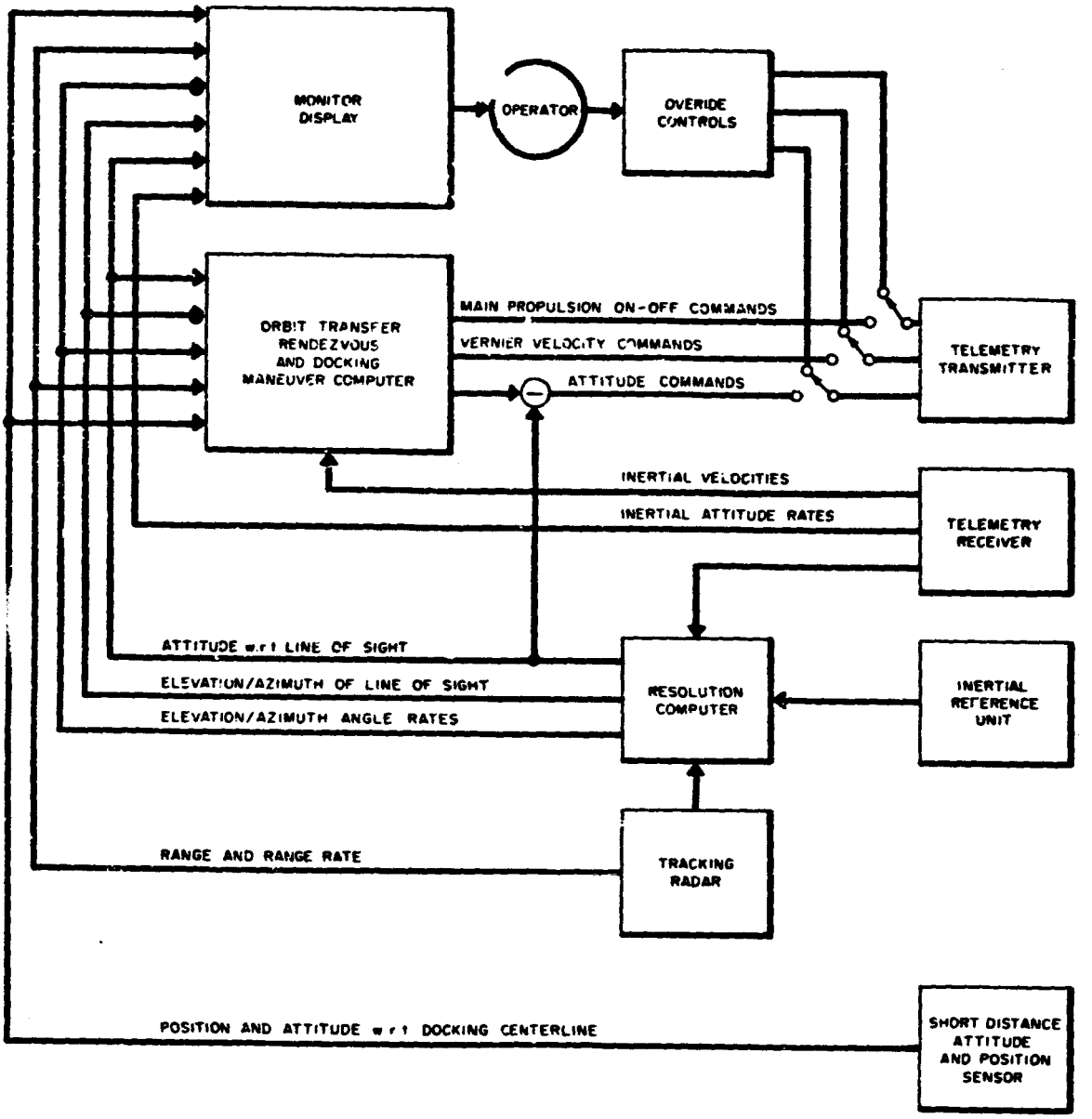


Figure 4-6. OLF Equipment for Automatic Control of Unmanned Payloads

During rendezvous, the velocity-control loop is open and the payload will rotate about its center of gravity in response to attitude commands. When docking, however, both loops are operating and the summed accelerometer signal in the velocity loop will sense the velocity of a point whose location is determined by the manner in which the signals are summed (see figure 4-7). The velocity loop will prevent this point from translating by redistributing the torque commands called for by the attitude loop. This will cause the vehicle to rotate about the selected point rather than the center of gravity. The ability to force the payload to rotate about an arbitrary point which is obtained in this manner may be useful in the docking maneuver.

In the lateral velocity-control loop, velocity is measured by integrating the sum of two linear accelerometer signals as mentioned above. Figure 4-7 shows how the accelerometer signals are combined to measure the linear velocity of a selected point on the vehicle. The error between the commanded and measured velocity then commands a proportional lateral acceleration.

An angular rate damping signal is derived by passing angular acceleration signals, obtained from the same accelerometers, through a lag network. The manner in which the outputs of the two linear accelerometers are combined to measure angular acceleration is also shown in figure 4-7. Linear accelerometers have been suggested as angular rate sensors because they are already in the vehicle (being required for lateral path-damping signals), and because they are more reliable than rate gyros.

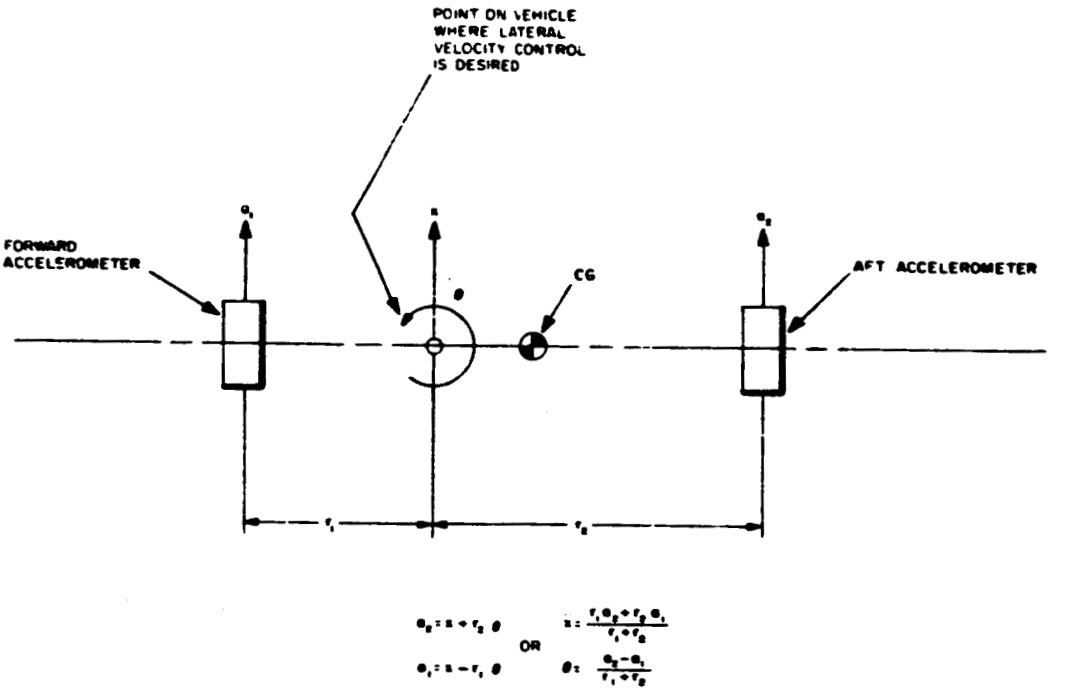


Figure 4-7. Schematic Arrangement of Lateral Accelerometers for Measuring Angular and Linear Accelerations

(3) Torque Generators

Four types of reaction-jet systems were considered for the unmanned payload. An analysis of the unmanned payload requirements resulted in the following weight breakdown for the leading two contenders:

Type of System	Propellant Weight (lbs)	Tare Weight (lbs)	System Weight (lbs)
Cold Gas-Nitrogen	139.35	263.65	403.0
Storable Bipropellant	32.8	119.2	152.0

The systems were evaluated using the same selection parameters and weighting factors as for the OLF. It is recommended that the storable bi-propellant system be used in the initial unmanned payloads, because of its tremendous weight advantage.

(4) Alternate Configurations and Trade-offs

Alternate sensors that have been considered for use in the unmanned-payload attitude-control loop are rate gyros in place of the accelerometers, and a stable-platform attitude reference in place of the integrating rate gyros.

A rate gyro can be used in place of the accelerometer/lag-network combination to provide an attitude-rate damping signal. The two techniques are dynamically equivalent. The factors involved in this choice of sensor are as follows:

- Accelerometers are required for lateral path damping and are already in the system
- Calculations show that simple E-pickoff type accelerometers will probably be satisfactory. These are more reliable than rate gyros
- The problems that arise in measuring aircraft angular acceleration with linear accelerometers, due primarily to gradient mismatch and null imbalance, do not appear to be significant in space. The difference is due to the absence of vibration and the low range of accelerations to be measured.

The inertial attitude-angle reference (platform) already in the payload could conceivably be used in place of the integrating rate gyros. However, resolvers would then be required to derive body-axes angles from the inertial angles, and a synchronizing integrator would also be required in each axis to synchronize the control system with the resolved platform outputs prior to initiation of remote control. The factors involved in the choice of sensors are as follows:

- The use of the integrating rate gyros eliminates the need for resolvers and synchronizers in the payload, and places them in the CLF where they can be used for each payload
- The integrating rate gyros provide an initial-orientation reference which does not involve preprogramming or complex computation.

Manual control of the payload from the OLF would be the same if the inertial attitude reference were used. The

operator would generate angular rate commands. These would be fed through the synchronizing integrators to give a "driven displacement" system similar to that obtained with the integrating rate gyros.

c. Recommended System for Manned Astrotug

One axis of a minimum attitude-control system configuration for the manned astrotug is shown in figure 4-8. The system provides control of attitude rate and lateral acceleration. The attitude-control loop will produce a vehicle rate proportional to the pilot's input command and will provide short-period attitude hold in the absence of an input. This basic, or minimum, system places the maximum workload on the pilot. The pilot must also be provided with a display (not shown) of tug position, orientation, and angular rates with respect to the payload or OLF in order to accomplish his mission. An inertial-reference unit and a tracking radar will be required in the tug to provide information for the display.

The ability of the pilot to accomplish efficient docking maneuvers with the minimum system is questionable. The augmented tug control system shown in figure 4-9 is recommended, even though more complex. It will provide pilot relief as well as more precise control by using additional equipment. Both attitude-angle and lateral-velocity control loops are provided. The attitude-control loop will accept attitude-rate commands from the pilot and hold the existing attitude angle accurately in the absence of a command. The lateral-velocity control loop will accept lateral-velocity commands and hold essentially zero velocity in the absence of a command. Most of the additional equipment required in this augmented attitude-control

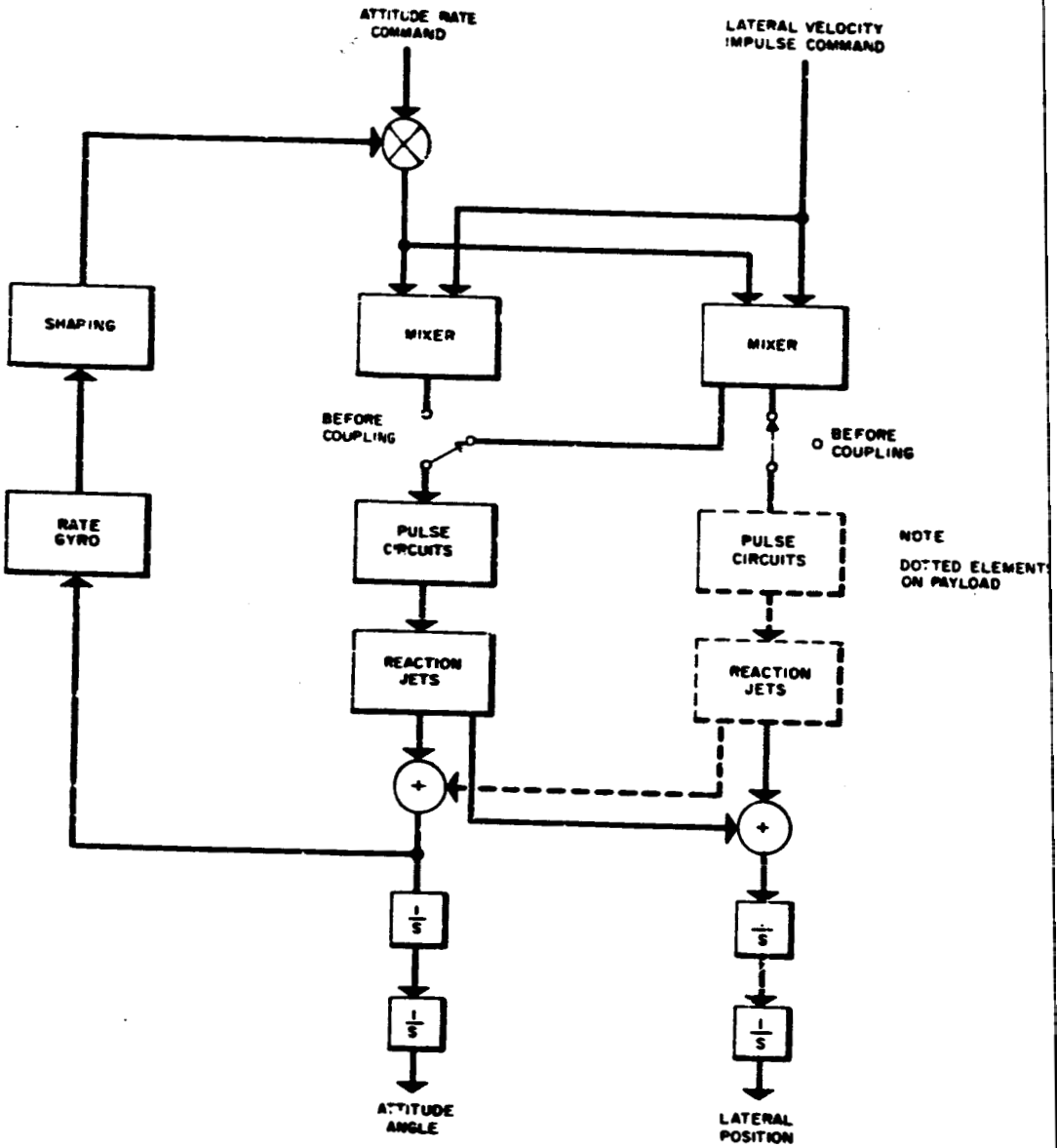


Figure 4-8. Minimum Attitude Control System for Manned Astrotug

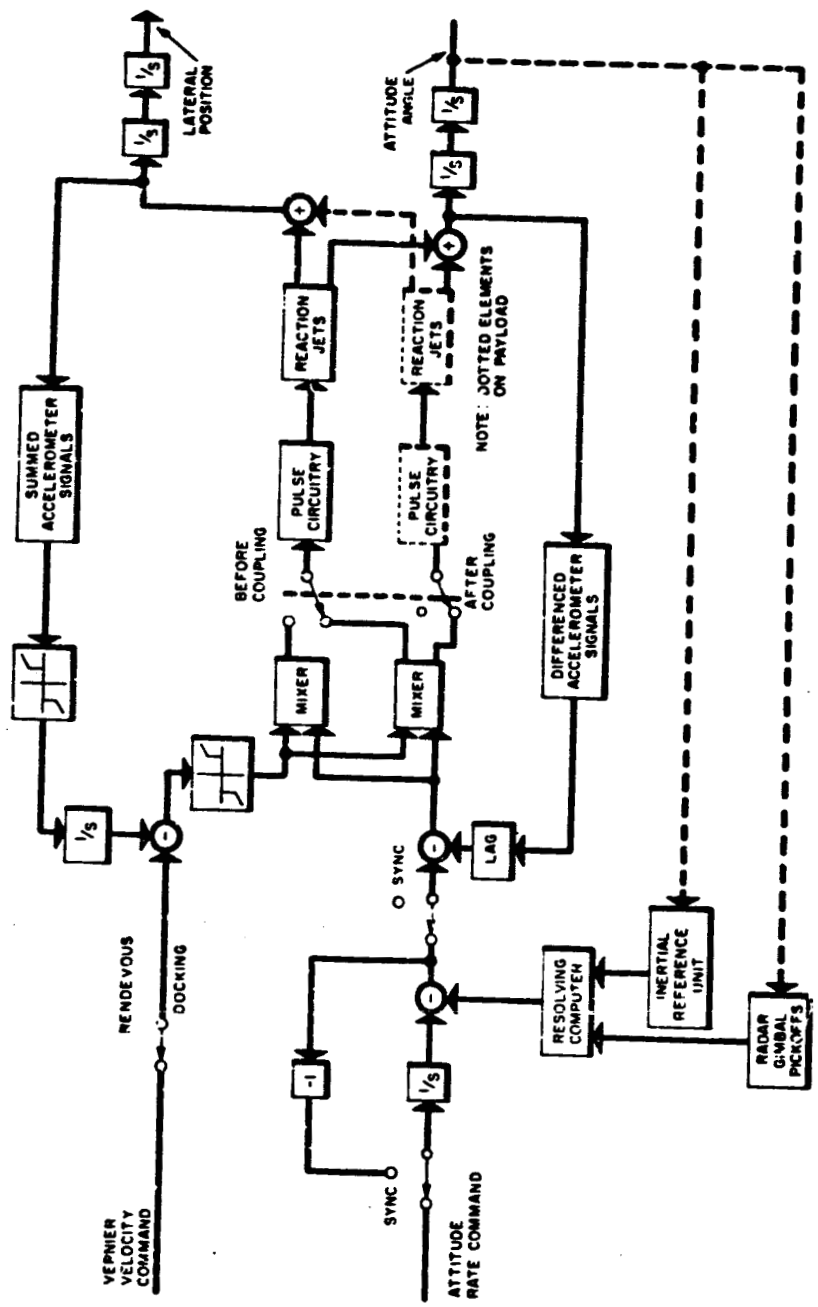


Figure 4-9. Augmented Attitude and Vernier-Velocity Control System for Manned Astrotug

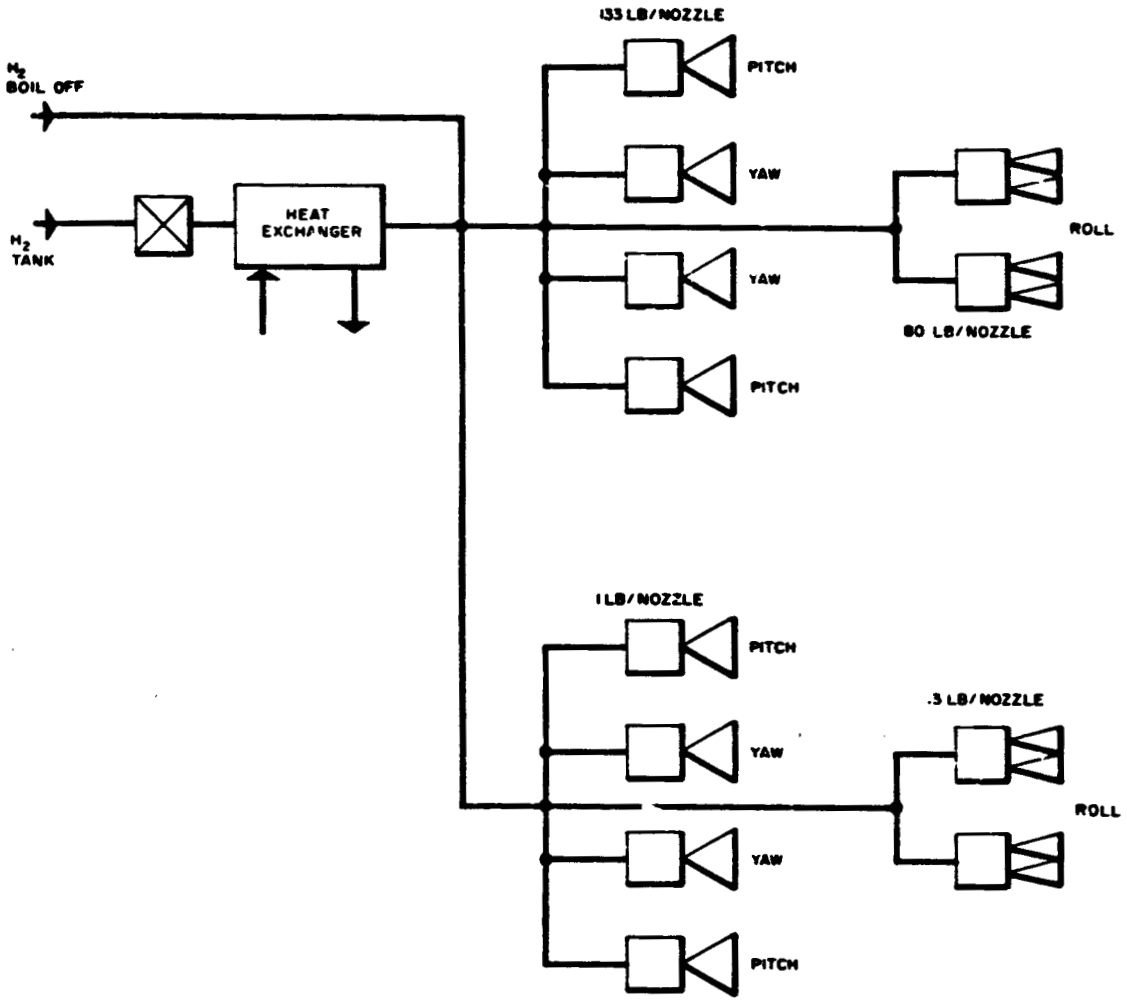


Figure 4-10. Tug Attitude Control Schematic

system will have to be in the tug anyway, since the platform, resolving computer, radar gimbal pickoffs, etc. will be required to generate information for the pilot's display. An emergency operating mode, in which the augmented system reverts to the basic system, will be provided to assume satisfactory reliability of the more complex augmented system.

(2) Operation

The augmented tug control system is similar to the unmanned-payload system described earlier. Both systems make use of summed and differenced linear-accelerometer signals to measure lateral and angular accelerations. The two systems differ only in their methods of deriving an angular-displacement error and of inserting commands into the attitude loop. The tug attitude-control system derives the attitude of the vehicle by resolving signals from an inertial-reference unit and radar-gimbal pickoffs. Synchronizing integrators are used to assure a zero attitude-angle error signal prior to engaging the attitude-control system. Manual attitude-rate commands are inserted through the synchronizing integrators. This results in a "driven displacement" type of control.

The operation of the tug velocity-control loop and the interaction of the velocity and attitude-control loops during docking is identical to that of the unmanned-payload control system.

(3) Torque Generators

A schematic diagram of the reaction-jet system recommended for the manned astrotug is shown in figure 4-10.

The system operates on hydrogen from the main propulsion tanks, assuming that this is one of the propellants used for the tug thrust engine. For the majority of operations, there will be an adequate supply of boiloff from the cyrogenic hydrogen storage. However, for a condition where additional hydrogen is required, a supply of liquid hydrogen will be taken from the main tank and flashed in a heat exchanger operating on waste heat from the crew or electronic compartments. A gas pressure of 10 psi and a gas temperature of 460°R are assumed for this system.

There are two sets of nozzles for control about each axis. One set will produce high torque levels for payload rendezvous and return, the other set will produce low torques for the payload coupling. The maximum thrusts required per nozzle for the four tug functions are tabulated below.

	Payload Rendezvous	Payload Coupling	Payload Return	Payload Docking
Max. Thrust per Nozzle (lbs)	Pitch 73.4	1.0	133.0	26.7
	Yaw 73.4	1.0	133.0	26.7
	Roll 40.0	0.3	80.0	8.0

Due to the variation in thrust levels required for the four tug functions, a variable-throat-area nozzle is proposed to provide the three highest values shown above. A small unit will be used for payload coupling.

(4) Alternate Configurations and Tradeoffs

The two configurations which were considered for the astrotug, that is, the minimum and augmented systems, have already been described. Although the minimum system has the advantage of being simpler and probably more reliable, the augmented system has been recommended. The advantages of the augmented system are as follows:

Control of attitude displacement through a long-period displacement reference is more accurate and provides more pilot relief than short-period control through a rate reference, particularly in the docking maneuver

Closed-loop control of lateral velocity is more accurate and provides more pilot relief than open-loop control

The "driven-displacement command" type of control combines a desirable maneuver characteristic (rate inputs from the pilot) with accurate attitude-displacement control.

An alternate sensor configuration considered for the astrotug was the use of integrating rate gyros in place of the inertial reference, radar pickoffs, resolving computer, and synchronizers. This tradeoff was evaluated for the unmanned-payload control system with the result being the choice of the integrating rate gyros. The factors involved in the choice of the inertial reference unit, et. al., for the astrotug are as follows:

The inertial-reference unit, radar-gimbal pickoffs, and resolving computer are already required on the astrotug for driving the pilot's display. (This was not the case in the unmanned payload tradeoff)

With "driven-displacement" control, the output of the synchronizing integrator, which is the actual displacement command, can be displayed to permit a more precise control of attitude. The displacement command can be positioned by the pilot and the system will follow-up on it with no error.

d. Recommended System for Unmanned Payload, with Tug Available

(1) Configuration

The recommended control-system configuration for the unmanned payload, if a tug is available, is shown in figure 4-11. Integrating rate gyros are used as the attitude reference. The integrating rate gyro is sensitive to body-axis rates and produces an error signal proportional to the integral of the rate, i.e., a displacement error. The gyro is caged during the injection phase and released just prior to booster decoupling. This establishes a reference attitude which the control system will hold while waiting for the tug to arrive. The shaping network is used to provide damping.

Two modes of operation are provided in the control system. One is an equivalent low-gain mode for the period before tug arrival, when it is only necessary to hold attitude to ± 5 degrees (see figure 3-4). The second is a high-gain mode for the tug coupling maneuver when it is necessary to hold attitude to ± 0.5 degree. The two modes

may be implemented by changing the error threshold in the pulse circuitry on a command telemetered from the tug. The use of an equivalent low-gain mode will reduce the weight of propellant required for attitude hold.

(2) Alternate Configurations and Tradeoffs

During payload return it will probably be advantageous to couple the payload reaction jets into the tug control system to provide all or part of the attitude control and/or propulsion forces required. This technique is recommended. The factors involved in evaluating these alternatives are primarily the fuel requirements for the payload and tug, which were discussed in paragraph 3-2.

Turning to sensors, an inertial-reference unit could be used in place of the three integrating rate gyros in the payload control system. However, this alternate has nothing to recommend it. The IRG's are satisfactory and also enjoy a considerable size and weight advantage.

A rate gyro and an electromechanical integrator could also be used in place of an IRG. This alternate has a size, weight, and reliability disadvantage. The combined effect of the rate gyro and integrator null outputs in cascade gives it an accuracy disadvantage as well.

4-3. TYPICAL COMPONENTS

a. Sensors

(1) Integrating Rate Gyro Assembly

This single-axis assembly will mount one miniature integrating rate gyro with its required excitation circuitry.

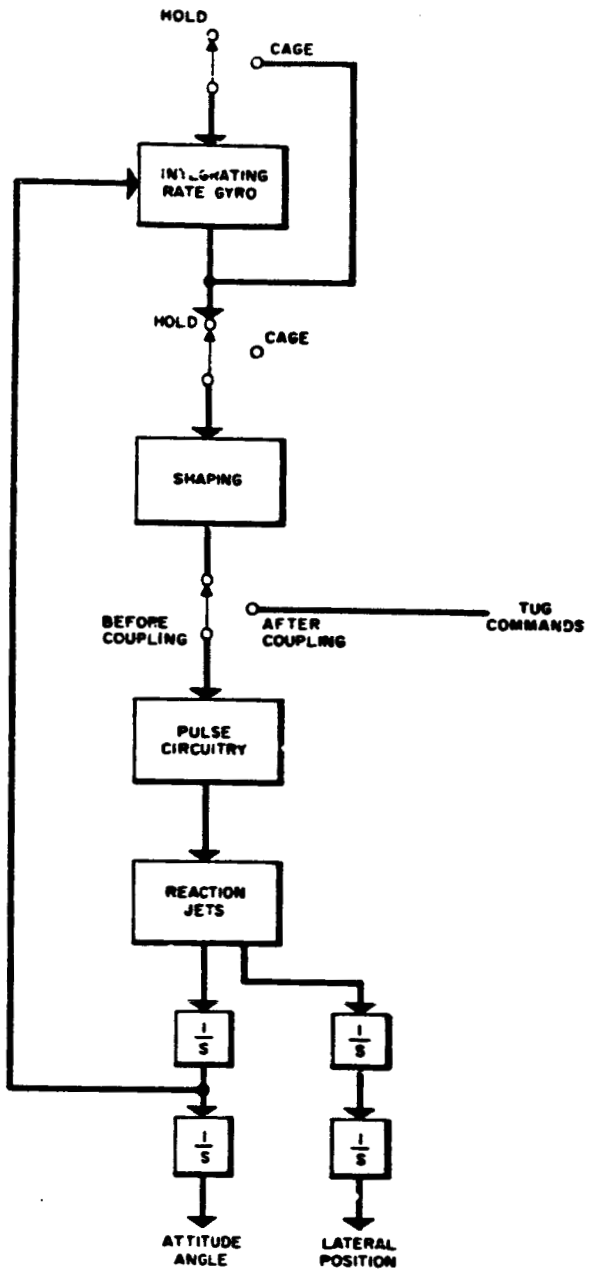


Figure 4-11. Unmanned Payload Attitude Control System with Astrotug

This gyro will be sensitive to the vehicle's yaw attitude. The Sperry SYG 1000 is an integrating rate gyro whose performance would satisfy the requirements of the proposed OLO stabilization and control systems. Performance characteristics for this unit are as follows:

Mass balance (any axis)	0.2°/hr/g
Fixed torque restraint at null	1°/hr max.
Random drift - Azimuth position	0.007°/hr
Vertical position	0.005°/hr
Aniosoelastic drift	$0.01°/hr/g^2$
Warm-up time	10 min. max.
Weight (less excitation and mounting)	0.95 lb.

(2) Rate Gyro Assembly

This three-axis rate gyro assembly will mount three miniature rate gyroscopes with their required excitation circuitry. This unit will be sensitive to vehicle yaw, pitch, and roll rates. Typical performance characteristics of a unit currently used are as follows:

Threshold	0.01°/sec.
Null	0.03°/sec.
Full scale rate	± 20°/sec.
Full scale output	5.6 volts
Motor and pickoff excitation	26 volts, 400 cps, single phase
Natural frequency	14 cps
Damping ratio	0.7 ± 0.3
Weight	3 pounds max.

NOTE: Both the above units are available with self-contained heaters.

(3) Linear Accelerometers

Preliminary calculations indicate that the linear accelerometers used to measure angular and linear accelerations can be of the E-pickoff type. These devices are rugged, small, and inexpensive. They have been used successfully in aircraft as angular rate/lateral rate sensors, but some difficulty is usually encountered because of null errors and gradient mismatch.

It is true that a null output error will result in an attitude angle error and a residual lateral acceleration, and that any mismatch of accelerometer gradients will result in cross-coupling effects. However, the magnitudes of these effects have been calculated and appear to be negligible.

Any accelerometer null output will require an attitude-positioning error to cancel it. A null output of 1 percent of full scale will cause an attitude hang-off of approximately 0.1 degree, which is within the positioning accuracy tolerances. Similarly, the accelerometer null output also enters the velocity-control loop and results in a steady-state acceleration. Since this is intolerable, the null error must be removed by deadzones on the accelerometer output and on the velocity error, as shown in figure 4-5. This deadzone will impose a velocity threshold on the system. The minimum value of this threshold is a function of the dynamic response of the accelerometer-integrator combination (not shown), and the characteristics of the pulse-modulation circuitry and the reaction jets. It does not presently appear to pose a serious problem.

The cross-coupling errors which are caused by mismatched accelerometer gradients have also been estimated. A constant angular rate of 12 degrees per second will

couple into the lateral velocity-control loop and cause a 0.02 fps velocity standoff. A constant linear acceleration of 10 fps ^{2}, mismatch will cause a 3-degree attitude error. Both of these effects correspond to a one percent gradient mismatch. Cross-coupling errors are not expected to be a problem in the docking maneuver, since the angular rates and lateral accelerations are low and are of short duration.

b. Torque Generators

(1) Cold-Gas-Nitrogen System (Backup)

The cold-gas-nitrogen system operates in the following manner: high-pressure nitrogen is stored at 5000 psi in a fibre glass container. The system is energized by a squib operated stop valve, and the nitrogen pressure is reduced to 400 psi by a pressure reducing valve. A schematic diagram of the OLF cold-nitrogen system is shown in figure 4-4.

The control valve and nozzle package is shown in figure 4-12. The valve is energized by a pulse width modulated signal. This approach employs a solenoid-staged poppet valve that has no external leakage and develops high closing forces due to the area unbalance. The response time of this system can be adjusted by matching the valve orifice area with the solenoid orifice area. The best design configuration will minimize the gas volume behind the bellows. The position vs time curve for this valve will resemble a ramp output with a transfer function as shown in figure 4-12. Inspection of this transfer function reveals that as the pulse width input signal is decreased, the impulse applied to the vehicle will approach the impulse response of the nozzle and chamber.

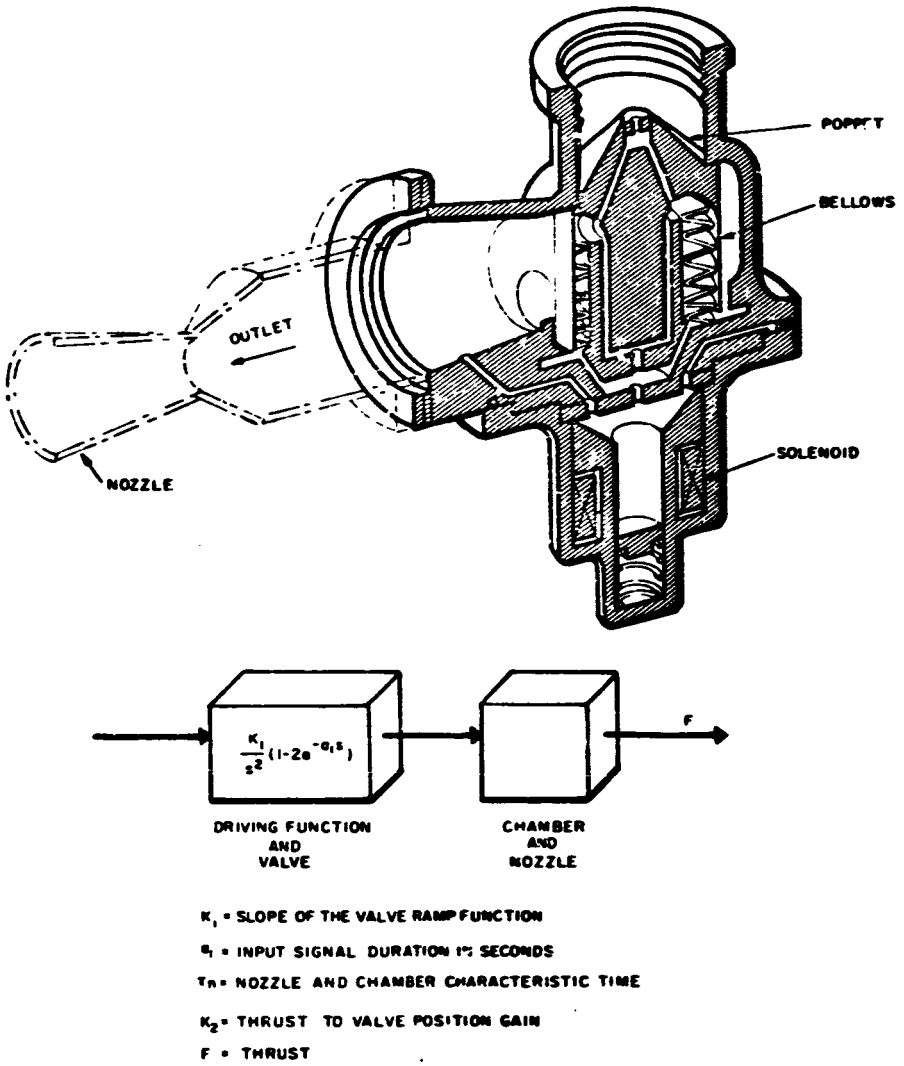


Figure 4-12. Solenoid-Staged System

(2) Storable Bipropellant System (Selected System)

Improvements in the state-of-the-art in the next two years are expected to increase the reliability of the storable bipropellant system and make it the optimum choice for later OLO vehicles. A schematic diagram of a system designed for the OLF is shown in figure 4-13. The fuel and oxidizer are stored in separate lightweight containers. A squib-operated stop valve allows nitrogen to pressurize the propellant storage containers. The nitrogen is stored at 5000 psi and reduced to 400 psi by a reducing valve. The propellant combination is hypergolic. Therefore, the combustion is spontaneous in the valve combustion chamber, and no external ignition device is required. The valve is energized by a pulse-width-modulated signal. The fuel and oxidizer controls are tied together mechanically to ensure correct oxidizer lead under all conditions. The valve block diagram and a drawing of a typical unit now in operation at Vickers is shown in figure 4-14.

Bipropellant systems, especially for long-term storage operation, present new problems in material selection since the propellants are compatible to relatively few materials. This is especially true of seal materials. However, a large number of companies have been using these propellants over the last two years, not only for attitude control but also as propellants for main boost engines, e.g., first state of Titan II, ICBM. This means that the backlog of information on the use of these propellants is increasing at a rapid rate and will continue over the years, especially when the Apollo program gets under way.

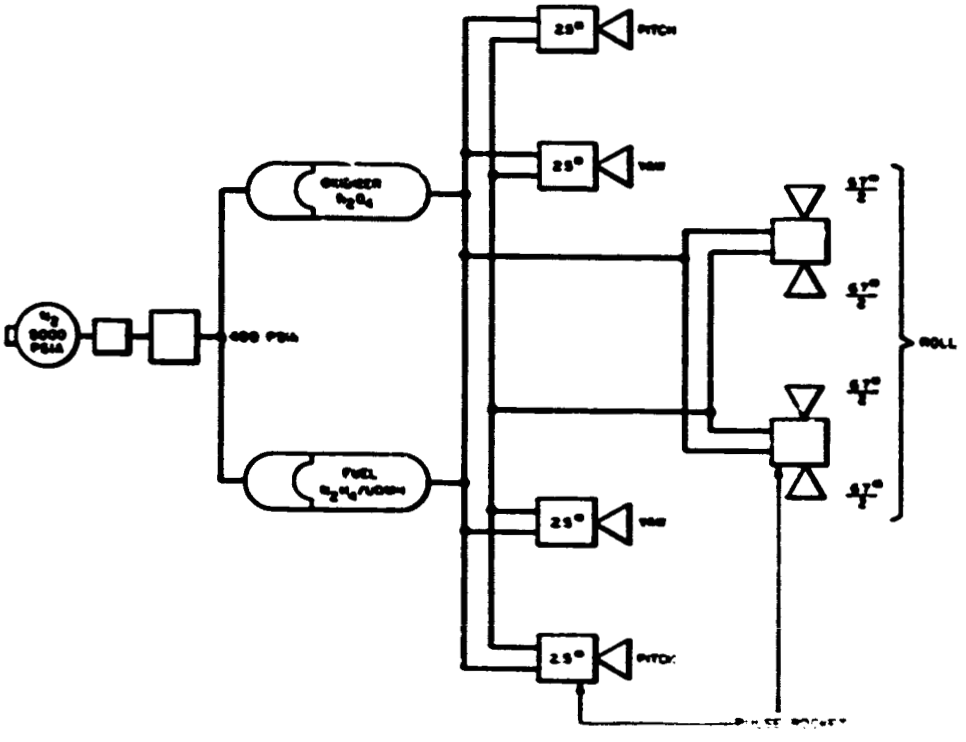
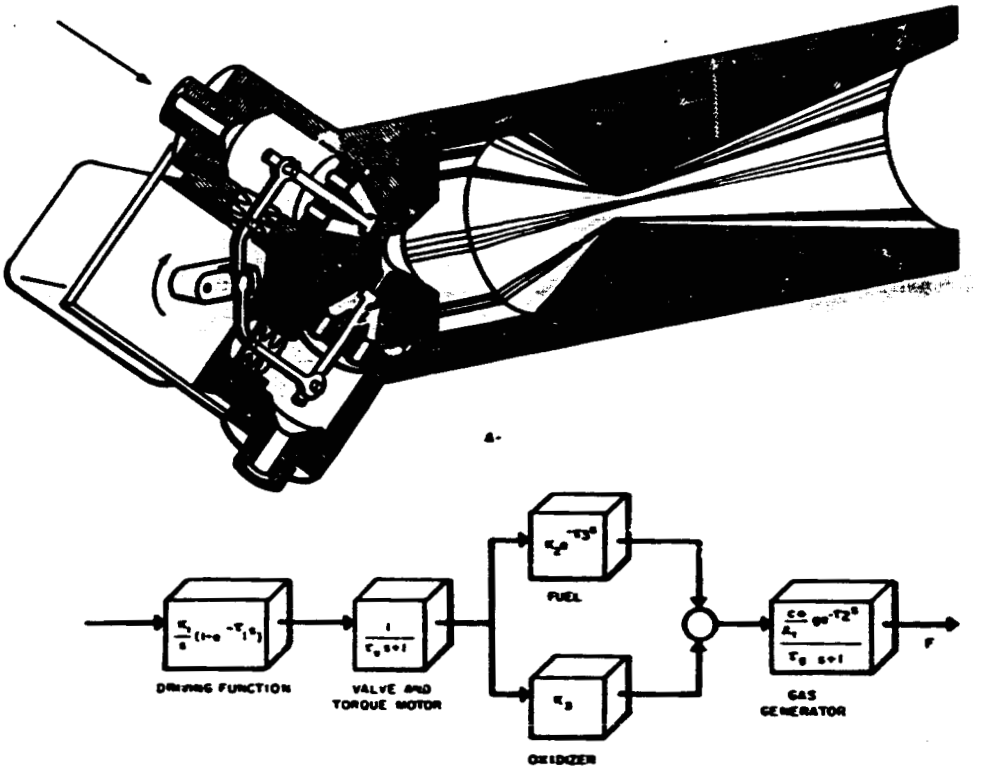


Figure 4-13. Storable Bi-propellant System



.008 SEC τ_v = VALVE AND TORQUE MOTOR CHARACTERISTIC TIME
τ_i = PULSE WIDTH - SECONDS

.002 SEC τ_2 = TRANSPORT AND COMBUSTION DELAY

.002 SEC τ_3 = OXIDIZER LEAD AND OVER-RIDE DESIGNED INTO VALVE

.001 SEC τ_0 = GENERATOR CHARACTERISTIC TIME

K_i = INPUT SIGNAL STRENGTH - AMPS

$\frac{K_2}{K_3}$ = FUEL TO OXIDIZER RATIO = $\frac{1}{1.62}$

c_o = GENERATOR CHARACTERISTIC VELOCITY

A_t = THROAT AREA

g = GRAVITATIONAL CONSTANT

Figure 4-14. Storable Bi-propellant Reaction Control System

The propellants chosen for the bipropellant attitude-control system are $N_2 O_4$ - nitrogen tetroxide - as the oxidizer, and Aerozine 50 as the fuel. The latter is a 50-50 combination of UDMH (unsymmetrical Dimethyl Hydrozine) and Hydrozine, $N_2 H_4$.

Since the vapor of Aerozine 50 is 90 percent UDMH, the toxicity of Aerozine 50 is the same as UDMH. UDMH is fairly toxic, a maximum allowable concentration of 0.5 ppm being recommended. It is not corrosive. A respirator with an ammonia cannister, splashproof goggles and vinyl-coated gloves should be worn. Detection levels by odor are much lower than the level of hazardous concentrations.

Nitrogen tetroxide is classified as a poisonous gas and artificial breathing apparatus is required to handle it. In an OLO application, the oxidizer and control equipment would not be stored in a crew compartment. Since personnel handling the material outside the new compartment would require breathing apparatus in any event, this requirement is not a serious one.

The fire hazards of Aerozine 50 may be either of the fuel type or the fuel-and-oxidizer type. The fuel type fire presents the same problems as a gasoline fire, except that firefighting is easier in the case of Aerozine 50 because it is soluble in water. The fuel-and-oxidizer type fire is much more vigorous but it is controllable by the use of large volumes of water, and is free of the detonation hazards of the non-hypergolic, immiscible types of propellants.

Nitrogen tetroxide is a stable, non-flammable, shock-insensitive material. In the absence of combustible

materials, and when it is free of dissolved organic contaminants, there are no fire or explosion hazards associated with its use.

4-4. RELIABILITY

The realistic determination of the reliability of a complex attitude-control system is an extremely difficult task. Reliability estimates were not included in the summary tables because reliability depends so heavily on factors which have not yet been specified. Examples of such factors include the packaging design details, type of circuitry employed, environment, component limitations, degree of redundancy, and maintenance considerations. However, quantitative reliability goals compatible with the present state-of-the-art have been established for the OLC attitude control equipment. These goals, the basis for their establishment, and the program required to reach them are discussed in the following paragraphs.

The first step in establishing a reliability goal is to estimate the capabilities of state-of-the-art equipment. Failure-rate data for a typical single-axis, attitude-control system is tabulated and summed in table 4-5. The pressure regulator, valving, tank, and chamber failure rates have been extrapolated from Sperry missile experience.

Applying the usual reliability formula

$$R = e^{-kt}$$

leads to the reliabilities listed in table 4-6.

TABLE 4-5

COMPONENT FAILURE RATES

Component	Failure Rate, K, failures per million hours
Threshold Detector Amplifier	127
Gyro Torquer Amplifier	36
Coupler	72
Rate Gyro	400
Integrating Rate Gyro	600
Pressure Regulator	120
Valves	50
Tanks	5
Chamber	15
Sub-total	1425
Miscellany	275
Single-Axis Attitude-Control System Failure Rate	1700

TABLE 4-6

SINGLE-AXIS ATTITUDE-CONTROL SYSTEM RELIABILITY

Time of mission	Probability of completing mission without a failure in one axis of attitude-control system.
one day	0.96
one week	0.75
one month	0.32

Most of the vehicles will be manned so that maintenance will be possible, and the mission duration of the unmanned vehicles will be much shorter than shown above. In any case, three important conclusions can be drawn:

- The reliability of OLO attitude-control equipment must be better than the current state-of-the-art
- Periodic maintenance and/or replacement must be possible. This, in turn, means that modular construction must be utilized and that emergency operating modes must be provided for use during maintenance
- Redundant circuitry must be provided to reduce the failure rate by 50 percent. This can be accomplished by providing automatic or semi-automatic replacement of some of the more critical components.

Based on the above calculations, a failure-rate goal of less than 500 failures per million hours for each axis of the attitude-control system in a typical OLO vehicle must be obtained. This will increase reliability to the order of 0.99 for a one-day period. This goal represents a definite improvement over the state-of-the-art, yet is both realistic and attainable.

SECTION V

COST AND SCHEDULE

5-1. GENERAL

A summary of the estimated cost of developing and procuring attitude-control equipment hardware for OLO vehicles through 1966 is given in this section.

The program and expenditures recommended for calendar year 1962 are discussed in detail in Section VI.

5-2. BASIS FOR ESTIMATE

Development cost estimates are based on a generalized three year program plan for all vehicles. Major milestones in this program include the following:

First Year - Establishment of system specifications
Preparation of engineering models
Performance evaluation of system models
Reliability evaluation tests
Environmental evaluation tests
Component design
Product follow-up
Design of support equipment

Second Year - Product follow-up
Preparation of operating and service instructions
Formal environmental tests
Formal reliability evaluation tests
Field test follow-up

Third Year - Product modifications
Product follow-up
Reliability evaluation and follow-up
Field test follow-up

The production schedule consists of delivery of the first engineering prototype early in the second year, with further prototypes delivered throughout the second year. The first production model would be delivered for field testing early in the third year.

Hardware procurement-cost estimates were obtained by comparison with production hardware of similar complexity. It should be noted that both development and procurement costs reflect the heavy emphasis placed on reliability. The proposed reliability program is discussed in Section VI.

Development and procurement expenditures are scheduled in accordance with a program outlined by Vought Astronautics. The major milestones in this program are

Circumlunar Mission	October 1965
Lunar Orbit Mission	March 1966
Lunar Landing Mission	October 1966

Requirements for operational availability of the individual vehicles in the OLO complex have not been established by Vought Astronautics and are beyond the scope of an attitude-control-system study. The following assumptions were made for purposes of estimating nominal delivery rates:

- The first application of the initial orbiting facility, the initial unmanned payload, and the initial orbital launch vehicle/spacecraft, will be in the circumlunar mission
- Unmanned orbital flight tests of the initial orbiting facility, the initial unmanned payload, and, the orbital launch vehicle/spacecraft will be performed in the 1963/64 period. These vehicles must attain operational status by the end of 1964 in order that manned tests can be conducted during 1965. At least ten sequentially successful unmanned test flights will be required before a vehicle achieves a man-rated status
- The first application of the other OLO vehicles will occur in the lunar landing mission
- Unmanned orbital flight tests of the intermediate orbiting facility, the intermediate unmanned payload, and the astrotug, will be performed during the 1964/65 period. Manned tests will follow in early 1966.

5-3. ESTIMATED COSTS AND SCHEDULE

A summary of attitude-control equipment development costs is given in table 5-1. Several assumptions are implicit in the figures in addition to the basic estimating procedures outlined in the preceding section.

- The cost estimates are based on concurrent development of all systems by Sperry Rand. If this is not possible, the development price will be approximately 30 percent higher for each individual vehicle attitude-control system.
- The cost of the Orbital Launch Facility includes development of sensors and attitude controls for use during retrieval and docking. This represents a major OLO problem area.
- The initial orbiting facility and unmanned payload utilize storable bi-propellant reaction controls. The quoted price includes the funds necessary to complete development of the storable bi-propellant systems.

A summary of attitude-control hardware procurement costs is also shown in table 5-1. The costs are, of course, contingent on the reference OLO schedule given in the table.

Schedules other than the one shown may be established by the operations analysis now underway at Vought Astronautics. If a particular vehicle is not required until a later date than shown, the initiation of development of an attitude-control system for this vehicle may be delayed.

Assuming the program length is as shown in table 5-1, the price will increase roughly 4 percent per year of delay because of price-index increases and growth in system complexity. Program stretchouts will raise the total cost, since a basic organization must be supported for a longer time. Development of those vehicle attitude-control systems which must be operational in 1965 cannot be delayed. The extensive qualification and flight test program required for manned orbital vehicles makes it imperative that system specifications be established early 1962.

TABLE 5-1

SUMMARY OF ATTITUDE CONTROL EQUIPMENT
COSTS FOR ALL CLO VEHICLES IN THOUSANDS OF DOLLARS

	Unit Cost (Thousands of Dollars)	1962	1963	1964	1965	Four-Year Total (Thousands of Dollars)
INDIRECT OPERATING EXPENSES						
Attitude-Control Hardware Development Costs						
• Orbital Launch Facility						
Initial		630	600	350		1580
Intermediate		110	430	510	300	1350
• Orbital Launch Vehicle						
Chemical		440	420	250		1110
• Miscellaneous						
Unmanned Payload						
Initial		280	270	150		700
Intermediate		70	270	330	180	850
• Orbital Support Equipment						
Astrotug		130	340	460	290	1310
TOTAL INDIRECT OPERATING EXPENSES		1660	2380	2050	770	6900
DIRECT OPERATING EXPENSES		Rate of Procurement				
Attitude-Control Hardware Procurement Costs						
• Orbital Launch Facility						
Initial	110		AABBBB	CCCCCCCCC	DE	
Spare Parts			F	F F		2310
Intermediate	120			A A BBBB	C C	
Spare Parts				F		1080
• Orbital Launch Vehicle						
Chemical	75		AABBBB	C C [1]	DE	
Spare Parts			F		F	900
• Miscellaneous						
Unmanned Payload						
Initial	80			A A BBBB	C C [2]	
Spare Parts				F		720
Intermediate	100				AABBBB	
Spare Parts					F	700
• Orbital Support Equipment						
Astrotug	100			A A BBBB	C C [2]	
Spare Parts				F		900
TOTAL DIRECT OPERATING EXPENSES			1,295	3,570	1,745	6,610

Nomenclature: A = Ground Test
B = System Ground Test
C = Flight Test
D = Manned Flight Test
E = First Operational Use
F = Spare

Notes: 1 Ten tests not required for man-rating because vehicle is modification of existing vehicle

2 No man-rating required

20-40% spare parts costs during initial evaluation tests is an average price

SECTION VI

PROBLEM AREAS AND RECOMMENDED PROGRAMS

6-1. PROBLEM AREAS

There are two pressing attitude-control problems peculiar to the orbital launch concept. These are

- Measuring and controlling the relative attitude of two vehicles in close proximity
- Attaching the astrotug to a second vehicle in such a manner that stable, efficient operation of the coupled pair is possible. The question of where and who applies what control torque is of particular importance from the standpoint of attitude control.

In addition, many problems not specifically arising from the orbital launch concept must be solved. The role of the human, control accuracies, and dynamic response are typical examples. A broad study of a few of the problems associated with development of attitude-control systems for the OLO vehicles was attempted in the preceding sections, but much work remains to be done.

a. The Relative-Attitude-Control Problem

There are three aspects of the relative-attitude-control problem: relative attitude of two vehicles must be sensed for control purposes, it must be maintained while

relative-velocity impulses are applied, and it must be controlled to the requisite accuracy.

Many methods of sensing relative attitude can be envisioned, particularly when it is recognized that the important control parameter is not vehicle attitude, but rather the displacement of the nose and tail from the desired line of approach. Sensing techniques leading to more or less independent control of nose and tail position seem particularly attractive from the viewpoint of automation. Examples would include measurement of nose or tail displacement through hyperbolic phase-comparison techniques, use of semi-independent beam-rider control of both the nose and the tail, or tugging on a rope attached to the vehicle while applying compensating thrusts to the tugged vehicle. Other sensors, such as radar, optical, and infra-red devices, as well as direct visual observation should also be considered.

Maintaining the relative attitude of the two vehicles while correcting their relative velocity (or position) requires that the point on the vehicle whose velocity is to be controlled be defined. Restating the problem, it is necessary to select a pivot point when commanding a change in attitude with no change in position. The desired location of this pivot point will probably vary as the docking maneuver proceeds. It will not generally be at the CG. In fact, the CG location will change with time and will vary from vehicle to vehicle. Again, some measure of independent control of the nose and tail seems desirable.

The method used to control relative attitude must also be established. Such factors as the degree of automation and the division of command authority between the two vehicles

have an important bearing on operational success. As an example, maximum use of the crew is normally desired for reliability and conservation of equipment weight, space and power. However, relative-attitude control of two vehicles in close proximity is a function which seems particularly amenable to automation. Every docking operation will be the same. A high degree of accuracy but little judgment is required to control relative attitude. Finally, manual control would require extremely close attention for extended periods of time.

b. The Coupled Tug/Payload Attitude-Control Problem

Since the tug will probably not be employed in early orbital launch operations, this problem is not considered as urgent as the preceding one and initiation of work in this area may, if necessary, be delayed. Summarizing briefly, aspects of this problem include

- Optimizing tug location with respect to the second vehicle
- Deciding between rigid and flexible couplers
- Dividing the thrusting and attitude control torquing functions between the vehicles
- Investigating the degree to which electrical interconnection of the two vehicles is desirable
- Establishing the maneuverability, dynamic response, stability, and fuel efficiency of the coupled pair.

c. Attitude-Control-System Development Problems

Many stability and control areas require study and analysis in order to determine the optimum configurations

consistent with maximum reliability and safety, and minimum cost and weight. Design studies must be initiated immediately if the OLO vehicle attitude control systems are to be operational by 1965. Areas which should be studied as part of the system design study include

- Establishment of stability and maneuverability requirements for the vehicle
- Investigation of the role of the crew. Primary, back-up, and emergency functions of the crew must be defined and the remaining functions automated
- Selection of firm system configuration
- Selection of sensors, control technique (pulse-width modulated vs. on/off), reaction-torque generators, displays, and manual control equipments
- Investigation of vehicle stability and dynamic response to establish gain-switching and shaping-network requirements
- Allotment of error and unreliability margins to each subsystem and/or component
- Establishment of integration factors such as electrical, hydraulic, and pneumatic power; equipment weight and space; and the form, level, and impedance matching required for all signals flowing between subsystems. It is also necessary to establish which sensors may be common-use items with other systems and to define the interface equipment required for successful integration.

One particular problem worthy of special attention is that of developing bipropellant attitude-control systems for the first set of OLO vehicles. Extended performance and environmental testing of a representative bipropellant system would lead to increased confidence in system reliability. The state of the bipropellant attitude control system art should then be comparable with that of cold-gas systems, ensuring a trouble free qualification test program on the OLO systems.

6-2. PROGRAM RECOMMENDATIONS

Initiation of the following programs early in CY 1962 appears imperative if the OLO schedule is to be maintained.

a. Relative-Attitude-Control Study

Tasks comprising this study are

- Comparison and evaluation of alternate methods of sensing relative attitude
- Synthesis of several tentative attitude/vernier-velocity control systems and preliminary analytical comparison of them
- Analog computer simulation of the several most promising systems followed by detailed computer evaluation. Manual control effectiveness would be investigated as part of this phase
- Preparation of system specifications, including sensor selection, definition of a control technique, and component requirements.

b. Attitude-Control-System Development

This task would include the following items for each vehicle control system considered:

- Investigation of the role of the man
- Selection of system configurations, sensors, attitude control torquers and control techniques
- Analog computer investigation of vehicle/control-system dynamics
- Establishment of component specifications and interface requirements.

It is recommended that such a program be undertaken for development of control systems for the initial OLF, the Orbital Launch Vehicle/Spacecraft, and the Unmanned Payload, in the near future. If at all possible, work should also be initiated on the storable bipropellant development program and the coupled tug/payload attitude-control study. Development of attitude controls for other vehicles can be delayed if absolutely necessary.

Finally, a discussion of a proposed reliability program, essential to the success of the recommended studies, has been included in the following section.

6-3. RELIABILITY PROGRAM

It is recognized that reliability is an indispensable requirement for successful Orbital Launch Operations. The only way to achieve high reliability is through an intensive, reliability-oriented design effort. This requires a developmental program in which

- Reliability is a primary design objective
  All personnel involved with the development program will be made aware that management has established reliability as a major goal and is willing to pay the price required to reach this goal.

- Reliability is treated as a quantitative design parameter
  Every set of alternate configurations or components shall be compared on the basis of reliability. Every proposed system modification aimed at improved performance shall be evaluated in terms of its effect on system reliability. Quantitative trade-offs are to be the basis for design decisions whenever possible.

- Reliability-improved design techniques are utilized whenever possible
  Several of the most important techniques to be used include: the use of standard, field-proven components; component derating; careful attention to packaging design in order to assure an environment suitable for the selected components; and a conscious striving for design simplicity.

- Reliability Control is exerted positively through a Reliability-oriented design review.
  Every drawing is to be reviewed and given:

  A functional check - will it work?

  A component application check - are the components suitable for the application? Derating, environmental limits, and tolerance acceptability are to be emphasized.

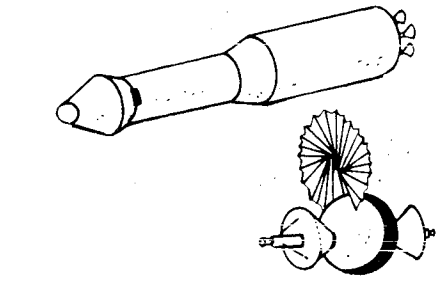
A packaging check - are the components mounted and located in such a manner that they will withstand the expected environment? Vibration resistance and ability to dissipate heat will be given special attention.

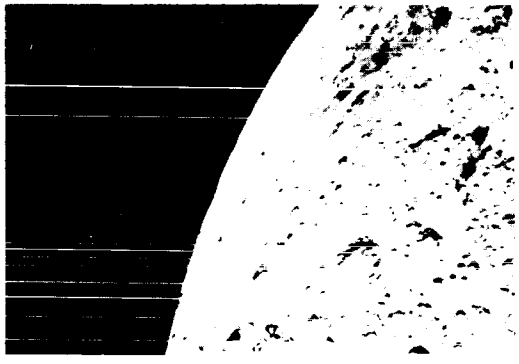
A failure-loading check - does failure of any one component impose undue loads on the rest of the system? Can the failure of a component be made non-critical so that continued degraded operation is possible?

Sperry Rand will make such an engineering reliability effort, to be followed by a thorough quality-assurance program. A particular effort will be made to maintain very close liaison between Quality Control and Engineering in the early phases of the development cycle, in order that Quality Control can inform Engineering of reliability-critical areas, and in order that Quality Control may become aware of potential difficulties.

PHASE II PROGRESS REPORT

16 JANUARY 1962





ORBITAL LAUNCH OPERATIONS

VOL. V
COMMUNICATIONS, ATTITUDE CONTROL AND SECONDARY POWER

VOUGHT ASTRONAUTICS

SPERRY-RAND CORPORATION

Control No. 64003
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George C. Marshall Space Flight Center
(Future Projects Office)
Huntsville, Alabama

Study of

ORBITAL LAUNCH OPERATIONS

Volume V

COMMUNICATIONS, ATTITUDE CONTROL, and SECONDARY POWER

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INTRODUCTION

During Phase I of this study, Chance Vought made preliminary investigations in several electrical and electronic subsystem areas. These included:

Communications

Stabilization and Control

Auxiliary Power

At the conclusion of Phase I, it was realized that many problem areas existed and that it would be desirable to pursue the studies further toward the broad objectives of resolving those problems, optimizing systems relative to OLO requirements and outlining development plans needed to assure availability of operational hardware to meet the OLO development schedule.

In order to secure maximum penetration of those subjects in the limited time available, it was decided to invite members of industry who have special capability in those specific fields to participate with Chance Vought in the last phase of the study.

Sperry Rand Systems Group of Sperry Rand Corporation has done an over-all study of subsystems in orbital launch operations, including communications, attitude control and secondary power. The results of those studies are included herein as "Volume V, Part 1, Communications; Part 2, Attitude Control; and Part 3, Secondary Power."

In order to provide guidance for this work and to insure that their studies would be compatible with the work of Chance Vought, Sperry was furnished the following information:

1. Copies of the Chance Vought Phase I report.

2. Design information previously developed by Chance Vought relative to their study subjects.

3. Need to know to visit NASA and other agencies to secure background information.

4. A statement of work approved by the NASA project office stating the study objective, OLO program description and schedule, study ground rules, study areas, study approach and schedule. Emphasis was placed on those items of greatest interest to NASA and Chance Vought.

5. Liaison, including visits and meetings to review work progress and verify compatibility with the Chance Vought effort.

Information developed in these subcontracted studies has been incorporated into the Chance Vought work principally in the form of design characteristics of equipments and reliability estimates as they affect mission success probability.

VOLUME V

ORBITAL LAUNCH OPERATIONS

PART I

COMMUNICATIONS

TABLE OF CONTENTS

Paragraph		Page
	SECTION I	
	SUMMARY	1-1
1-1	Introduction	1-1
1-2	Evaluation	1-1
	SECTION II	
	INTRODUCTION	2-1
2-1	Purpose of Report	2-1
	SECTION III	
	BASIC CONSIDERATIONS	3-1
3-1	Organizational Plan	3-1
3-2	Basic Concepts	3-1
3-3	Differences Between OLO Communication and Other Communications Systems	3-5
	SECTION IV	
	CONTINUOUS COMMUNICATIONS, GROUND TO OLF	4-1
4-1	Requirements for Continuous Communications	4-1

TABLE OF CONTENTS (Cont)

Paragraph		Page
4-2	Extension of Existing Facilities	4-3
4-3	Cost of Ground Extension	4-4
4-4	Use of Satellites	4-5
4-5	Comparison of 350 Mile and 22,300 Mile Satellite Systems	4-7
4-6	Non-coplanar Satellite	4-10
4-7	Coplanar Satellite	4-10
	SECTION V	
	CANDIDATE OLF TO GROUND SYSTEM	5-1
5-1	Satellite Ring	5-1
5-2	Ground Fence	5-1
	SECTION VI	
	OLF R-F EQUIPMENT	6-1
6-1	R-f Equipment	6-1
6-2	Circuit to OLV	6-1
6-3	Circuits to Astro Tug and Payloads	6-2
6-4	Circuits to the Extra-Vehicular Man	6-4

TABLE OF CONTENTS (Cont)

Paragraph		Page
	SECTION VII	
	MODIFICATIONS OF SATELLITE RING FOR CONTINUOUS TRACKING COMMUNICATIONS	7-1
	SECTION VIII	
	OLF TELEMETRY DATA EQUIPMENT	8-1
8-1	OLF Data Handling Requirement	8-1
8-2	OLF Communications System - Operating Modes and Evolution	8-2
8-3	OLF Reliability	8-13
8-4	OLF Maintainability	8-14
8-5	OLF System Flexibility	8-14
8-6	Role of Man	8-15
	SECTION IX	
	CONCLUSIONS	9-1
9-1	Other Systems	9-1
9-2	Candidate System	9-2
	APPENDIX A	
	BASIC NETWORK GEOMETRY	A-1

TABLE OF CONTENTS (Cont)

Paragraph		Page
	APPENDIX B	
	TYPE OF CHANNELS	B-1
	APPENDIX C	
	NET CONTROL CONSIDERATION	C-1
C-1	General	C-1
C-2	Simplex Radio Communication	C-1
C-3	Duplex Radio Communication	C-3
C-4	Net Control Stations	C-3
	APPENDIX D	
	THE FENCE	D-1
	APPENDIX E	
	THE SATELLITE RING	E-1
E-1	General	E-1
E-2	Frequency Plan	E-1
E-3	200-KC System	E-3
E-4	Satellite Equipment Costs	E-8
	APPENDIX F	
	CANDIDATE OLF TELEMETRY SYSTEM	F-1

TABLE OF CONTENTS (Cont)

Paragraph		Page
F-1	Candidate Systems	F-1
F-2	Description of OLF Conceptual Communication System	F-7
	APPENDIX G	
	RELIABILITY DEVELOPMENT PROGRAM	G-1
G-1	Introduction	G-1
G-2	Objectives	G-1
G-3	Approach	G-1
G-4	Evaluation Criteria	G-24
G-5	Problem Areas	G-33
G-6	Conclusions	G-36
G-7	Assumptions	G-37
G-8	Recommendations	G-38
	APPENDIX H	
	CONCEPT OF MAINTENANCE AND LOGISTICS	H-1
H-1	Introduction	H-1
H-2	Objectives	H-1
H-3	Approach	H-2
H-4	Logistics	H-9
H-5	Evaluation Criteria	H-11
H-6	Problem Areas	H-12
H-7	Conclusions	H-15
H-8	Recommendations	H-15

LIST OF ILLUSTRATIONS

Figure		Page
2-1	Basic Requirements	2-2
4-1	Earth-Based Tracking Coverage of OLF for a 24 Hour Period	4-4A
4-2	Maximum Line of Sight Path	4-4B
4-3	Equatorial Satellite Ring Ground Coverage	4-4C
4-4	Pattern for Continuous Coverage	4-4D
4-5	Rough Estimate of Communication Ground Net Cost	4-5
4-6	Types of Satellite	4-6
4-7	Comparison of Characteristics of Three Types of Satellites	4-9
4-8	OLF Satellite Relay Ring - Ground Fence Concept	4-12
4-9	North-South Fence Geometry	4-13
5-1	Typical Space Relay Satellite	5-3
5-2	OLF Equipment	5-4
6-1	Path of OLV Relative to Path of OLF for Altitudes Below 6000 Miles	6-3
8-1	Estimate of Telemetry Measurement Requirements	8-4
8-2	Summary of Telemetry Requirements By Operation	8-10

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
8-3	Measurement Handling Operation	8-12
8-4	OLF Measurement Requirements	8-12
8-5	OLF Telemetry System Programming	8-16
9-1	Comparison of Three Basic Concepts for Continuous Communications	9-3
9-1A	Old Communications Net	9-4A
9-2	Candidate System	9-4B
9-3	Table of System Equipment	9-5
9-4	Program Schedule Telemetry and Communication	9-9
9-5	Projected Costs	9-10
A-1	Geometric Concept - Disc Rotating at Angular Velocity W_1	A-2
A-2	Geometric Concept - Segmented Disc	A-2
A-3	Geometric Concept - Interchanged Brush and Fixed Connections	A-2
A-4	Geometric Concept - Addition of Termination C	A-2
A-5	Geometric Concept - Addition of Termination C at Different Angular Velocity	A-2
C-1	Simplex Type of Radio Communication	C-2
C-2	Duplex Type of Radio Communication	C-2

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
C-3	Net Control Station - Method A	C-2
C-4	Net Control Station - Method B	C-5
C-5	Net Control Station - Method C	C-5
C-6	Typical Station B	C-6
C-7	Frequency Plan for OLF Communication Link	C-6
C-8	Sequence of Possible Communication Links	C-8
D-1	Fence Relay Station	D-2
D-2	2300 MC Section of Fence Relay Station	D-4
D-3	Half of the HF Section of Fence Relay Station	D-6
D-4	Fence Costs for 48 KC Bandwidth	D-8
E-1	OLF Satellite Ring - Ground Fence Concept	E-2
E-2	Ring Frequency Plan	E-2A
E-3	Ground Station	E-2C
F-1	Comparison of Candidate Systems	F-2
F-2	PCM System	F-8
F-3	Projected Electrical Characteristics of Communications System	F-8A

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
F-4	Physical Characteristics - Basic Telemetry System	F-8A
F-5	Physical Characteristics - Additional Telemetry Equipment	F-9
F-6	Alternate Hybrid System	F-10
F-7	Encoder Subsystem	F-10B
F-8	Programmable Multiplexer Section	F-12A
F-9	Basic Switch	F-12B
F-10	Programmed Multiplexer Driver	F-14
F-11	Waveforms of Driver Unit	F-14A
G-1	Basic Reliability - P = 0.50 and P = 0.90	G-6A
G-2	Basic Reliability - P = 0.95 and P = 0.99	G-6B
G-3	Reliability	G-8
G-4	Reliability - Spares Interchangeability	G-8B
G-5	Single Redundancy - Double Redundancy	G-11
G-6	Reliability Redundancy	G-12
G-7	Communications System	G-13
G-8	Reliability for Checkout and Telemetry Equipment	G-14A

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
G-9	Latin Squares - Sample of Three Items	G-27
G-10	Analysis of Variance	G-29
G-11	Unit Proficiency Growth	G-34
H-1	Maintenance Model	H-3
H-2	Preliminary Packaging Concept	H-6A
H-3	Effect of M $ Maintenance on POMS $ Cost	H-12A
H-4	Maintenance and Logistics Problems	H-13

SECTION I

SUMMARY

1-1. INTRODUCTION

The Sperry Rand Corporation is supporting Marshall's Orbital Launch Operations Study (Vought Astronautics Contract NAS 8-853) by investigating the major Astrionic requirements for the program. The subject of this volume is Communications and Telemetry, which covers the work accomplished by the Sperry Rand Systems Group assisted by the Sperry Utah Company, Salt Lake City, Utah and the Sperry Development Center, Sunnyvale, California.

1-2. EVALUATION

The principal conclusions of this work are:

a. The communications and telemetry aspects of the OLO program are within the time scale for state-of-the-art development as conceived by Sperry Rand Corporation. No major technological break throughs are necessary to successfully establish Orbital Launch Operations.

b. Certain follow-on study work is absolutely necessary to firmly establish plans for successful Orbital Launch Operations. In the area of Communications and Telemetry these are as follows:

(1) The requirement for continuous communications must be established in depth and fully documented since present

indications are that this communication continuity is a major requirement for successful Orbital Launch Operations.

(2) The continuous system utilizes satellites to great advantage. Sperry Rand Corporation proposes to develop the specifications and schedules for the necessary satellites if required.

(3) Specifications and schedule for the complete communications system should be developed by June 1962 to meet 1965 circumlunar flight schedule.

(4) Realistic reliability studies are needed for the spaceborne communications arrangement. These should include breadboarding and testing of key elements.

Since the work is well within the capabilities of the SUCO and SDC Divisions of the Sperry Rand Corporation, it is requested that assignment of this work to Sperry Rand for the next funding period be considered. Budgetary cost information for this work is presented in Section 9.

c. Other recommended follow-on work is as follows. These items, while not placed in the absolutely necessary category, are still considered necessary from the standpoint of well balanced planning and proper initiation of work programs necessary to meet the objectives of the national OLO program on schedule.

(1) The NASA plans for integration of communications, telemetry, ranging and range rate determination through the use of the same equipment using different forms of modulation

should be incorporated in the OLO operation if timely. The work that JPL, MIT and STL have been conducting indicates a strong trend in this direction but considerable study is required to see whether or not any of the concepts will be generally implemented soon enough.

(2) After the broad communication plan has been partially detailed, a check should be made on uniformity and interchangeability in order to minimize the logistic problems aloft.

(3) Work should be continued concerning the use of low flying satellites as the transmission medium from outlying radar sites to the operation center.

(4) Investigate methods for reducing intentional interference from an enemy and also possible unintentional interference.

d. These programs are also recommended for Sperry Rand support of the Marshall Space Flight Center OLO Development Program. The funding requirements for each of these work items is presented in Section IX.

e. Some of the major technical conclusions which are of interest to the Marshall Space Flight Center as an aid in planning the work are listed.

(1) Continuous communications and telemetry is of paramount importance to the probability of OLO mission success and on the order of 75 KC of bandwidth can be provided by low flying satellites in the time period required.

(2) Switchable duplex circuits should be provided from the OLO operations center and computer to various portions of the OLO complex aloft in order to provide the necessary flexibility and allow for expansion if required.

(3) Low flying satellites may be used to supply data input paths to the operations center from outlying radar sites.

(4) The low flying satellites system is superior to the 22,300 mile high flying system by a launch weight ratio of about 21:1.

(5) RF equipment should be readily tunable to 3 or more frequencies in the same band.

SECTION II

INTRODUCTION

2-1. PURPOSE OF REPORT

The purpose of this report is to present the results of investigations conducted by the Sperry Rand Systems Group in the field of communications and telemetry. The Sperry Rand Corporation is conducting an across the board analysis of the various phases of Orbital Launch Astrionics to aid the Marshall Space Flight Center in developing a firm and well documented plan for establishing the highly advantageous technique of Orbital Launch. The work is being conducted in connection with Vought Contract NAS 8-853, and is being guided by Vought work statements and coordinating documents. Every advantage is being taken of the results obtained by Vought in the earlier phases of the contract.

The work included in this volume is an investigation of the various concepts and techniques for communications.

Figure 2-1 shows a model of the desired communications and telemetry system. High quality, reliable, and continuous communications is required between the:

a. OLF and Ground

b. OLF and Spacecraft

c. OLF and Astrotug

d. OLF and Extravehicular Man

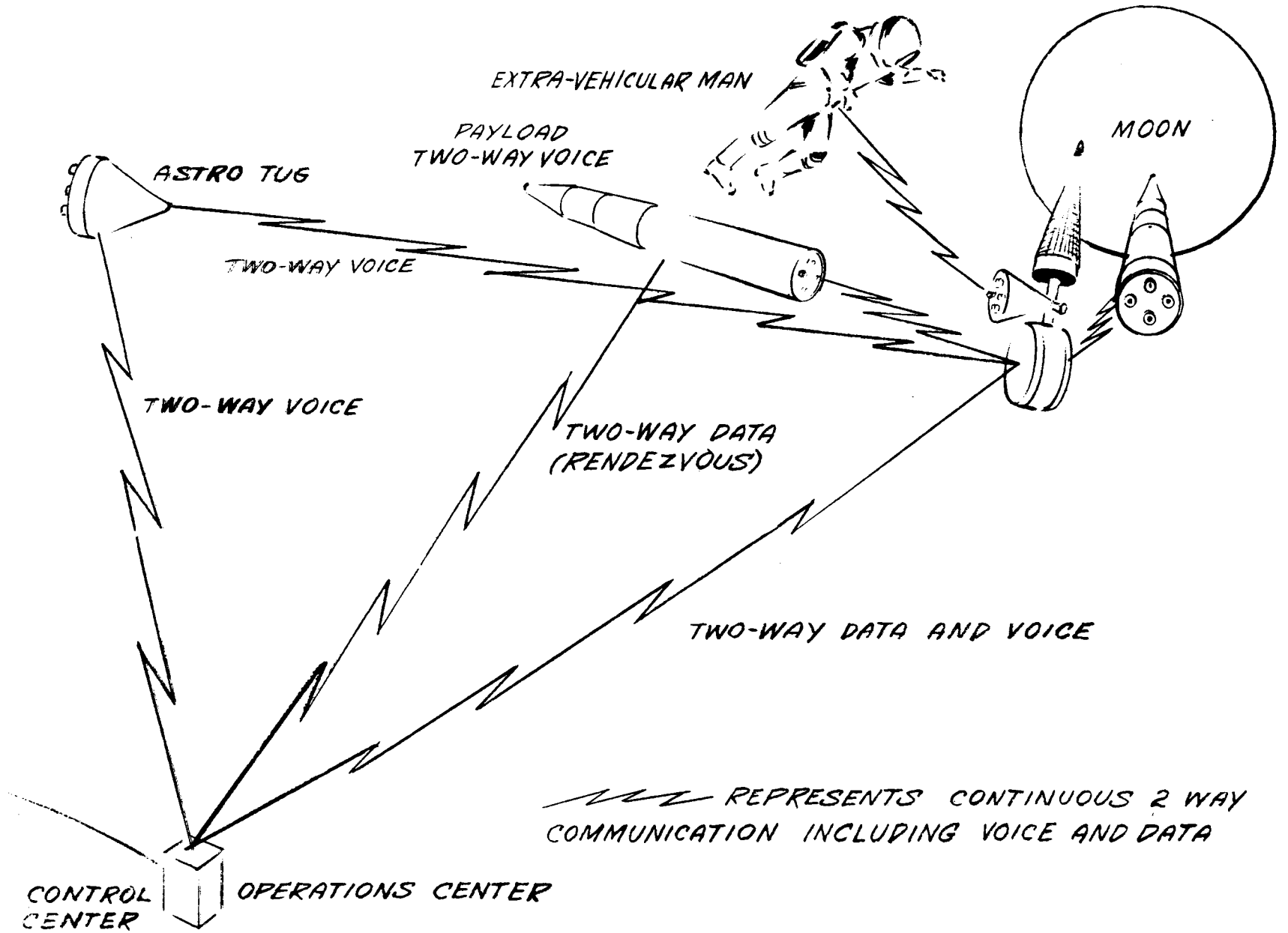


FIGURE 2-1 BASIC REQUIREMENT

This system must provide:

a. 2 way data links and voice channels

b. Data detection, processing and storage

c. Telemetry

Various concepts and techniques are investigated and a candidate system is derived by considering the following trade-off factors:

a. Performance

b. Overall cost

c. Reliability and life

d. Maintenance and Logistics

e. Weight, size, and power drain of spaceborne equipment

f. Schedule

The resulting candidate system that is described in this report has the following outstanding characteristics:

a. High quality, continuous duplex communications between all stations at all times.

b. Only one frequency band for entire system, thus providing:

(1) A minimum number of antennas

(2) Simple logistics

(3) Ease of maintenance

c. Every communications channel provides 50KC or more of bandwidth for multipurpose and flexible use.

This phase of the Astrionic work is being funded by Sperry Rand Corporation as a contribution to establish early the key engineering programs which are necessary to achieve success in this important endeavor. An outline of these key programs is presented in Section I of this volume. A more complete presentation may be found in Sections V, VI, VIII and IX.

SECTION III

BASIC CONSIDERATIONS

3-1. ORGANIZATIONAL PLAN

A discussion of the OLO communication problems is introduced by establishing several concepts by which one may test any conclusions as the work progresses. Following this discussion of concepts the differences and similarities between the OLO communication problems and the communication problems of other systems are outlined. This leads directly into the broad background of the candidate systems itself. The RF considerations will be presented first, followed by the telemetry terminal equipment. After presenting the best available facts and figures on the candidate system commensurate with the total time and effort available to date, the areas needing further study will be discussed.

3-2. BASIC CONCEPTS

a. Line Switched Versus Message Switched Systems

Although in concept several investigators in the same environment may arrive at similar answers to the same problem, it is usually found that the grading of the answer is scaled to the experience of the investigator. For OLO communications it is important to take advantage of the major collective experience of the industry. For example, we may profit from established solutions to the problem areas encountered in our national and international

telephone systems, in aircraft control and other communication systems. These and other multiple user systems have something in common with the OLO operations. A requirement for "real time" operation is a fundamental of voice type operation. However, voice operation is sometimes reduced to non real time by handling recorded messages relayed through from one station to the next on a store and forward basis. On the other hand, there are cases where real time hot circuits are a necessity no matter how inefficient they may be. Fire alarm and police call boxes might be placed in this category. In other cases transmission capability can not be allowed to lie idle and must be shared with other users, for example, normal telegraph systems. The "real time" type circuit for multiple users will be referred to as a line switched system and the non real time circuit as a message switched or store and forward system. There are definite uses for each. However, there are occasions when the proponent of one type of system forgets of the existence of the other. An occasional test of the candidate system to see which system or combination of systems best satisfies the requirements is in order. The results of the subsequent study show the superiority of the line switched system for the OLO communications and as a subsidiary consideration duplex circuits are recommended.

b. Signalling and Supervision

As will be indicated later the OLO communication complex is a multiple user system. Multiple user systems require certain functions that may be forgotten if one has been working in two terminal, essentially single purpose systems. These added functions are signalling and supervision. For this discussion these terms are used in the broadest

sense. Signalling covers the area of addressing as well as the process of alerting the addressee of a call from another user. Supervision is related to signalling in that both may be considered in the category of necessary overhead in order to assure delivery or to inform the originator of progress in delivery. Supervision may be considered as information exchange between terminal points of a circuit that give the condition of the circuit and contribute only indirectly to the text or prime data being transmitted. Supervision includes such information as no-such-address, circuit busy, circuit out of commission, and message accounting numbering systems. These functions have quite a different form depending on whether one considers the line switched system, the message switched system or other systems that might be classed as hybrids or are not well differentiated from the point of view of how the routing is achieved. Very simple systems reduce the signalling and supervisory information to a priori information. As the system becomes more flexible much of this class of information varies from day to day, minute to minute, and must therefore be transmitted at approximately the same time as the text. There is a complete spectrum from bunched supervision once only (a priori) to fully distributed supervision with supervisory elements associated with each binary digit transmitted (error checks or correction). Another variable, on a somewhat orthogonal axis, is the way in which the supervision shares or does not share the path facilities of the text or prime data. A priori information is really one extreme on this axis, that is, certain exchange of information concerning the operation was made without using the circuit in question; the supervisory information did not share the path facilities of the text or prime data. Some portion of the supervisory

information is always handled in this manner. An example of a system that is toward the other end of this axis is a teletypewriter system on the poorest type of transmission line. Almost all the supervision is carried on through the use of spelled out words that are time shared with the text. On the other hand most telephone systems provided some measure of supervision simultaneous with the text although the great majority is time shared with the text. Many military communication centers provide separate circuits for the supervisory information and send address and text only on the main circuits. Attention has been given this concept since it is an area requiring detailed study before equipment specifications can be written.

c. Flexibility and Expandability

Another concept that should be used as a check on a communication system is its flexibility, including expandability. It has been shown that it is economical to plan for the future in communications and use a building block principle wherever possible. Can one satisfy one user today and another slightly different user tomorrow without completely rebuilding? In this regard one observes that there is little to distinguish between communications and telemetry, it is all data transfer. The capability used in communicating between machines needs little change to be able to exchange data between men. In the case of the OLO, there is a definite need for freedom in expansion. All of the requirements of 1967 cannot be forseen today.

d. Step Forward

A further concept that is a bit difficult to define is worth considering. That is, after investigating

utilization of existing concepts, equipment and installations and finding them inadequate one must think ahead somewhat beyond the solution of the immediate problem to be sure to take a step forward that is great enough to be worth the effort in breaking away from the more conventional scheme. In testing, one should perhaps go to a limit, even if absurd, and then retrench to a reasonable position.

3-3. DIFFERENCES BETWEEN OLO COMMUNICATIONS AND OTHER COMMUNICATION SYSTEMS

a. Multi Station Complex, Continuous Communications

So much for basic philosophy, now consider what major differences and similarities there are between the OLO communications and other communication projects. Of course there are other space communication problems that are quite similar but most of those to this date do not include several "birds" of various natures in simultaneous and cooperative operation in orbit. Another difference between this and other space problems that can be rather severe in its consequences is that of continuous communications over an extended period of time (years). This follows the reasonable assumption that the OLF is to be permanent and not consumed by launch of the OLV. Both parts of this statement are differences from existing techniques. First, continuous communications implies a need for "real time" communications which has not been available previously. Second, the idea of having men aloft continuously for years places a permanence on the communication facilities that can alter the approach considerably over that of a one-shot approach. Reference to the figure 2-1, showing the overall communication complex, immediately demonstrates that there

may be several "birds" in operation simultaneously and provisions must be made for the concepts of signaling and supervision; for example, addresses are required that are not required in two-point communications.

b. Continuous Communications Vs. Continuous Tracking

Truly continuous communications is a severe requirement and requires a forthright attack that may not be a mere extension of the facilities used for other programs. For the present discussion, the requirement for continuous communications is divorced from that of continuous tracking of the OLF. In the event that continous tracking should be found to be a system requirement, the conclusions reached in this discussion would be somewhat modified. Reference will be made to the area requiring modification after the discussion excluding the possible modifications have been completed.

SECTION IV

CONTINUOUS COMMUNICATIONS, GROUND TO OLF

4-1. REQUIREMENTS FOR CONTINUOUS COMMUNICATIONS

Continuous communications requires at least one continuous channel and preferably several between the OLF and ground, the ground point being connected with the operations center with a set of channels of the same capacity as that from ground to OLF. Appendix A gives some of the details of this configuration as a simplified network. This provides background details for the difficult problem of continuous ground to OLF communications. Further, it requires at least one channel either to ground or to the OLF from any other "bird" in the near range of the OLF. For truly continuous operation this means that each of these circuits must be duplex so that one cannot lock another out by extended one-way use of reversible simplex. Appendix B gives an expansion of the terms simplex and duplex for those that may not be familiar with this terminology. It also explains the lock-out danger mentioned above. Simple switching is implied between the above circuits connecting the operations center to the OLF and interconnection at the OLF with the other "birds" in its immediate vicinity. Some of the network control problems which may be encountered in interconnection are indicated in Appendix C. This important concept of network control has been detailed in Appendix C since it is rather lengthy and may be well known to some of the readers. Since it will be demonstrated below that there is considerable expense involved in the OLF to ground link, a few of the highlights to justify

the need for continuous communications must be given. First, it may be argued falsely that no communications is required. Everything can be programmed as a sequence of operations clocked out of storage. This would be a true statement if everything were known to the desired degree of accuracy and reliability. It is definitely not true when human failures, equipment failures, weather changes, and other occurrences of unknown or only statistically known character are accounted for. Then one may say that in "rule by exception" one needs only to transmit the corrections to the stored program required by failures or insufficient knowledge and consequently only a very small portion of the total operation is consumed in sending corrections and discontinuous communications are satisfactory. Again this statement is probably valid so long as the variables in the program change smoothly, but may be false for step function changes to the program such as may be introduced by catastrophic failures of man or machine. The OLO is a very complicated operation involving the coordination of many, many functions and people and a failure to be able to transmit a "hold" just before launch of new crew members might require a $100,000,000 abort due to a failure in the OLF. One takes a calculated risk in assuming that any given time may be allowed for interruption of communications. In view of the length of the total operation and the total number of launches required, it becomes increasingly risky to leave holes in the communications. Another reason for communications continuity may be drawn from experiences at sea or in the air. It concerns the logging of information about the voyage. For any voyage in which the log book is salvageable from the wreckage, a self-contained log is satisfactory. However, remote logging is a necessity if one has any reason to believe that circumstances may obliterate the information in the self-contained log. If the instant of obliteration is not predictable,

continuous entries into the remote log are desirable. If it is believed that circumstances leading to the catastrophe might prevent similar catastrophes in the future, one has a rather reverent respect for the concept of an adequate log book. For this purpose, the word "log" has been used in a very broad sense and includes information entered in the records by the machine complex as well as by the men aboard.

After justifying continuous communications on other grounds, one finds that there are many bonus effects accruing after the basic objectives are satisfied. Operational convenience is one of the simplest to understand. Instead of restricting certain functions of conferences to an erratic come-and-go timetable, scheduling can be done without restriction.

4-2. EXTENSION OF EXISTING FACILITIES

In order to determine the magnitude of continuous communication type operation, the following brief analysis is given based on an extension of existing ground facilities. The chart of figure 4-1 gives the time-in-view of the OLF for ten selected locations on the earth. Fifteen consecutive orbits of the OLF are shown which very nearly close after 24 hours. On the first orbit the time-in-view for Cape Canaveral is 9.1 minutes, for Canary Island it is 1 minute, etc. Near the bottom of the column are given the times-out-of-view for over 8 minutes. There are other outages not shown less than 8 minutes. At the bottom of the page are the total times-in-view for each orbit. For the locations chosen, path ten is the worst, where the time-in-view is 8.3 minutes and the time-out-of-view is 88 minutes. Subsequent orbits 16 through 30 do not exactly repeat but there are enough samples given here to give a representative picture of what may occur in any 24-hour period. The locations were selected with some care

but even so are probably only representative, not optimum. Disregarding the distribution of in-view versus out-of-view, one finds that during 30.8 percent of the 24-hour period the OLF is in view of the ten ground stations. From this it may be seen that many more stations are needed to affect continuous coverage. To see how many stations are required, refer to figures 4-2, 4-3 and 4-4. Figure 4-2 shows that a 350 mile OLF can see the point 1420 miles away from the point directly under it, assuming 4/3 earth radius and minimum elevation 5 degrees above horizon. Figure 4-2 shows that the OLF at 28 degrees North and 350 altitude could see a satellite at the same longitude and altitude at the equator. This fact will be referred to later. Figure 4-3 shows how individual coverages may be overlapped to fill in the deep cusps. At 2400 statute miles between stations cusps fall to ±11 degrees of latitude from the line of centers. Figure 4-4 shows how two rows overlapped the same amount in both axes as figure 4-3 are inadequate for ±28 degrees latitude coverage and how three rows are more than adequate. As a matter of fact, three rows allow for some inaccuracies in location. This shows that 36 stations uniformly spaced will be the maximum number required. This number gives an opportunity to estimate the costs of such a ground complex.

4-3. COST OF GROUND EXTENSION

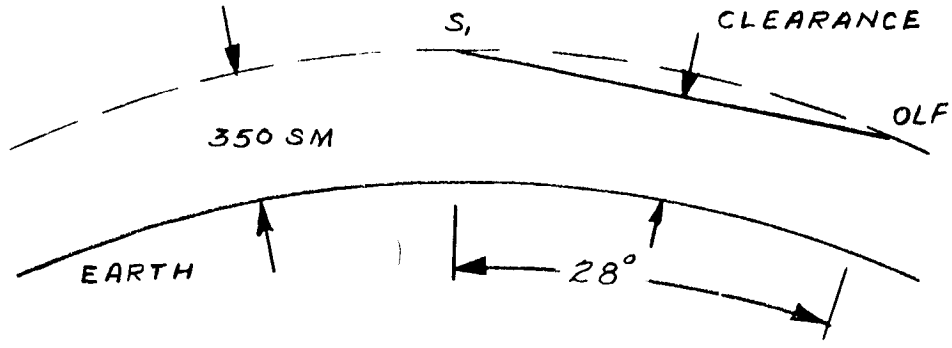
Of these 36 stations, a very rough estimate allows for 10 existing stations, 13 on land, and 13 at sea. Considering that whatever economies there are in outfitting existing stations will be absorbed in the stations at sea, one may see that to an order of magnitude the cost of ten years' operation of a ground net of 4kc bandwidth is $115,000,000. Figure 4-5 shows this estimate. Although this estimate needs refinement,

Station / Orbit Period Number λ = Long.(Deg.) ø = Lat. (Deg.)	Continuous Time in View (Minutes)														
	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
Cape Canaveral, Florida 80,48W 28,48N	9.1	9.1	7.3	-	-	-	-	-	-	-	-	-	-	7.8	9.1
Canary Island (near)* 15, OW 28,48N	1.0	-	-	-	-	-	-	-	-	-	7.0	9	9.1	8.9	7.8
Ascension Island 14,41W 7,95S	-	8.7	8.0	-	-	-	6.2	8.6	5.5	-	-	-	-	-	-
Pretoria, South Africa 28,28E 25,75S	6.6	9.0	9.1	9.1	8.9	5.8	-	-	-	-	-	-	-	-	-
Mauritius Island (near)* 60E 20S	8.6	7.7	8.2	9.2	7.0	-	-	-	-	-	-	-	-	6.0	9.0
Woomera, Australia 136,79E 31,10S	6.0	-	-	-	-	-	-	-	-	-	-	7.0	8.7	9.2	8.6
Volcano Island (near)* 145E 27N	-	-	-	4.5	8.6	9.0	9.0	8.7	6.0	-	-	-	-	-	-
Hawaii 159,63W 22,16N	-	8.2	9.0	8.5	8.5	9.0	6.8	-	-	-	-	-	-	-	-
Society Island 150W 20S	-	-	-	-	-	-	-	7.3	9.0	8.3	7.6	8.5	9.0	3.8	-
Point Mugu, California 119,15W 34,12N	6.3	8.3	8.3	7	-	-	-	-	-	-	-	-	-	-	-
Continuous TOV values (≥ 8 minutes)	11 15 11 20	11 26	14 25	36 15	44 13	46 21	37 30	14 29 30	15 28 31	88	55 26	37 11 23	35 10 23	18 12 13 12	16 11 27 8
TOTAL TIME IN VIEW	37.6	51.0	49.9	38.3	33.0	23.8	22	24.6	20.5	8.3	14.6	24.5	26.8	35.7	34.5

*Ship Stations

Grand Total
TIV MIN
4451

Figure 4-1 Earth-Based Tracking Coverage of OLF for a 24 Hour Period

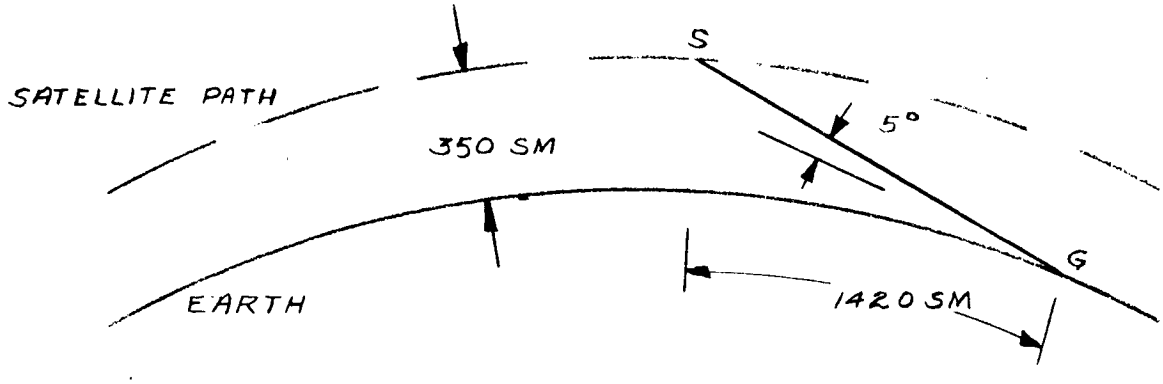


OLF TO EQUATORIAL SATELLITE PATH (OLF AT MAXIMUM LATITUDE)

S, - SATELLITE
OLF - ORBITAL LAUNCH FACILITY

A

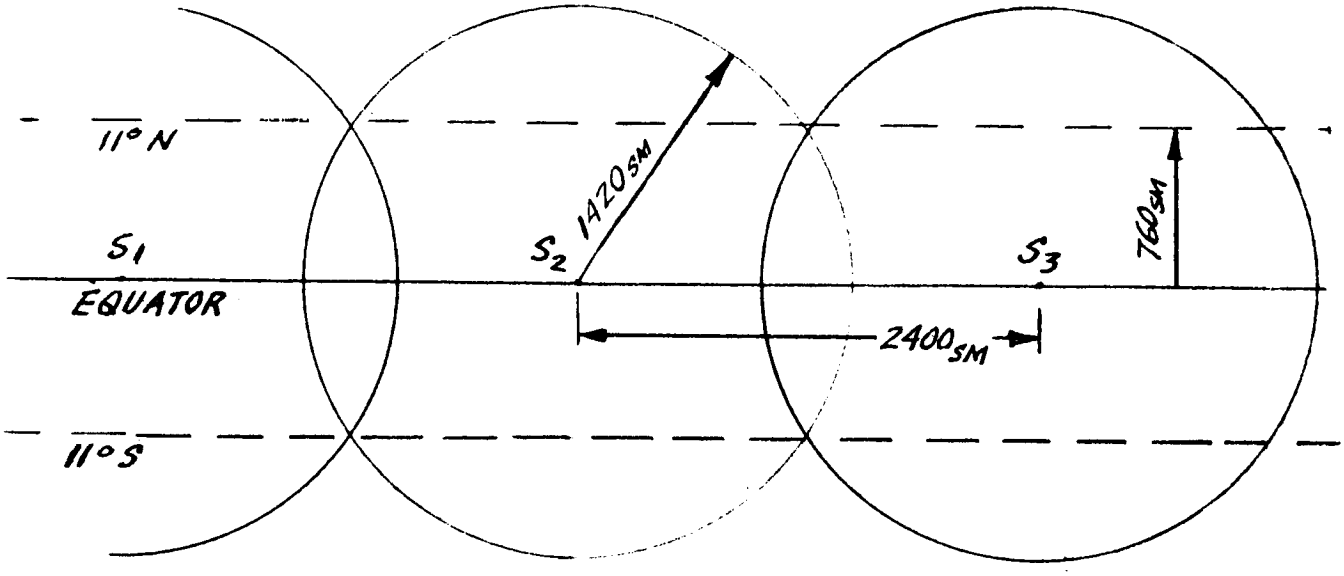
S - SATELLITE
G - GROUND



SATELLITE TO GROUND PATH
350 M ALTITUDE
5° ELEVATION ABOVE HORIZON

B

FIGURE 4-2 MAXIMUM LINE OF SIGHT PATH



S_1 SATELLITE 1
S_2 " 2
S_3 " 3

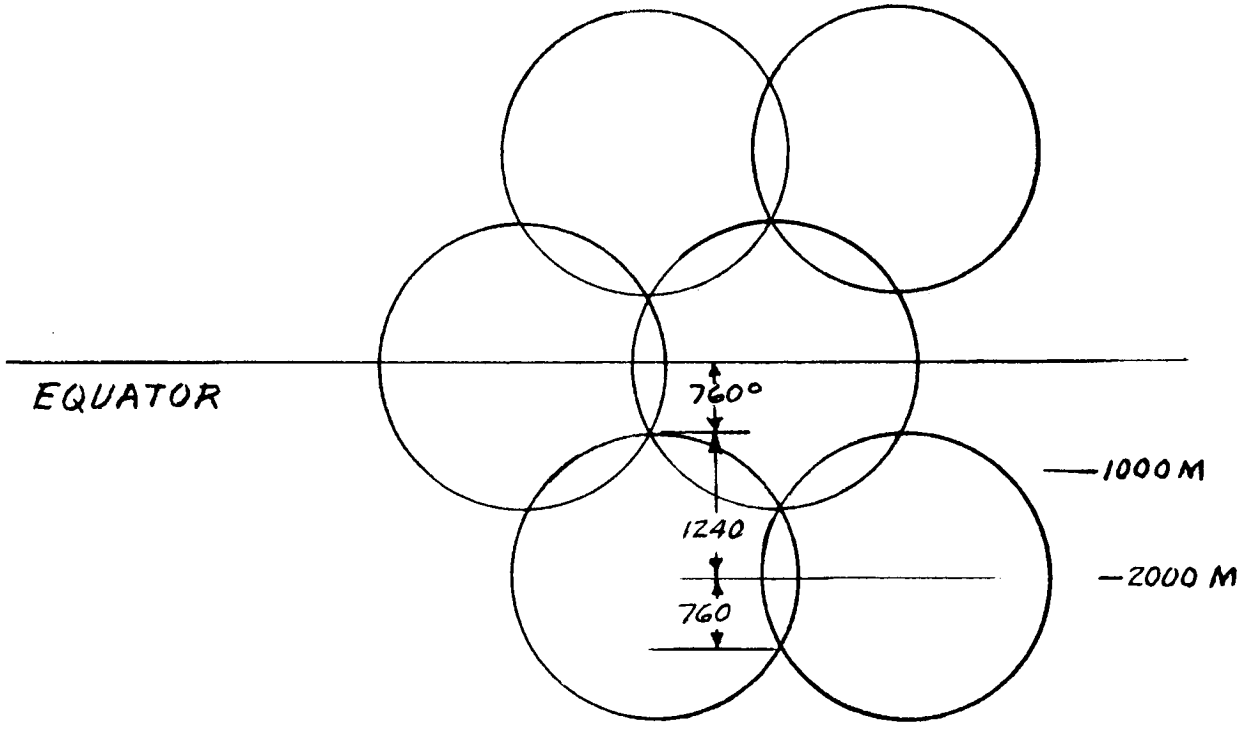
5° VIEWING ANGLE ABOVE TANGENCY

350 S. MILES ALTITUDE

2400 SM SPACING

FIGURE 4-3 EQUATORIAL SATELLITE RING GROUND COVERAGE

TOTAL 36 STATIONS
~ 10 EXIST
~ 13 LAND
~ 13 SEA



12 IN ONE ROW
2 ROWS FALL SHORT
3 ROWS MORE THAN ADEQUATE
(ALLOWS FOR CONSIDERABLE
INACCURACY IN LOCATION)

FIGURE 4-4 PATTERN FOR CONTINUOS COVERAGE

it shows that alternate schemes may be considered that cost on the order of hundreds of millions if they do a better overall job, such as more bandwidth, fewer propagation outages, or more operating convenience.

	MILLIONS
36 STA at $1,000,000	$36
LEASED LINES (10 YRS) 36 STA 5,000,000 MIN at $5	$25
10 MEN/STA 10 YRS at $15,000/YR	$54
	$115 MILLION

Figure 4-5

Rough Estimate of Communications Ground Net Costs

4-4. USE OF SATELLITES

This leads one to consider the use of satellites. At first glance one assumes that shared use of communication satellites would be very economical. Further study shows that this may not be practical except as a means to connect together ground stations. However, a few limiting conditions may be understood from figure 4-6. The earth is well covered by the view from three high-flying satellites at 22,300 miles in stationary orbits. Without discussing the relative merits of the high-flying satellites one may turn momentarily to what other possibilities exist. A lower limit in altitude appears to be the height of the OLF orbit. As indicated in figure 4-3, such satellites would give a band 22 degrees wide around the earth. These must be mutually synchronous as must the 22,300 mile satellites. Two other possibilities immediately suggest themselves. Mutually synchronous orbits between 350 and 22,300 miles and random orbits, both polar and

KIND	QTY	REMARKS
22300M	3	ABSOLUTE MIN. QTY
350 M	12	MIN. QTY FOR COMM WITH FENCE OF 3 GND STA
350 TO 22300 M	3<N<12	
7000 M	40-50	RANDOM

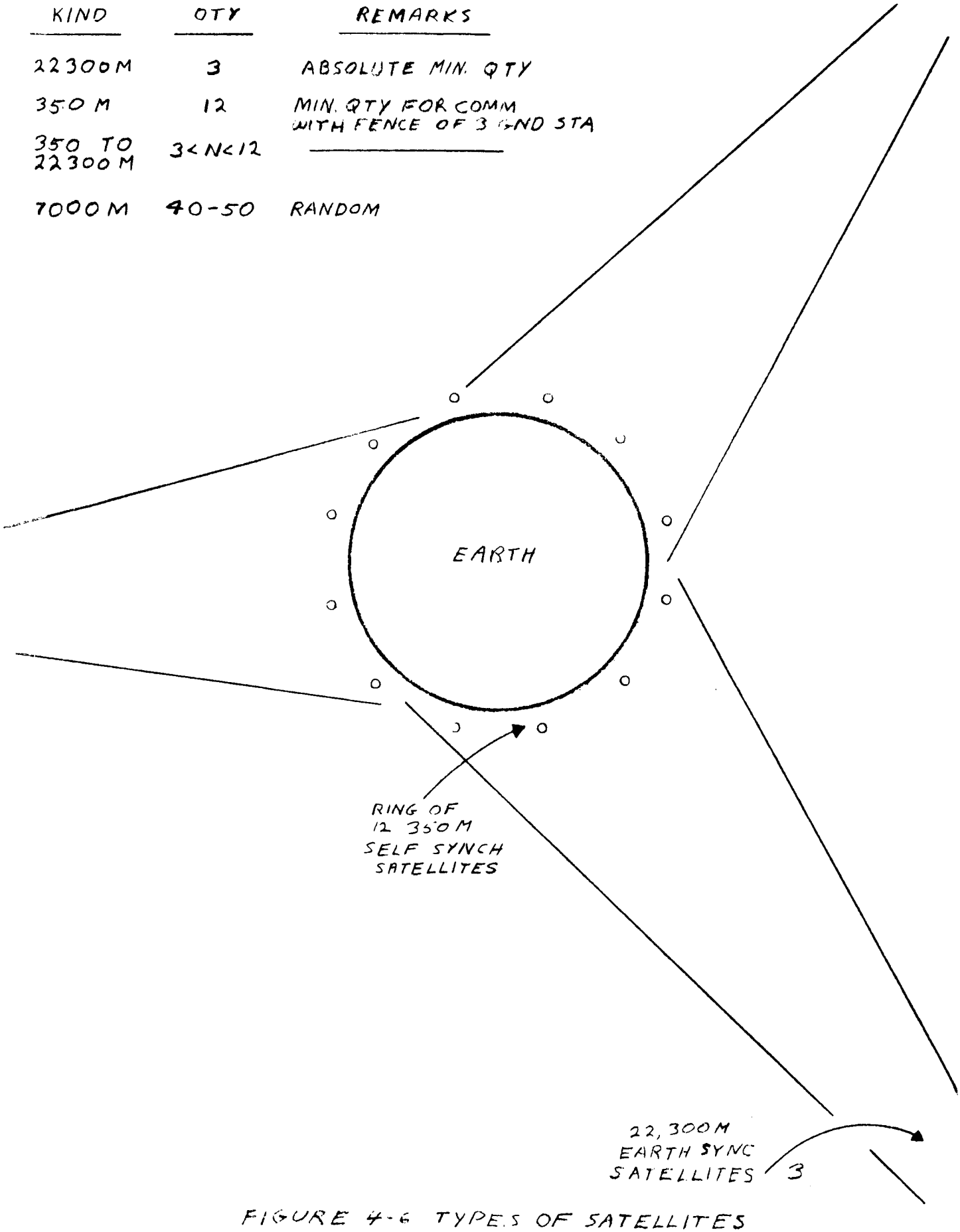


FIGURE 4-6 TYPES OF SATELLITES

equatorial at various altitudes. Figure 4-7 gives a few comparative characteristics of three types of satellites. Intermediate synchronous satellites have been omitted for the sake of brevity. The line discussing random satellites has not been fully completed since a check with Bell Labs indicates that OLF use of random satellites would probably require shutting down a pair of the ground stations that were originally designed to use the satellites. In addition, the acquisition and track column indicates that the random satellites would require considerable information about where to look next transmitted to and stored in a computer in the "bird" and upon loss of this information the system would have difficulty in learning how to heal the damage (loss of communications). The many coordination difficulties associated with random satellite systems indicate that the emphasis should be placed on comparing low-flying satellites versus high-flying satellites.

4-5. COMPARISON OF 350 MILE AND 22,300 MILE SATELLITE SYSTEMS

In order to make some comparisons of these two satellite systems, one must establish firm ground rules. The single rule that can change the picture more than any other has already been discussed but should be re-emphasized--continuity of communications.

It makes little difference in the following discussion whether one assumes shared facilities or new parallel facilities, certain masses must be injected into orbit to do this job. Sharing is particularly difficult if one considers that most satellite systems assume nonsymmetrical configurations with simple gear in the satellite and relatively sophisticated gear on the ground, weight and size

being more or less directly related to sophistication. If one wishes to utilize the satellite to communicate with another satellite, this nonsymmetrical system fails to bear dividends. Therefore, this analysis does not exactly parallel a ground-to-ground system. An accurate analysis should consider the cost of injecting the masses into the given orbits. A relative measure of this is given by the differences of the masses that must leave the ground. Assuming a ΔV of 7,000 ft/s to be required to go from a 350 mile orbit to 22,300 mile, it can be shown that with a solid fuel of $I_{sp} = 250$ the ratio of total mass to mass of payload plus structure is about 2.5.

Normalized weights will be computed with reference to the chart of figure 4-7. The least common denominator is the weight required to provide a given bandwidth at a given signal to noise ratio at the longest range in line 1. The maximum path length for line 1 is 2400 miles satellite to satellite (the 1600 miles slant range to ground does not govern). The weight of the equipment required to terminate one end of the 2400 mile path will be taken as unity. The 22,300 mile satellite has a maximum path length of its altitude plus one earth radius or 26,300 miles. Since the power required goes as the square of the distance, the power for line 3 is $\left(\frac{22,300}{2400}\right)^2$ or 121. Since the weight may be considered to go directly as the power (for this power range and continuous power), the ratio of weights per path is 121 to 1. For line 3 there are four installation aloft, three satellites and the OLF at mass 121 against 12 satellites plus OLF at unity mass or 13. Accounting for the difference in altitudes and utilizing the mass ratio of 2.5 on the three 22,300 mile satellites gives the ratio 13 to 1029. This is not the correct number as will be

Type	Qty	Pwr	Unit Mass Per Path	Nr of Terminals	Mass X Nr. Terms	Acquisition & Track From OLF	Mass Ratio	Rel. Mass Y Mass Ratio Adjusted
Synchronous 350 MI	12	Low 1	1	12 + 1	13	Simple 1 Axis ±15°Max	1	13
Random 7000 MI	~40	Med	>1 <121	~40	Inter-Mediate	Complex 2 Axis ±90°	>1 <2.5	Complex
Synchronous	3	121	121	3 + 1	484	1 Axis ±90°	2.5	1029

Figure 4-7

Comparison of Characteristics of Three Types of Satellites

shown later since each of the 350 mile satellites must be capable of looking in 4 directions. This yields (12 x 4) +1 or 49 versus 1029 or a ratio of 21. This is quite a large number and shows that further attention must be given the 350 mile satellites to determine what details of implementation are required.

4-6. NON-COPLANAR SATELLITE

The most attractive low-flying satellite is one at the same altitude as the OLF but in a non-coplanar equatorial orbit. A single satellite of this kind could aid the ground connection problem materially, since theoretically 12 ground stations at the equator would be in contact with it 100 percent of the time. Such a satellite could see the OLF at all times if maintained abreast of the OLF in an orbit of equal period, as shown in figure 4-2A. Such a concept may be extended to 12 satellites rather than 12 ground stations but this concept was rejected due to the difficulties in equatorial launching and maintenance.

4-7. COPLANAR SATELLITE

A derivative of the above system is a satellite in the path of the OLF. One leading and one lagging satellite spaced 30 degrees from the OLF, used as relays to each ground station, is an interesting concept. It triples the time in view over each station on the ground.

An obvious extension of this scheme is a limiting procedure. Extend these OLF relays to a ring of 12, one every 30 degrees. This reduces the number of ground stations required to a minimum of 3 placed in a north-south fence.

Appendix D develops practical hardware for the configuration as depicted in figure 4-8. The OLF is shown at the upper right in contact with the ring, numbered 1 to 12 and the ring in contact with the ground fence shown between 10 and 11. The ground station in contact with 1 is not part of the prime configuration, but gives direct contact to a given station, say the Cape, for certain intervals of time. A practical fence may include more than three stations. Figure 4-9 shows a fence using as many existing stations as possible, Antofagasta, Lima, Quito, Panama, Jamaica and the Cape.

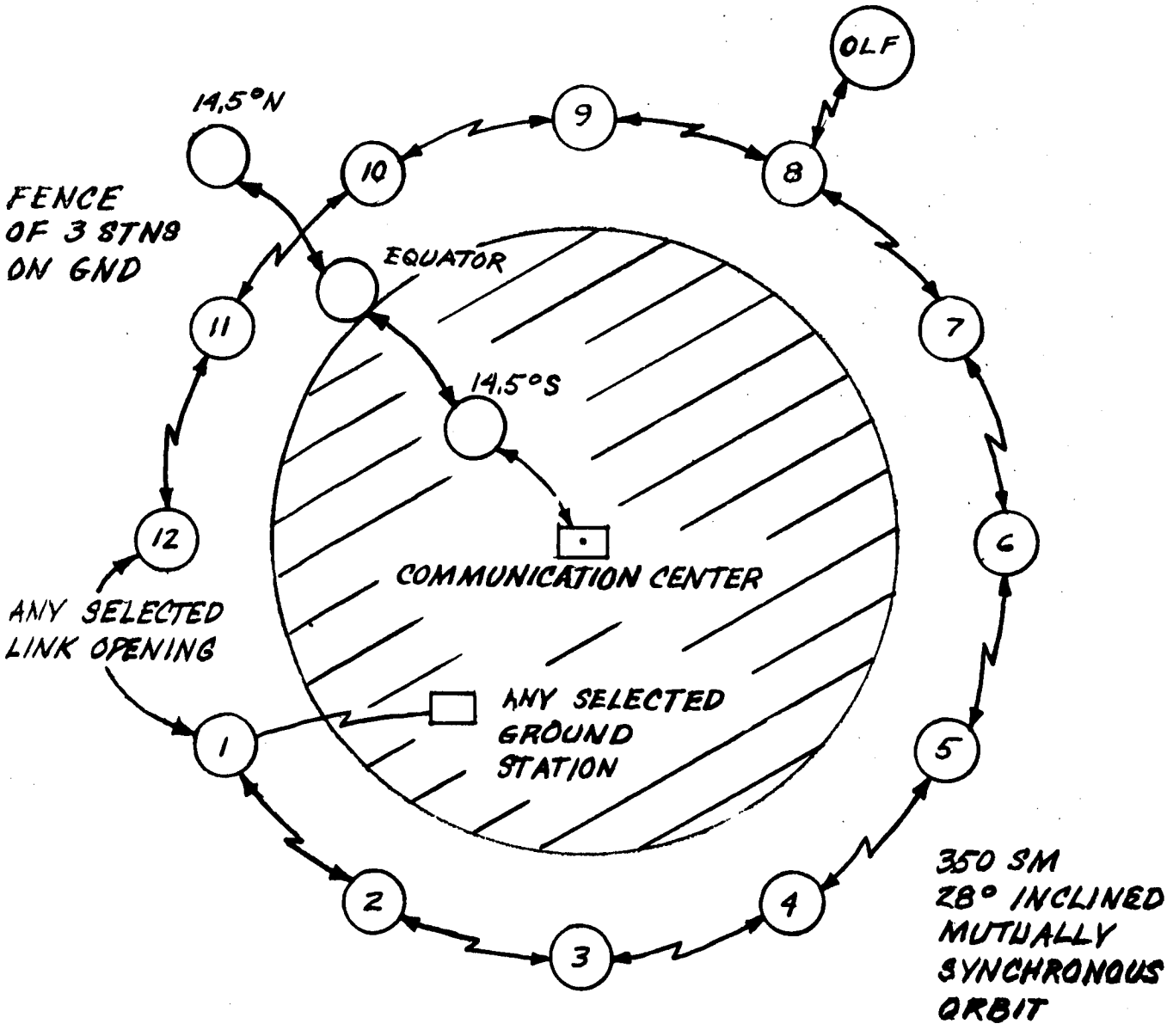


FIGURE 4-8 OLF- SATELLITE RING-
GROUND FENCE- CONCEPT

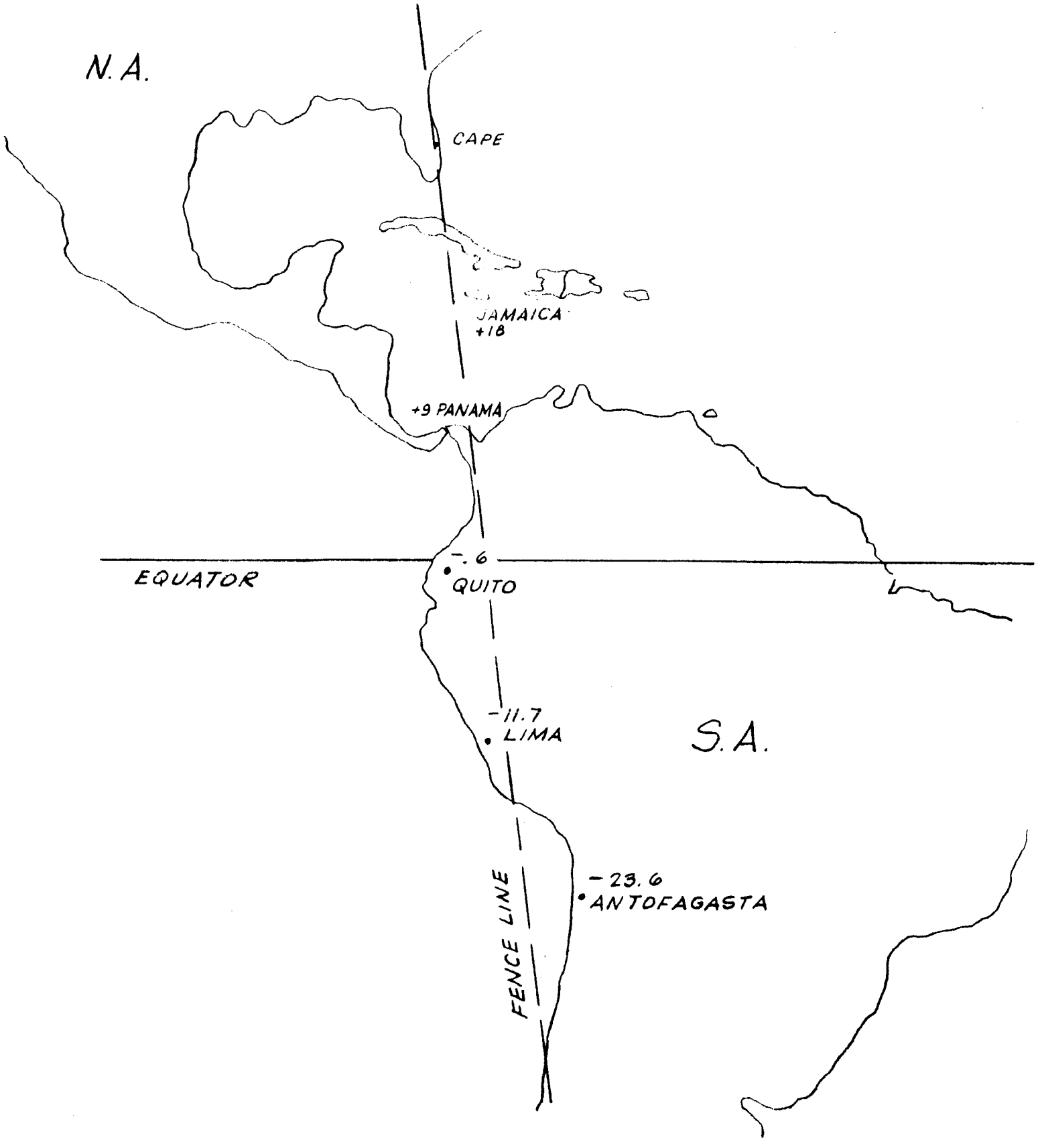


FIGURE 4-9 NORTH-SOUTH FENCE GEOMETRY

SECTION V

CANDIDATE OLF TO GROUND SYSTEM

5-1. SATELLITE RING

Having established a possible means of continuous coverage through satellite relays it is of interest to have a look at such a system in further detail. First, the chances of launching such a system on time with sufficient reliability to meet all the OLO requirements is proven by the basic ability of launch and rendezvous OLO components of all kinds. The total weight of 12 satellites is on the order of 6,000 lbs which could be boosted with other equipment to the OLF and subsequently each satellite may be given a small boost, of ΔV of about 50 ft/sec, at regular intervals in order to space the satellites ever 30 degrees around the earth. This would form a ring which slowly rotates with respect to the OLF. Directive antennas have been used on the satellites since the complication of orientation is considered well worth while in order to be able to achieve relatively wide bandwidth at a good signal to noise ratio. The components of the candidate system may be summarized with reference to figure 4-8.

5-2. GROUND FENCE

The ground fence is expanded to 6 stations each having UHF equipment looking up to the ring and HF equipment up the fence to the Cape as further described in Appendix D.

The block diagram of figure 5-1 shows the equipment required in each of the 12 satellites of the ring. There are four sets of duplex radios in each satellite, one looking forward in the ring, one looking backward in the ring, one looking up to the OLF, and one looking down to the ground fence. All the equipment is in the 2300 mc band. Eight foot dishes are used for the paths around the ring, while a two foot dish is adequate up to the OLF and a slot antenna is adequate looking toward the ground. The one watt transmitters for transmission around the ring are operative all the time. The 4 watt transmitter looking to the OLF and the 2 watt transmitter looking to the ground operate on a 10 to 20 percent duty cycle. The calculations leading to the configuration of figure 5-1 are detailed in the Appendix E. The OLF equipment for connection into the satellite ring is shown at the top of figure 5-2. This shows an eight watt transmitter and a parametric receiver operating with an eight foot dish. Calculation of this circuit is also given in Appendix E. The remaining three circuits of figure 5-2 are discussed in Section 6.

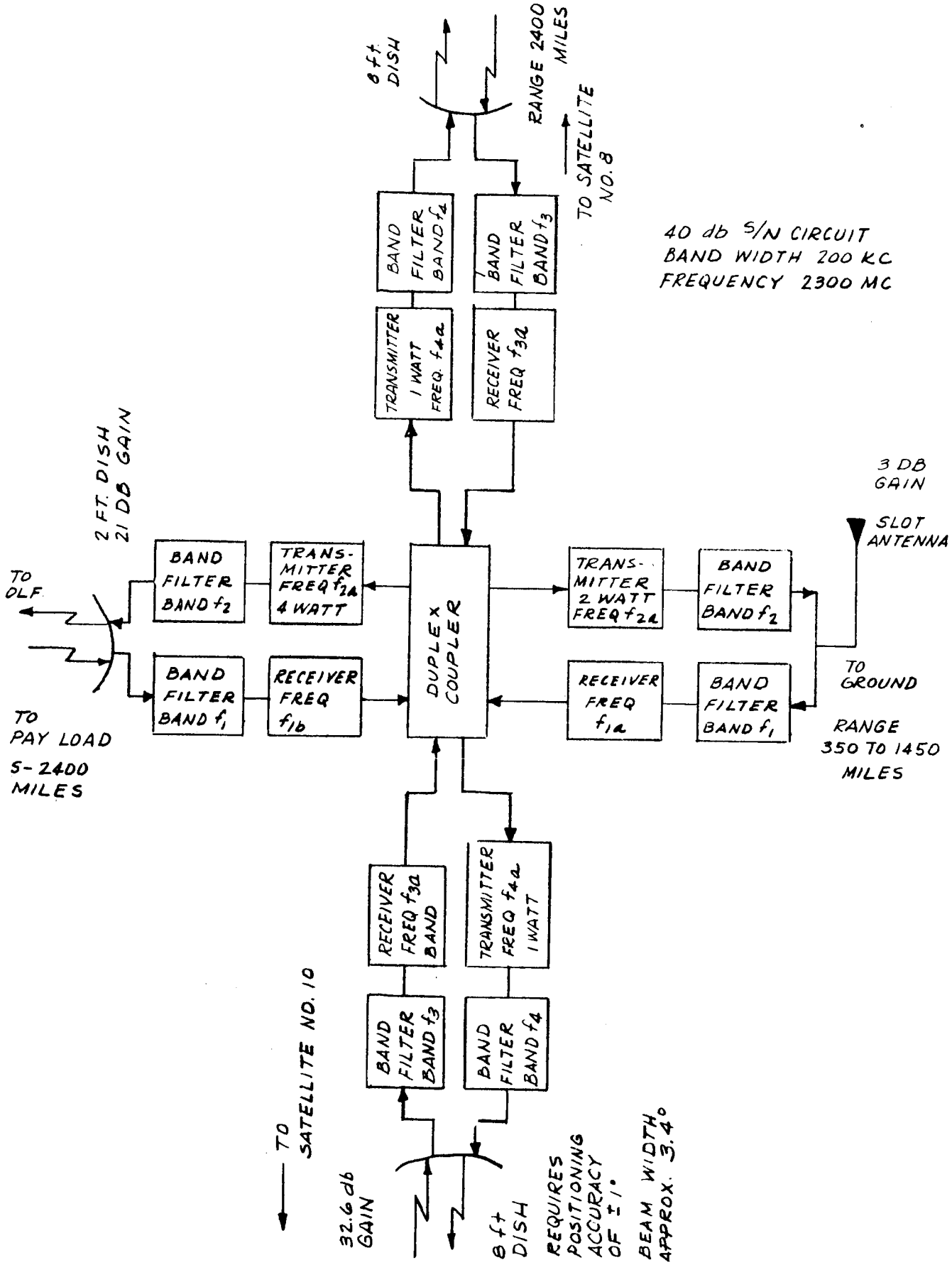


FIGURE 5-1 TYPICAL SPACE RELAY SATELLITE

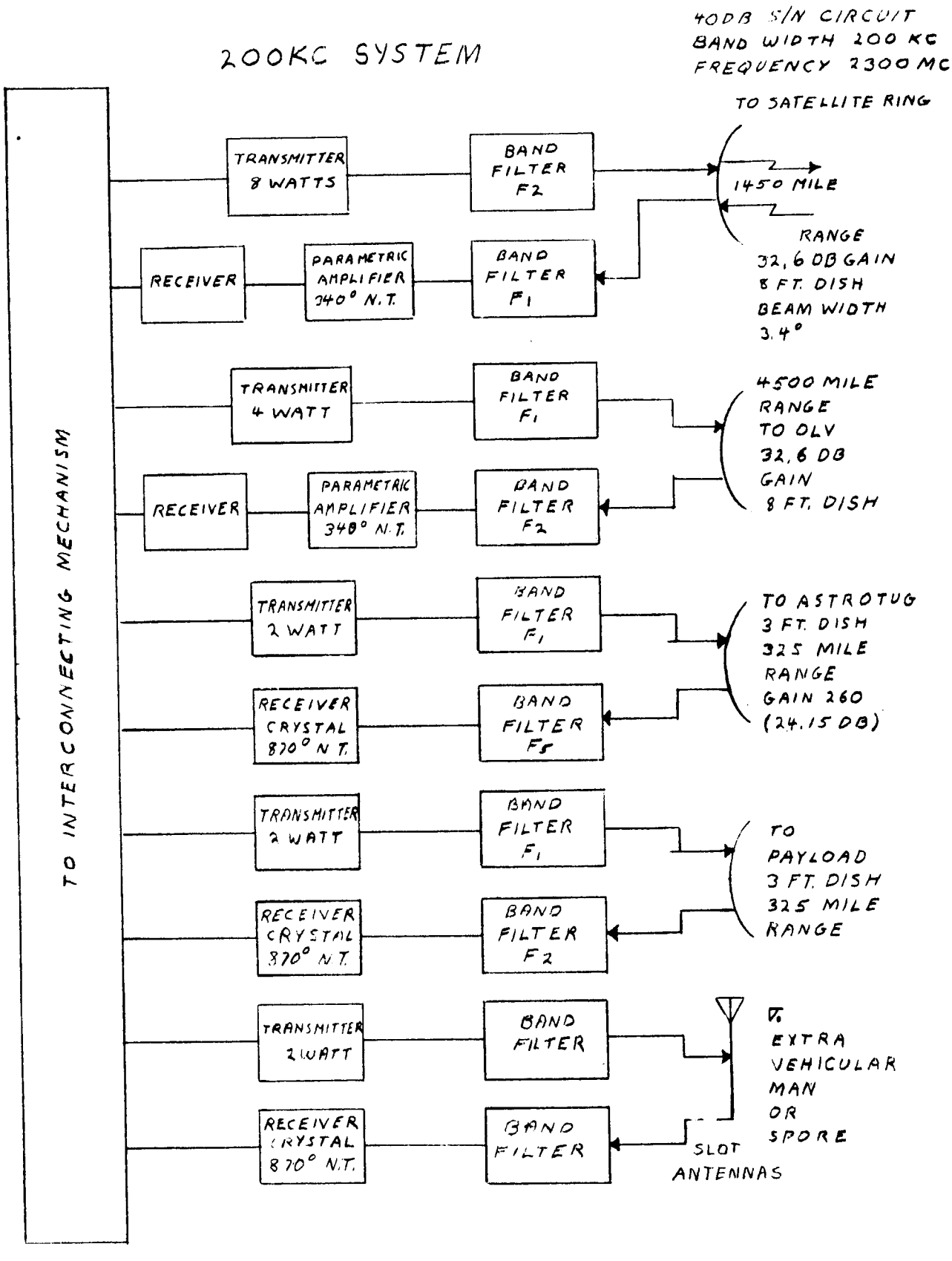


FIGURE 5-2 OLF EQUIPMENT

SECTION VI

OLF R-F EQUIPMENT

6-1. R-F EQUIPMENT

The OLF equipment has been summarized in figure 5-2. Supporting calculations are given in Appendix E. All of this equipment is in the 2300 mc band and requires 2 8-foot dishes and two three-foot dishes. Although every attempt should be made to keep the powers and receiver noise temperatures the same in order to increase the flexibility and built in redundancy (at reduced capability) one eight-watt transmitter, one four-watt transmitter and a pair of two-watt transmitters have been selected. Antenna and frequency switching should be provided, however, so that even the less sophisticated equipment could be placed on the most important circuit with somewhat reduce performance. In addition the modules of these equipments should be the same, for direct interchange in the case of failures.

6-2. CIRCUIT TO SPACECRAFT (S/C)

The circuit to the S/C is similar to that to the satellite ring except that not quite so much power is required for greater range since symmetry has been observed with the S/C, whereas in the case of the satellite ring a non-symmetrical circuit was utilized to keep the 12 satellites simpler and lighter. The range of 4,500 statute miles was selected to avoid the shadow area that might occur before the S/C would be high enough to be received by the

DSIF net. The geometry of this consideration is shown in figure 6-1. This figure shows the earth, the path of the OLF at 350 miles altitude, the path of the S/C up to 6000 miles altitude, and the near triangular cross-sections of shadow volumes that are below the horizon of the DSIF stations. Letters on the two paths, a to q, show the relative times so that relative look angles may be estimated. For simplicity the DSIF stations have been shown in the plane of the paths spaced 120 degrees apart. This approximation is close enough for the purposes of this discussion. The figure shows the DSIF shadow area covering the first 2000 miles of the path. This has no particular significance since the injection point might have any position relative to the DSIF stations. It may be seen that the DSIF shadow areas could be coincident with the first 4000 miles of the S/C's path. To eliminate the possibility of this problem the S/C may maintain communications with the OLF for the first 4500 miles of altitude and at this point transfer control to the DSIF net. This approach presents several advantages in that it reduces the required S/C communication payload and the required capabilities of the NASA ground complex while allowing a single station to maintain control of the S/C during initial powered flight.

6-3. CIRCUITS TO ASTROTUG AND PAYLOADS

The circuits to the Astrotug and payloads are identical and both are similar to the longer haul circuits. The range of 350 miles is a compromise between simplicity of equipment and early continuous communications for rendezvousing "birds" in the vicinity of the OLF. These might not be simultaneously required, particularly that

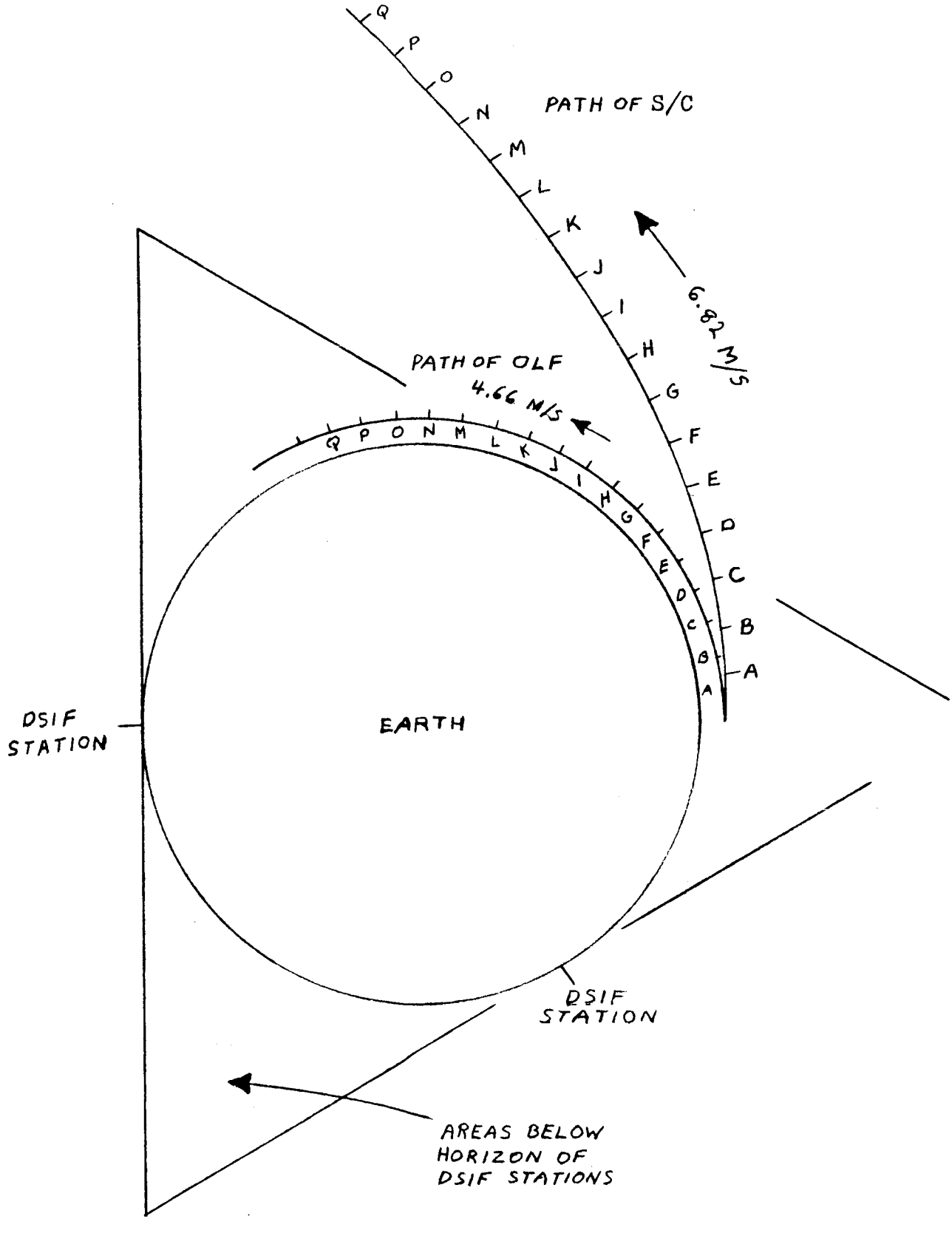


FIGURE 6-1 PATH OF S/C RELATIVE TO PATH OF OLF FOR ALTITUDES BELOW 6000 MILES

to the S/C, in which case the unused circuit should be considered as a spare. Again this points out that frequency agility and input/output switching are a firm requirement.

6-4. CIRCUITS TO THE EXTRA-VEHICULAR MAN

The communications to the extra-vehicular man may be handled primarily by telephone. It is assumed that there are safety lines to each of these men much like the man in the diving suit under water. However, for emergency, or in the case where he wishes to free himself of the hinderance of such a line radio should be used. This should not be in the low frequency bands, VLF through VHF, since very strong interfering signals may be received from the ground only 350 miles directly underneath. For this reason the UHF range has been selected once again. The fact that duplex operation may be obtained more easily at the higher frequency (since sharp filters are smaller in size) and compatibility with the rest of the system makes one more interconnection capability without added equipment if time sharing is allowed when other facets of the operation are shut down. A 10-milliwatt transmitter and a 30-db receiver is more than adequate for the job. The OLF equipment is the same as that for communications with the Astrotug or payload. A simple multiplexer can allow simple bio-medical data to be run on this circuit continuously if required, since the duplex capability is available. Duplex is also very important for the reason stated much earlier, that of prevention of unintentional monopolization of a reversible simplex circuit.

SECTION VII

MODIFICATIONS OF SATELLITE RING FOR CONTINUOUS TRACKING COMMUNICATIONS

As was mentioned earlier the study so far has separated the continuous communications problem from the continuous tracking problem. It is interesting to consider what use may be made of the satellite concept as proposed herein to send radar tracking data back to the central computer. Assume that a considerable number of radar stations widely dispersed throughout the world are made available to the project. These may be ships, or added land-based stations in order to get the necessary coverage. These stations individually have a limited use factor since, even if they are at the proper longitude, only less than one third of the OLF passes will be visible due to the changes in latitude of the OLF. Even though the satellite ring proposed is also only in view at any selected spot on the earth for the same percentage of the time, this portion of the time brackets the pass, or several passes of the OLF. This means that the satellite ring may provide a path which does not provide continuous communications to any given ground station back to the Cape, but is available as much as an hour or more before and after the given OLF pass. This is exactly when the data is generated that must be handled in real time or near real time. Some of the available bandwidth of the ring could be diverted for this job. The information would be available for both the operations center and the

OLF, if the OLF were to have sufficient computing capability to absorb the information. The total bandwidth can be divided with rather simple passband filters since there would never be more than one station, ground or OLF, simultaneously entering a single receiver in the ring.

SECTION VIII

OLF TELEMETRY DATA EQUIPMENT

8-1. OLF TELEMETRY DATA EQUIPMENT

The OLF Radio Frequency Communications and Telemetry circuits were presented in Section VI. Requirements were established for communications nets, OLF voice communications and data channels. In this section, the OLF telemetry channels are treated in detail: types of data to be handled are discussed and telemetry channel assignments estimated.

The OLF telemeter will be required to handle diverse data. Its capabilities will include data transmission and reception between both the ground complex and the S/C.

a. OLF Transmitted Data. The data to be transmitted by the OLF telemeter is as follows:

(1) OLF to Ground Complex

(a) OLF telemetry data showing the condition of the OLF and its crew and the results of scientific data acquisition. Specifically, this will include monitoring of OLF guidance, attitude controls, propulsion, temperatures, lift support parameters, bio-medical information, and miscellaneous equipment performance.

(b) Single frame video for the transmission of written displays, pictures of local conditions, and other images as required.

(c) S/C telemetry information, relayed to the ground complex via the OLF during countdown and the initial portion of powered flight by the S/C. Relay distances in excess of 4,500 miles do not appear feasible because of transmitter power requirements; consequently, DSIF must assume control early in flight. This approach presents several advantages in that it reduces the required S/C communication payload and the required capabilities of the ground complex while allowing a single station to maintain control of the S/C during initial powered flight. However, use of the OLF as a relay station for S/C intelligence will require that the data transfer capabilities of the OLF be expanded.

(2) OLF to S/C Transferred Data

(a) OLO mission data consisting of guidance and trajectory data for the S/C.

(b) Parameters required for checkout and countdown.

(c) Command signals relayed from the ground complex.

b. OLF Received Data. The data to be received by the OLF telemeter is as follows:

(1) Ground Complex to OLF

(a) OLO mission data consisting of guidance and trajectory data for the OLV during countdown.

(b) Commands selecting operating modes of the OLF telemetry equipment and performing other functions as required.

(c) Commands for relay to the S/C while the OLF is acting as a relay station.

(2) S/C to OLF Transferred Data. All S/C telemetry data for relay while OLF acts as S/C communications link.

c. Estimate of Telemetry Channel Assignments. Before conceptual hardware can be derived, some knowledge of telemetry measurement requirements is mandatory. Figure 8-1 presents a listing of representative measurements for a mission of this type together with estimated accuracy requirements and sampling rates. Figure 8-1 contains a total of 288 measurements listed under the broad categories of equipment monitor and evaluation, cabin environment, video, bio-medical, human abilities, and general scientific. About 95 percent of the measurements require only limited sampling rates. It is anticipated that OLO mission data will be presented to the telemetry system in serial binary form and that required rates will not exceed 1,000 bits per second.

8-2. OLF COMMUNICATIONS SYSTEM-OPERATING MODES AND EVOLUTION

The OLF communications system will possess a variety of operating modes. These modes include the telemetering of OLF performance and the condition of its crew, the acquiring of scientific data, the receiving of ground commands, the transferring of OLO mission data, and the relaying of spacecraft telemetry data.

In addition, requirements will exist for data storage and transmission on command. As the OLO program progresses, a gradual evolution can be expected in the functions monitored.

FIGURE 8-1

Estimate of Telemetry Measurement Requirements

A - Indicates 0.1 to 1%
B - Indicates 2 to 7%

OLF Equipment Monitor and Evaluation

Function	Quantity	Accuracy	Maximum Sampling Rate (samples/sec)
Pitch acceleration	1	A	20
Linear acceleration	3	A	20
Angular rates	3	B	20
Attitude	3	B	20
Reaction jet control	12	A	20
Reaction jet valve position	12	A	1
Fuel flow (rate)	12	B	20
Fuel supply	6	B	1
Power supply voltages	6	AB	20
Power supply frequency	4	AB	20
Programmer signals	10		1
Radio altimeter	1	A	20
Position data	2	A	20
External temperature	8	B	1
Inside air temperature	4	B	1
Equipment temperature	20	B	1
Meteorite impact	1	B	20
Command receiver signals	10		20
Pitch jets	1	On-Off	20
Roll jets	1	On-Off	20
Yaw jets	1	On-Off	20
Miscellaneous	38	AB	10

FIGURE 8-1 (Cont)

Estimate of Telemetry Measurement Requirements

Function	Quantity	Accuracy	Maximum Sampling Rate (samples/sec)
General Warning (On-Off)			
Dangerous CO_2 level	1		1
Cabin decompression	1		1
Radiation warning	1		1
Electrical system failure	1		1
Environmental control system failure	1		1
Attitude control failure	1		1
	Total Functions = 167		
Cabin Environment			
Oxygen concentration	1	B	1
CO_2 concentration	1	B	1
Miscellaneous gasses	6	B	1
Total gas pressure	1	B	1
Gaseous flow	2	B	10
Cabin wall temperature	2	B	1
Cabin air temperature	2	B	1
Radiation rate	4	B	5
Humidity	1	B	1
CO_2 filter status	2	B	1
O_2 reserve	1	B	1
	Total Functions = 23		

FIGURE 8-1 (Cont)

Estimate of Telemetry Measurement Requirments

Function	Quantity	Accuracy	Maximum Sampling Rate (samples/sec)
Assumed Bio-Medical			
Electrocardiogram (EKG)	1	B	320
Phonocardiogram	1	B	180
Respiration	1	B	20
Electroencephalogram (EEG)	3	B	120
Galvanic skin resistance (GSR)	1	B	20
Blood pressure	2	B	2
Electro-oculograph (EOG)	1	B	20
Body temperature	1	B	20
Exhaled gases	2	B	20
	Total Functions = 14		
Human Capabilities			
Compensatory tracking	1		1
Auditory vigilance	1		1
Warning light monitoring	1		1
Arithmetic computation	1		1
Pattern discrimination	1		1
Probability monitoring	1		1
Scale position monitoring	1		1
Miscellaneous	10		1
	Total Functions = 17		

FIGURE 8-1 (Cont)

Estimate of Telemetry Measurement Requirements

Function	Quantity	Accuracy	Maximum Sampling Rate (samples/sec)
Assumed Scientific Date			
Scientific experiments will consist of cosmic ray sensors, scintillation counters, magnetometers, ion chambers, lyman alpha telescopes, electric field measurements, magnetic field measurements, plasma detectors, etc. No attempt is made to define this area in detail. For system definition, the following assumption is made:			
Miscellaneous	67		1
	Total Functions = 67		
Narrow-band video - 100,000 samples per frame.			

a. Evolution of Data Requirements During Phase II. As the program progresses, the emphasis on acquired data will change as follows:

(1) OLF During Ground Checks. During ground checks, all operating modes will be checked for proper operation.

(2) OLF Orbiting, Unmanned. Cabin environment and equipment performance will be closely monitored. Scientific data acquisition will not be commenced until the crew is on-board.

(3) Manned OLF. During the early period of manned operation, equipment performance, cabin environment, and bio-medical and psychological reactions will be heavily telemetered for detailed ground evaluation. Scientific experiments will be of secondary importance. This orientation will be required to perform field evaluation of all OLF equipment, detect environmental conditions dangerous to health, and detect dangerous medical conditions which have not been observed during previous programs but which might occur during extended orbital operations.

After the initial proving phase, psychological experiments will acquire higher priority since the over-all OLO concept requires that man be able to function effectively in the orbital environment for extended periods. During this same period, the feasibility of OLV checkout and countdown while in orbit will be proved.

b. Evolution of Data Requirements During Late Phase II. After the initial Phase II operations, equipment performance, bio-medical reaction monitoring, and cabin environment checks

will be performed on a scheduled basis. The acquisition of scientific data will now assume a role of importance secondary only to actual OLO operations, and more sophisticated psychological experiments will be performed.

c. Evolution of Data Requirements During Phase III. No marked changes in approach are envisioned during this phase. Two primary considerations will be made:

(1) Keep the system sufficiently flexible to allow the incorporation of a succession of scientific measurements.

(2) Retain sufficient capabilities to allow for modifications in the OLO program.

d. Evolution of Data Requirements During Phase IV and Subsequent Phases. It is projected that a redesigned communications package will be required at the beginning of Phase IV to keep the OLF facilities current with the evolving philosophy of OLO. A summary of telemetry requirements by operation is shown in figure 8-2.

e. Data Storage and Transmission. After the OLF has been established in orbit, continuous data transmission will be undesirable because of the extremely large quantities of redundant data which would be collected. Some data is ideally suited for storage with or without time correlation, while other data is of value primarily in real time. The requirements on the various types of data are discussed below.

FIGURE 8-2

Summary of Telemetry Requirements
By Operation

	OLF Ground Checks	Injection to Orbit	OLF Unmanned in Orbit	Initial Manned OLF	Rendezvous and Docking	Orbital Launch Operations	Routine Operations (Phase II)	Routine Operations (Phase III)	Routine Operations (Phase IV)	Routine Operations (Phase V)
Equipment Performance and Monitoring	X		X	X	X	X	X	X	X	X
Cabin Environment Monitoring	X		X	X	X	X	X	X	X	X
Bio-Medical Monitoring	X			X			X	X	X	X
Psychological Monitoring	X			X			X	X	X	X
Receive Commands	X	X	X	X	X	X	X	X	X	X
Transfer OLO Mission Data	X					X				
Relay of OLV Telemetry	X					X				
Video Transmission (Slow Rate)	X			X		X	X	X	X	X
Scientific Data Acquisition	X				X	X	X	X	X	X
Operate Tracking Aids	X	X	X	X	X	X	X	X	X	X
Operate Voice Links	X			X	X	X	X	X	X	X
Two-Way Video (Real Time)									X	X

(1) Bio-medical and Psychological Data. The crew members of a facility such as the OLF could not be expected to wear a bio-pack at all times. Therefore, a bio-unit will probably be provided which will acquire data on each man after it is put on. A psychological test unit of the same type will be employed. Then data could be collected at any free period throughout the day and stored for later transmission. No real-time correlation would be required.

(2) Scientific Data. Scientific data is primarily of a slowly changing nature and is gathered over long periods of time. To circumvent the continual ground processing of this data, it can be stored and transmitted later at a much higher rate to a single station. Real-time correlation will be required in this area.

(3) Video. Image storage will be required in this area to preserve desired pictures until ready for transmission.

(4) OLF Equipment Operation and Monitoring. After the OLF is established in orbit, ground checks of the OLF equipment will be conducted at predetermined intervals. No data storage will be required, since the primary application of this type of data is in real time.

(5) OLO Mission Data. OLO mission data will be received by the OLF from the ground complex and transmitted to the S/C. Storage will be provided by the OLF computer with transfer initiated by either local or remote command. The OLF telemetry system will not possess storage capabilities for this type of data.

Based on the preceding discussion, the concept has been developed that surveillance of the OLF will be performed by ground facilities at scheduled times. Real time OLF equipment operation or stored bio-medical, psychological, or scientific data will be transmitted by the OLF as selected by ground command. Measurement handling operation is reviewed in figure 8-3.

FIGURE 8-3

Measurement Handling Operation

Measurement	Real Time	Store and Dump
Bio-medical		X
Psychological		X
Equipment performance and monitoring	X	
Scientific data		X
OLO mission data	X	

f. Evolution of S/C Telemetry Measurements. It is projected that the total number of telemetry measurements will remain constant throughout the OLO program. Equipment monitoring and evaluation will decline as confidence and experience increases, but will will be offset by an increase in scientific data acquisition. A graph showing the anticipated telemetry evolution is shown in figure 8-4.

g. Equipment Evolution and Hardware Schedule. It is projected that two generations of communications equipment will be required in support of OLO. First generation equipment

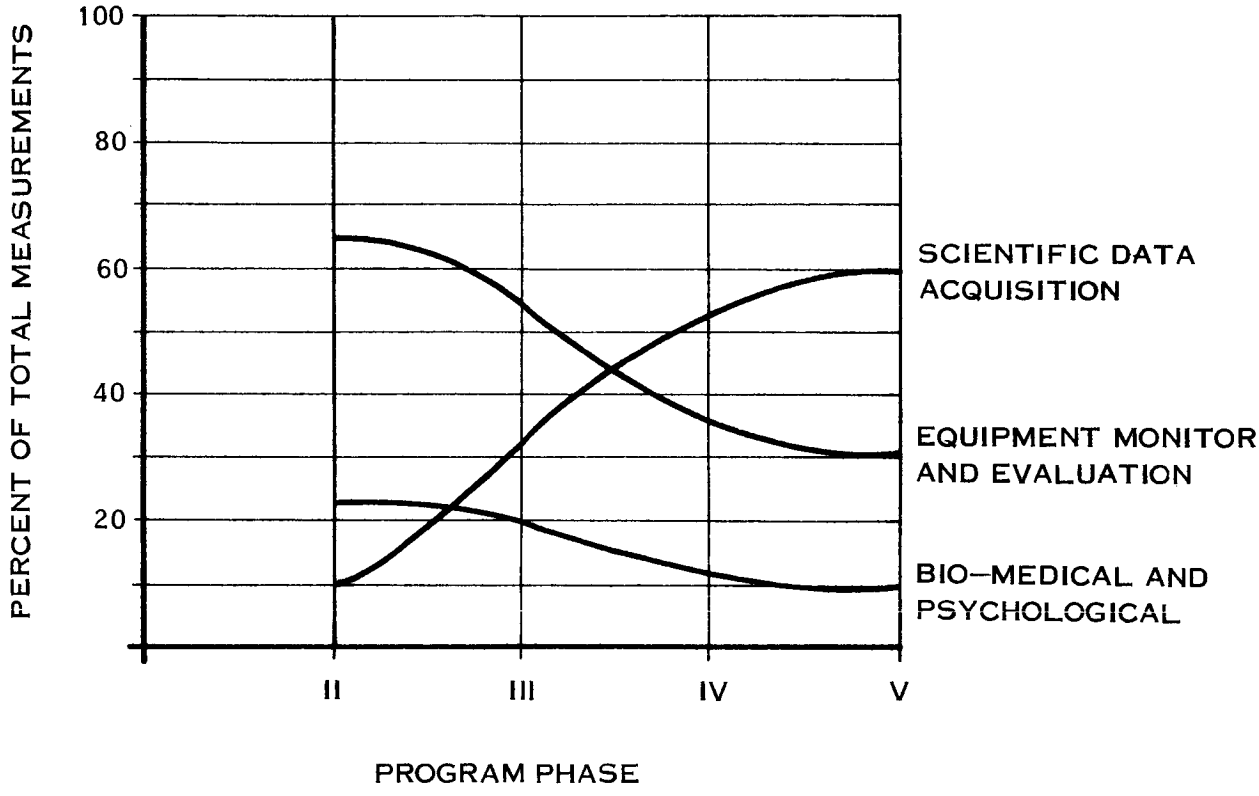
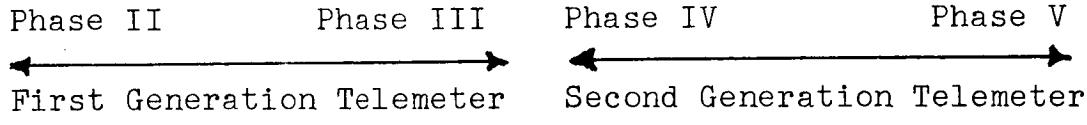


Figure 8-4 OLF Measurement Requirements

will be installed in the initial OLF. Changes in measurement requirements, refinements in orbital launch operations, a gradual shift of telemeter usage from an experimental to a research-tool, and OLF evolution will require a second generation communications system for the intermediate OLF. The second generation equipment will be sufficiently flexible to perform adequately throughout the remainder of the present program, including use with the ultimate OLF. Hardware evolution is as follows:



8-3. OLF RELIABILITY

The reliability of the OLF communications system will be an extremely critical factor. Because of the inherent high cost of supplying replacement units after the OLF is in orbit and because of the OLO operational requirements, a maximum effort must be made during all preflight operations, including design, fabrication, and checkout, to ensure that failure-free equipment is placed in orbit.

The reliability considerations for OLF communications equipment are presented in detail in Appendix G of this report. Component selection, burn-in, and extremely conservative circuit design will be employed to insure overall reliability. In addition, designs will incorporate multiple-use circuitry at the module, subassembly, and assembly levels wherever possible.

8-4. OLF MAINTAINABILITY

A detailed discussion of maintainability is presented in Appendix H. However, some additional specific comments and a summary of the Appendix H material is presented here for reading continuity.

Accessibility and fault isolation are two of the major problems to be considered. The design objective for the communications equipment will be fault isolation to the subassembly level through the use of self-check and ground-check techniques. For fault isolation at the module level, programming of the OLF checkout computer for subassembly testing and module isolation will be investigated. If the use of the checkout computer is feasible, module fault isolation by the computer will reduce the requirements for men as troubleshooting agents.

8-5. OLF SYSTEM FLEXIBILITY

Because of expected long-term operation and rapid evolution of system requirements as the OLO program progresses, considerable system flexibility must be maintained. Two basic approaches can be used to achieve this objective.

a. The telemeter can be provided with a memory unit and the program altered by local or ground command.

b. Man can act as a controlling agent and alter the program as required.

A completely automatic system for use with the OLF would be too complex and too costly. The optimum solution

appears to be a limited command program selection involving the basic telemeter only and man acting as agent for all measurement changes.

Sufficient system capacity will be provided to allow the addition of up to 30 percent of the original number of telemetry channels. The basic telemeter will provide the transmission capacity for this number of channels. The addition of a measurement will require only the insertion of a selected module or subassembly in the telemeter and wiring in the appropriate sensor.

The principle of on-board data reduction to limit transmitter bandwidth requirements and to reduce the amount of redundant data received on the ground was also considered. Extensive use of this approach was considered too complex and costly for the OLF communications system, especially in view of the previously developed concept of scheduled OLF surveillance instead of continuous data acquisition.

A single equipment design cannot be sufficiently flexible to be used throughout the OLO program. Program hardware requirements are discussed in paragraph 8-2.

8-6. ROLE OF MAN

The OLF communications system will not be required to operate in an unmanned condition except during the period immediately following initial injection of the OLF into orbit. During the remainder of the OLO program, the OLF crew will be expected to provide a fundamental portion of system programming, operation, maintenance, and repair. This program will obviate the need for extremely complex

equipment and will provide inherent system flexibility. However, man's responsibility in this operation is restricted by the limited time available for these duties.

a. OLF Telemetry System Programming. Figure 8-5 illustrates the concept for OLF telemetry programming. A fixed program under local or ground control determines the basic types of data being processed by the telemeter. Man will normally program only at the sub-level of basic data types (i.e., scientific data, bio-medical data, etc.) where measurements may be removed, added, or otherwise altered within the design limits of the system. This man-machine relationship appears to provide the best trade-off between complexity and flexibility for the OLF system.

b. OLF Communications System Maintenance. It is anticipated that most circuitry will be digital in nature, reducing significantly the problem of marginal operation. However, some analog circuitry will be employed, particularly in the signal conditioning equipment. This equipment will require periodic adjustment by the OLF crew. The design objective will be to provide measurement points such that a single unit of general-purpose test equipment common to other OLF equipment can be utilized.

c. OLF Communications System Repair. Fault isolation and repair were discussed in paragraph 8-4. Subassembly fault isolation will be performed by programmed self-checks and ground-checks, while module fault isolation will rely on the diagnostic ability of man in conjunction with checkout programming of the OLF computer. The candidate system is discussed in detail in Appendix F, where figure F-2 summarizes the system.

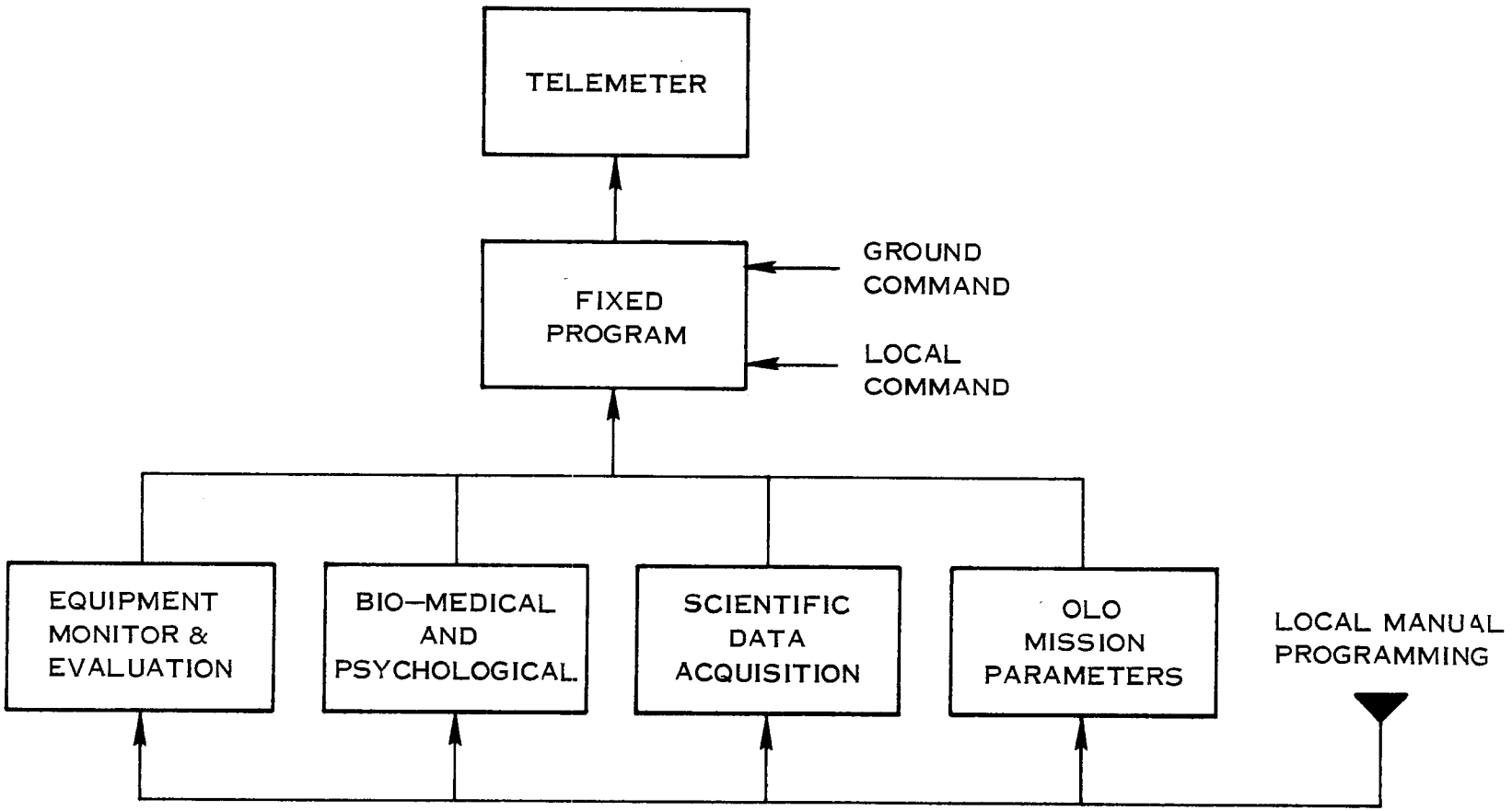


Figure 8-5 OLF Telemetry System Programming

SECTION IX

CONCLUSIONS

9-1. OTHER SYSTEMS

Before discussing the candidate system a comparison is made of several systems as shown in figure 9-1 in which the candidate system appears in the first column. A 36 ground station concept and an advent concept are shown in this comparison.

9-2. CANDIDATE SYSTEM

Four major aspects of the candidate system were discussed in Sections 5, 6 and 8 separately. These were the satellite ring and the fence which provided continuity of communication and telemetry from the ground operation center to the OLF. Two additional parts discussed, were the radio frequency equipment in the OLF and its immediate neighbors and the telemetry terminal equipment in the OLF. These four parts are now reviewed as a complete system. Figure 9-2 illustrates the candidate system including all of these four parts. Pictorially the telemetry terminal portion is not evident. The table of figure 9-3 has been provided summarizing the major equipments included in the fence satellites and the OLF. This table forms a basis from which the hardware cost estimates were made. Study, development, test, training and spare parts have not been

included in the table although they form elements of the total cost. These elements as well as the hardware have been called out in the program schedule of figure 9-4. This schedule basically covers the period from the beginning of the calendar year 1962 through the calendar year 1967. Although intermediate and ultimate OLF schedule is shown for reference purposes, none of the costs have been projected beyond 1967. It is important to note that the period of time from January 1962 to the end of the first quarter of 1965 is a very short time in which to provide thoroughly qualified hardware. The success of the program depends upon having prototypes in the latter part of 1963 in order to properly qualify equipment by the end of 1964. In order to achieve prototype units in the last half of 1963 extensive study programs must be accomplished during the first half of 1962. The number of hardware units have been indicated directly on the hardware schedule.

The costs are presented in graphic form in figure 9-5. The lower curve shows projected equipment and operating costs of the ground fence for the years 1962 through 1967 totaling 18.9 million dollars. The next higher curve shows the initial and equipment costs satellite ring totaling 26.3 million dollars. This figure does not include the boost cost in that consistant information was not available. The next higher curve shows the OLF communication and telemetry system equipment totaling 30 million dollars. The top curve is the sum of the lower three curves which results in a grand total of 75.2 million dollars for the period 1962-1967.

Figure 9-1

COMPARISON OF 3 BASIC CONCEPTS FOR CONTINUOUS COMMUNICATION

Item	36 Ground Station Concept	12 Satellite Repeater Concept	Advent Concept (3 equally spaced satellites)
Relay Equipment - to provide continuous contact between OLF and A designated point on earth	A net of about 36 ground stations located accurately around the world.	12 satellite repeaters located around the world at approx. OLF altitude.	Use Advent Facilities - both in space and on ground
Bandwidth Limit	About 3kc - for existing world network	75kc proposed-could be several megacycles	Might be in megacycle range if channel is available
Cost	About $115 million including operating costs	About $75 million - not including booster costs for satellite repeaters	Total cost unknown. High Power Transmitter and Large Antenna needed on OLF for tracking Advent Satellites - provides difficult OLF stabilization problem
Availability	About 25% of required ground stations available.	2 new fence stations plus additional equipment in 4 available stations.	Doubtful - if available on full time basis

Figure 9-1 (Cont)

COMPARISON OF 3 BASIC CONCEPTS FOR CONTINUOUS COMMUNICATION

Item	36 Ground Station Concept	12 Satellites Repeater Concept	Advent Concept (3 equally spaced satellites)
Other Considerations	Because of tracking requirements at each station it is doubtful that stations could be unmanned.	Would be available for other special communication requirements between earth and space vehicles located within the near vicinity of repeater.	Should be investigated more thoroughly.

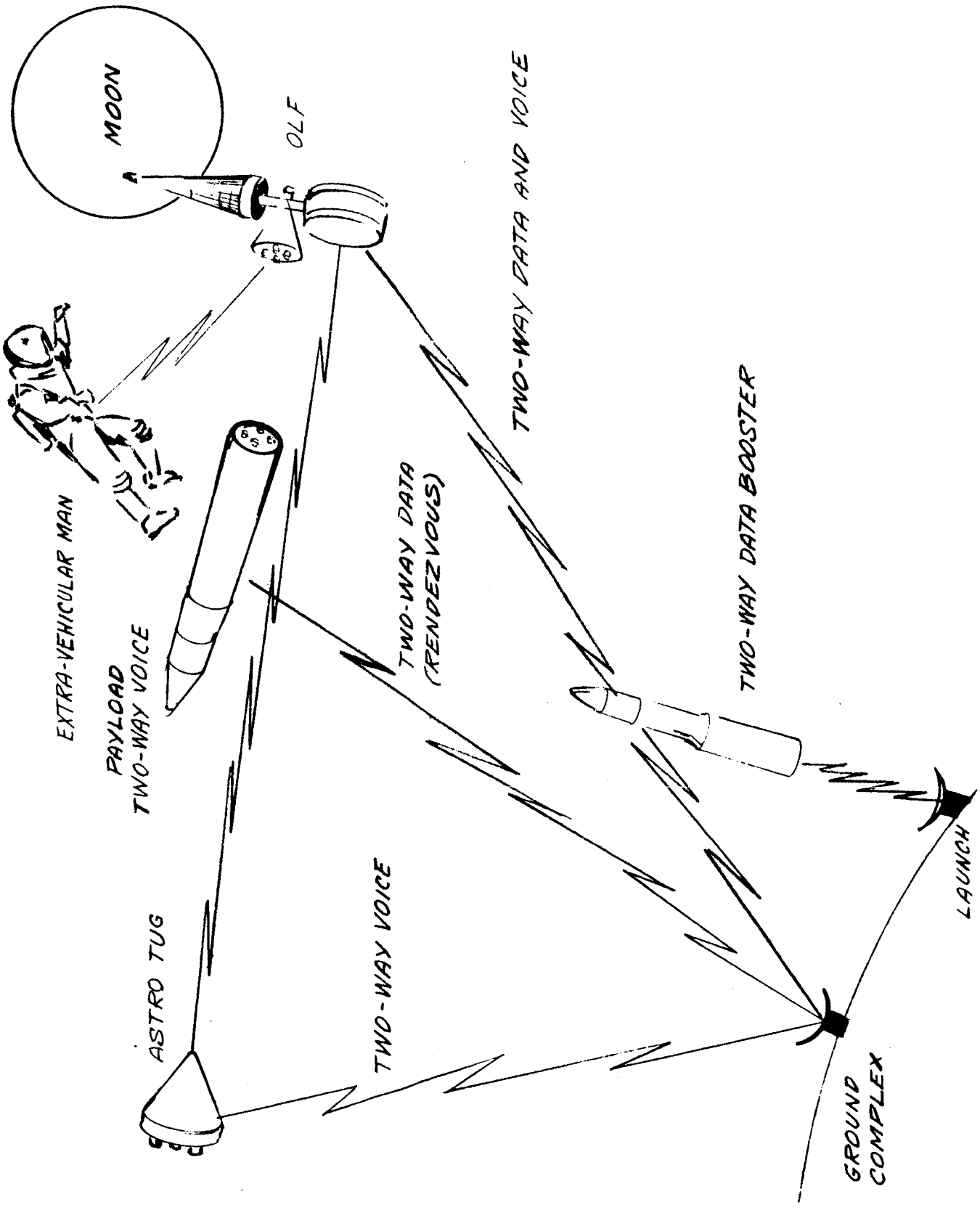


FIGURE 9-1A *OLO COMMUNICATIONS NET*

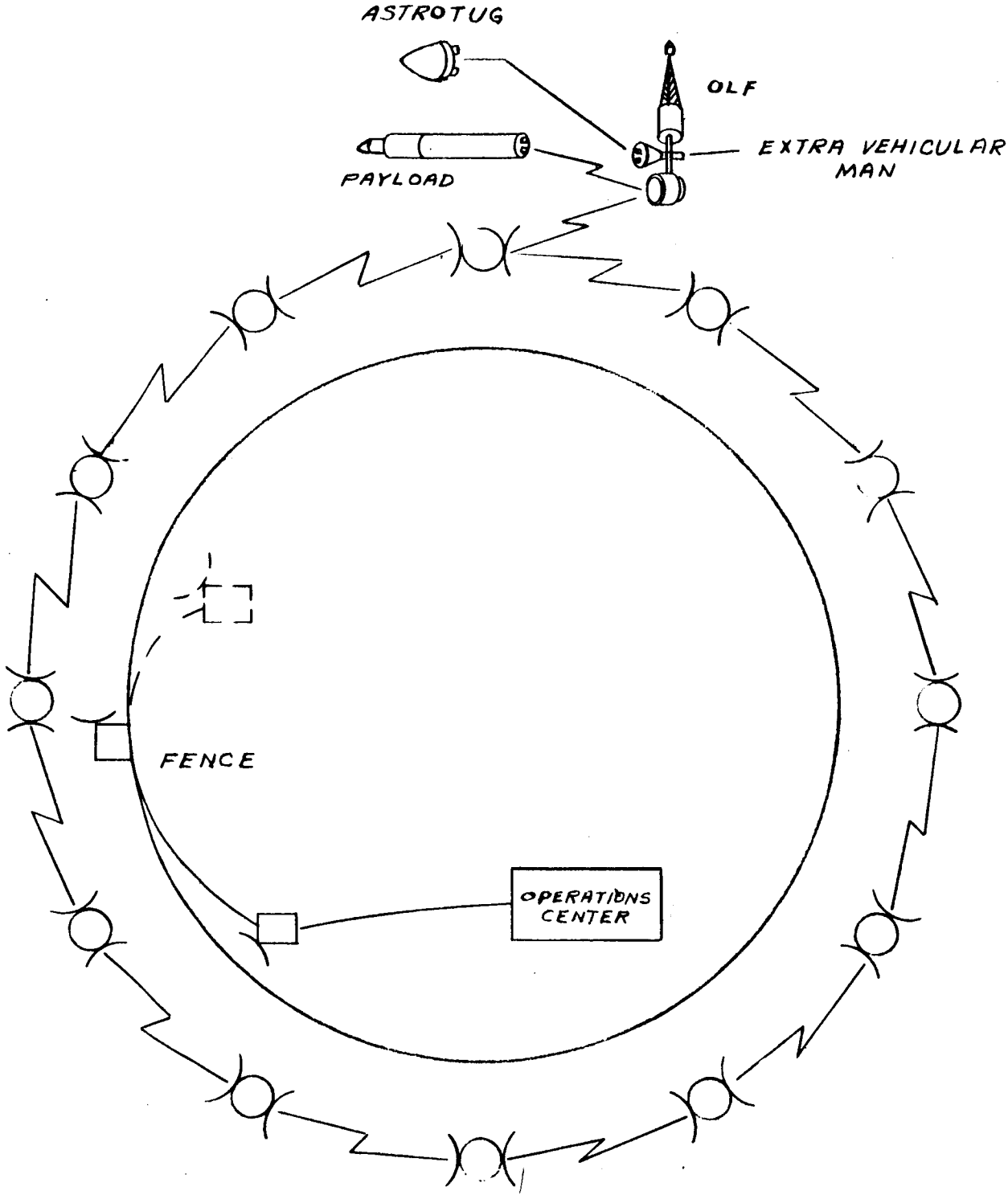


FIGURE 9-2 CANDIDATE SYSTEM

Figure 9-3

TABLE OF SYSTEM EQUIPMENT

Station	Item	Qty/Stn	Num of Stns	Total Qty
Fence	2300 mc Diplexer	1	6	6
	Tracking 48 DB Norm Ant	1	6	6
	2300 MC Rec, NT 200°K	1	6	6
	2300 MC Trans, 10 Watts	1	6	6
	Data Term. - UHF	1	6	6
	2-30MC Log. Per. Antenna	4	5*	20
	2-30MC Trans. Filters	6	5	30
	2-30MC Power Amps, 10KW	6	5	30
	2-30MC Rec., NF 15DB	6	5	30
	2-30MC Exciter	6	5	30
	Data Term., HF	6	5	30
	2-30MC Preamp.	2	5	10
	Building & Power	1	6	6
Satellite	8 Ft Dish Ant.	2	12	24
	Filter, 2300 MC, F_3	2	12	24
	Filter, 2300 MC, F_4	2	12	24
	2 Ft Dish Ant.	1	12	12
	Filter, 2300 MC, F_1	2	12	12

*4 + 1/2 + 1/2 = 4 relays + 2 ends

Figure 9-3 (Cont)

TABLE OF SYSTEM EQUIPMENT

Station	Item	Qty/Stn	Num of Stns	Total Qty
Satellite (Cont)	Filter, 2300 MC, F_2	2	12	12
	Trans, 2300 Mc, 1 Watt	2	12	24
	Rec., 2300 Mc, Nt. 870°K	4	12	48
	Trans, 2300 Mc, 2 Watt	1	12	12
	Trans, 2300 Mc, 4 Watt	1	12	12
	Base Band Coupler	1	12	12
	Control Eqpt.	1	12	12
	Frame	1	12	12
	Power Supply Solar & Battery	1	12	12
OLF	8 Ft Dish	2	1	2
	3 Ft Dish	2	1	2
	Trans. 2300 Mc 8 Watts	1	1	1
	Trans. 2300 Mc 4 Watts	1	1	1
	Trans. 2300 Mc 2 Watts	3	1	3
	Baseband Inter-connections	1	1	1
	Internal Communications	1	1	1

Figure 9-3 (Cont)

TABLE OF SYSTEM EQUIPMENT

Station	Item	Qty/Stn	Num of Stns	Total Qty
OLF (Cont)	Filter, 2300 Mc	8	1	8
	Rec, 2300 Mc, Blt. 870°K	5	1	5
	Preamp., 2300 Mc, N.T. 340°K	2	1	2
	Telephone to Extra-Vehicular Man	3	1	3
	Telemetry Terminal	1	1	1
	Slot Antenna	6	1	6
Extra Vehicular Man	Telephone	1	3	3
	Rec. 2300 Mc NF 30DB	1	3	3
	Trans. 2300 Mc 10 Mw	1	3	3
Payload or Astrotug (Not Priced under OLF Package)	3Ft Dish	1	X	1X
	8 and Filter	2	X	2X
	Trans, 2300 Mc, 2W	1	X	1X
	Rec, 2300 Mc, W.T. 870°K	1	X	1X
	Telemetry Terminal, simplified	1	X	1X

Figure 9-3 (Cont)

TABLE OF SYSTEM EQUIPMENT

Station	Item	Qty/Stn	Num of Stns	Total Qty
S/C (Not Priced under OLF Package)	8Ft Dish	1	1	1
	Band Filter	2	1	1
	Trans, 2300 Mc, 2W	1	1	1
	Rec, 2300 Mc, NT. 870°K	1	1	1
	Telemetry Terminal	1	1	1

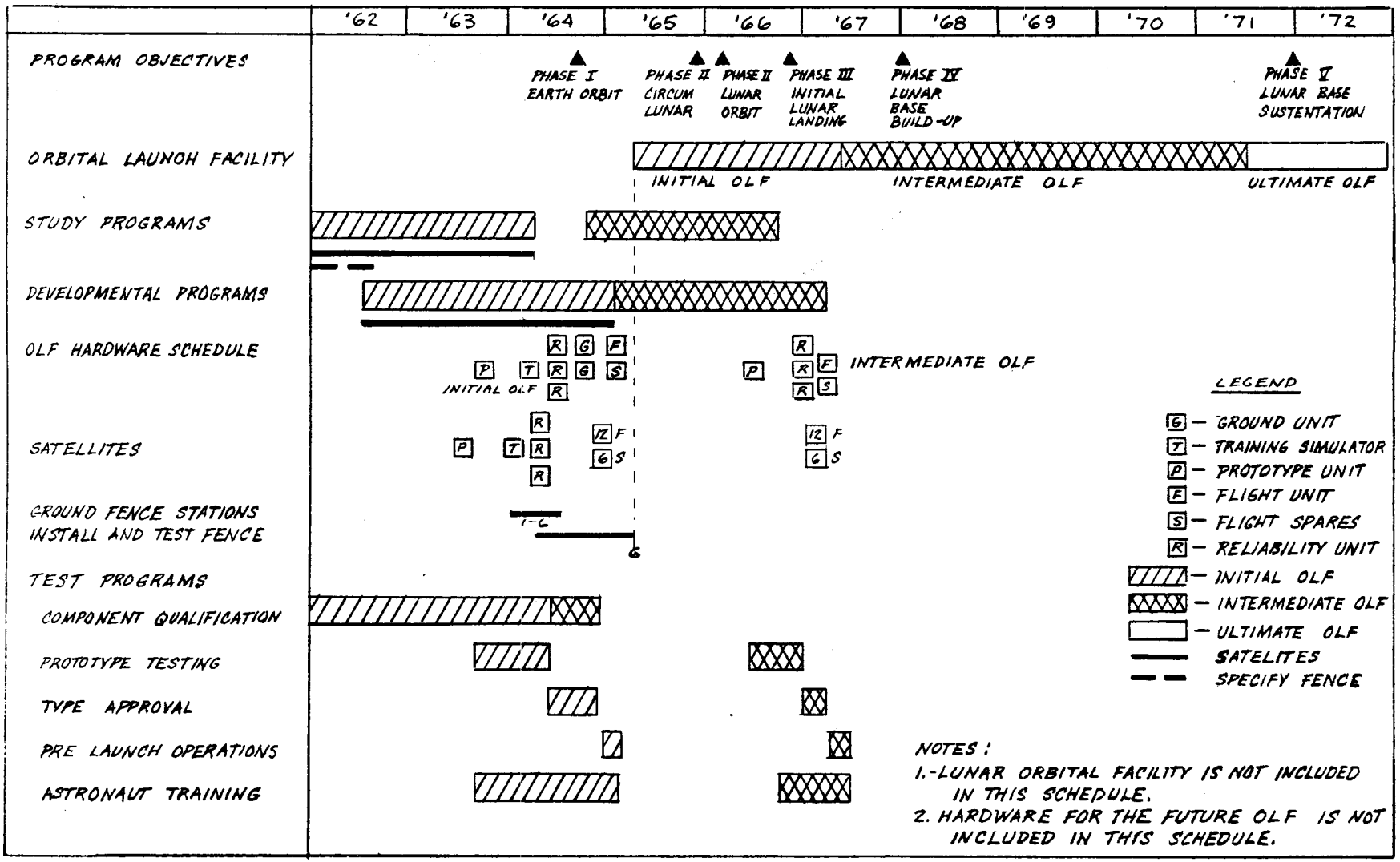


FIGURE 9-4 PROGRAM SCHEDULE TELEMETRY AND COMMUNICATIONS

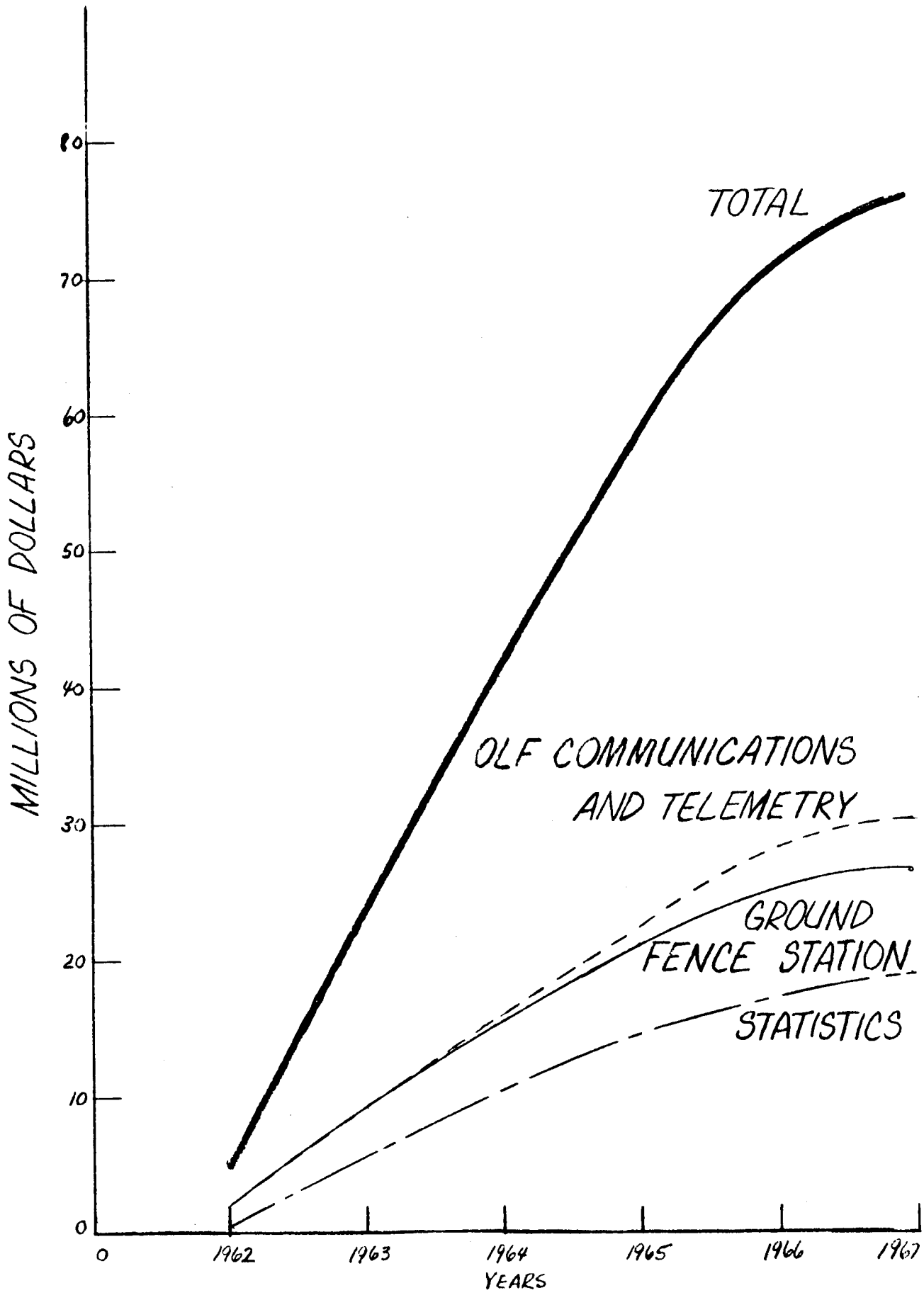


FIGURE 9-5 PROJECTED COSTS

APPENDIX A

BASIC NETWORK GEOMETRY

The Orbital Launch Operation communication system has many mobile station requirements. The following oversimplification is given to emphasize the elements of the geometry of the toughest of the communication problems, that of continuity. After the discussion of the five geometric concepts, each concept will be related to the OLO operation.

In figure A-1 a disc rotating at angular velocity ω_1, is shown for consideration. Located on this disc at a discrete location, is user B. External to the disc is source A, fixed with respect to the inertial reference. Source A is tied to the disc with a brush and, B is tied to the periphery of the disc with a fix connection. Figure A-2 shows essentially the same concept as figure A-1 except that the periphery of the disc has been segmented, and user B is connected to each segment. Figure A-3 is similar to figure A-1 except that the brush and the fixed connection have been interchanged and the motion ω_1 is confined to termination A while user B stands still. It is true that the periphery of the disc has been disassociated from the disc and becomes a ring rotating with termination A at rate ω_1.

The concept of figure A-3 may be extended to include other terminations at angular velocity ω_1, termination C for instance, as shown in figure A-4. This may be further extended to include termination C at a different angular velocity from that of termination A, ω_2, and connected to the ring by a brush as shown in figure A-5.

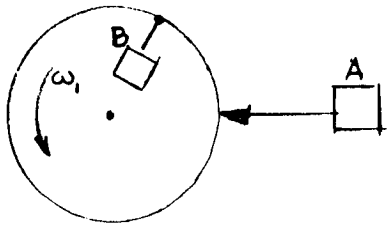


FIGURE A1. GEOMETRIC CONCEPT DISC ROTATING AT ANGULAR VELOCITY ω_1

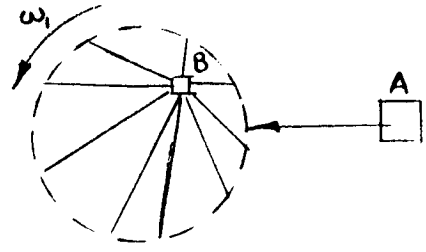


FIGURE A-2 GEOMETRIC CONCEPT SEGMENTED DISC

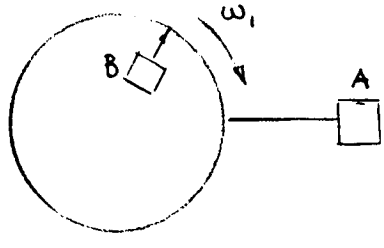


FIGURE A-3 GEOMETRIC CONCEPT INTERCHANGED BRUSH AND FIXED CONNECTIONS

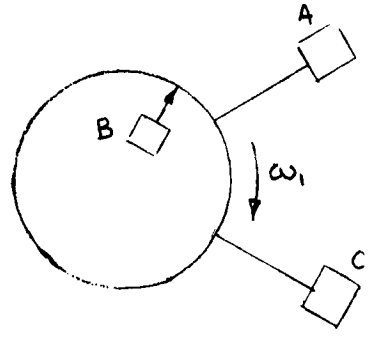


FIGURE A-4 GEOMETRIC CONCEPT ADDITION OF TERMINATION C

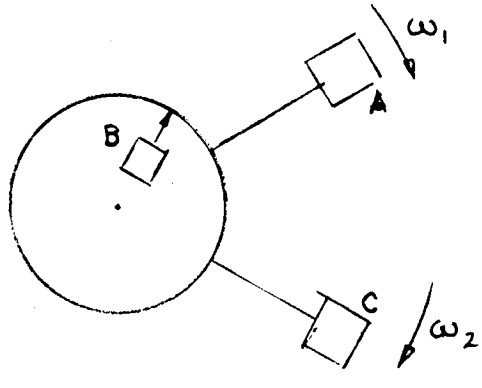


FIGURE A-5 GEOMETRIC CONCEPT. ADDITION OF TERMINATION C AT DIFFERENT ANGULAR VELOCITY

Figures A-1 and A-2 represent a ground station at user B and a mobile space borne station, the OLF, at termination A. The difference between figure A-1 and figure A-2 is the segmenting of the periphery. Figure A-2 represents the present method of a multitude of ground stations, one for each segment connected back to the central control point by radiating connections. Figure A-3 anticipates the possibility of generating a system where the segments are tied together peripherally and rotate with termination A, the OLF. Figure A-4 further extends this to include space borne terminations of angular velocities different from that of the ring.

APPENDIX B

TYPES OF CHANNELS

Two types of communication channels find extensive use in our communication systems today. The first type of channel is known as a simplex channel and the second type of channel is a duplex channel.

The simplex type of communication allows traffic to flow only in one direction between the source and the receptor at any given time. Since traffic flow is in one direction only at any given time, supervision of the traffic flow is difficult unless separate supervision channels are provided. For example, if a source is transmitting information, there is no assurance over a simplex system that the receptor has received and understood all of the information until the end of transmission when the direction of transmission is reversed and the receptor is allowed to acknowledge the message. The sender cannot be interrupted for any reason in this system.

The duplex system of communication allows traffic to flow in both directions simultaneously between the source and the receptor. This system then provides a natural supervision system between the ends of the communication channel since either end may interrupt the flow of traffic at any time.

Basically, communication networks may be designed around either of these two basic concepts.

APPENDIX C

NET CONTROL CONSIDERATIONS

C-1. GENERAL

When communication systems expand to more than two terminal stations the factors of interference between stations and system supervision must be considered.

With more than two terminal stations, it becomes necessary to consider the type of communication employed. With the employment of a wire system the communication is bounded and interference between stations is inherently not a problem. If radio links are employed, the only method of eliminating interference between stations is the use of highly directional antennas and multiple channel operation.

Since the communication problem presented in this proposal requires the employment of radio links the discussion which follows will be restricted to this type of communication.

C-2. SIMPLEX RADIO COMMUNICATION

The simplex type of radio communication will be discussed on the basis of three stations as shown in figure C-1. On any one given frequency at any one given increment of time any one of the stations will be the source and the other stations the receptors. During another increment of time a different transmitter will be the source and the other stations will be the receptors. As mentioned in Appendix B during the

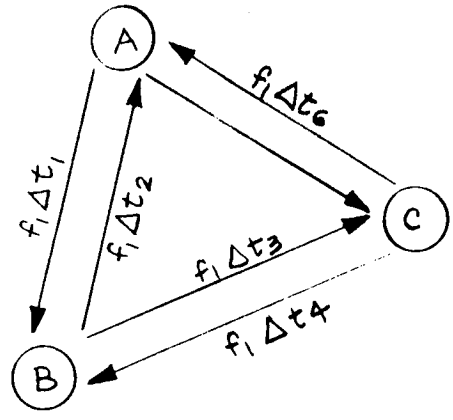


FIGURE C1 SIMPLEX TYPE OF RADIO COMMUNICATIONS

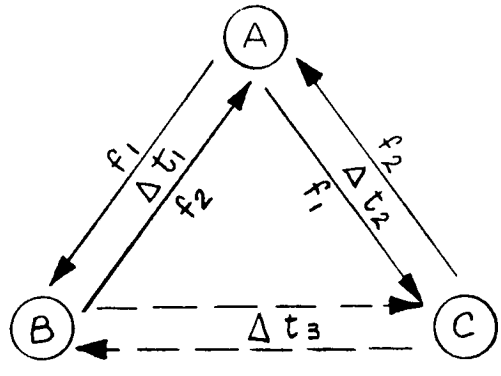


FIGURE C2 DUPLEX TYPE OF RADIO COMMUNICATIONS

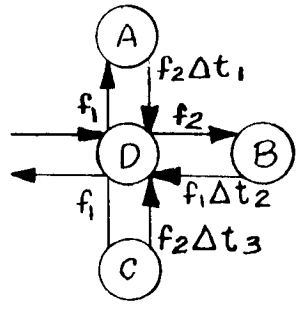


FIGURE C-3 NET CONTROL STATION-METHOD A

increment of transmission of any one given station transmission cannot be interrupted unless supervisory channels are provided.

This type of communication may assume several forms depending upon the length of the increment Δt. If Δt is long the system has the characteristics of point to point simplex communication. As the length of Δt becomes shorter the system may have the characteristics of point to point time-shared pulsed type of communication.

C-3. DUPLEX RADIO COMMUNICATION

The duplex type of radio communication will be discussed on the basis the three stations shown in figure C-2. In this system station A is assigned a transmission frequency of f_1 and a reception frequency of f_2. Therefore, the stations which are to communicate with station A must be assigned a frequency of transmission of f_2 and a frequency of reception of f_1. Immediately a problem becomes apparent in that station B cannot communicate with station C on these frequencies since they are both transmitting information on frequency f_2 and receiving information on frequency f_1.

The solution to this problem in such systems as the high capacity communication system employed by the Navy is the introduction of a net control station.

C-4. NET CONTROL STATIONS

One method of net control is shown in figure C-3 where station D has the responsibility of receiving all traffic from stations A, B, and C and transmitting all traffic to

A, B, and C. The net control station may also be assigned the responsibility of receiving other communication from other net control stations on other frequencies and distributing this traffic to stations A, B, and C.

A second method of net control is that shown in figure C-4. In this system several frequencies are assigned to the net control station N. The number of frequencies assigned will depend upon the phase of operation and the traffic density. Each station A, B, C or D requests permission for a frequency assignment on a supervisory channel to station N when traffic is to be transmitted. These frequencies may be assigned for any length of time required and then released back to the net control station when the particular transmission is completed.

A second solution to the frequency overlap problem introduced in figure C-2 is shown in figure C-5. Each station has one transmitter and several receivers. The number of receivers required depends upon the number of communication links. Some type of band filtering must be employed to allow operation of the transmitters and receivers simultaneously for full duplex operation. This filtering may be done by electronic filtering, transmitted and received wave polarization, transmission and reception directivity or large spacing between the transmitter and receiver. The method usually employed in restricted space is an electronic type of filtering. Stations A, B, C, and D are operated on frequencies f_{1a}, f_{1b}, f_{2a}, and f_{3a}.

The method of employment of the filters f_1, f_2, and f_3 is shown in figure C-6. The bands f_1, f_2, and f_3 are any band of non-overlapping frequencies chosen for the system. Frequency f_{1a} is any discreet frequency in band f_1. Frequency f_{1b} is another discreet frequency in band f_1. Frequency f_{2a} is some discreet frequency in band f_2. This type of assignment

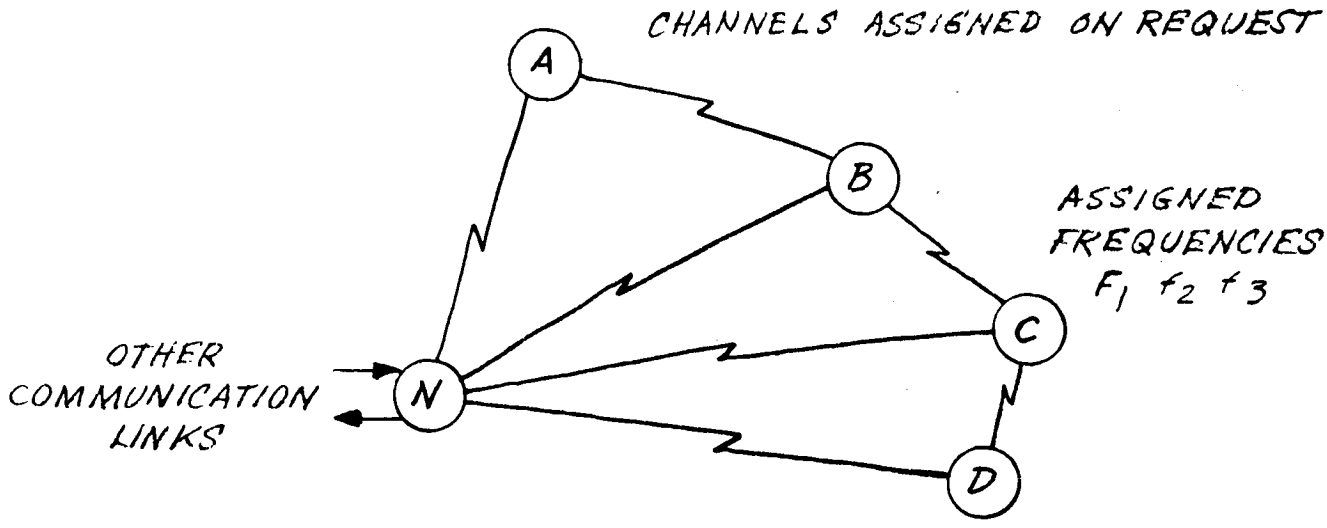


FIGURE C4 NET CONTROL STATION METHOD B

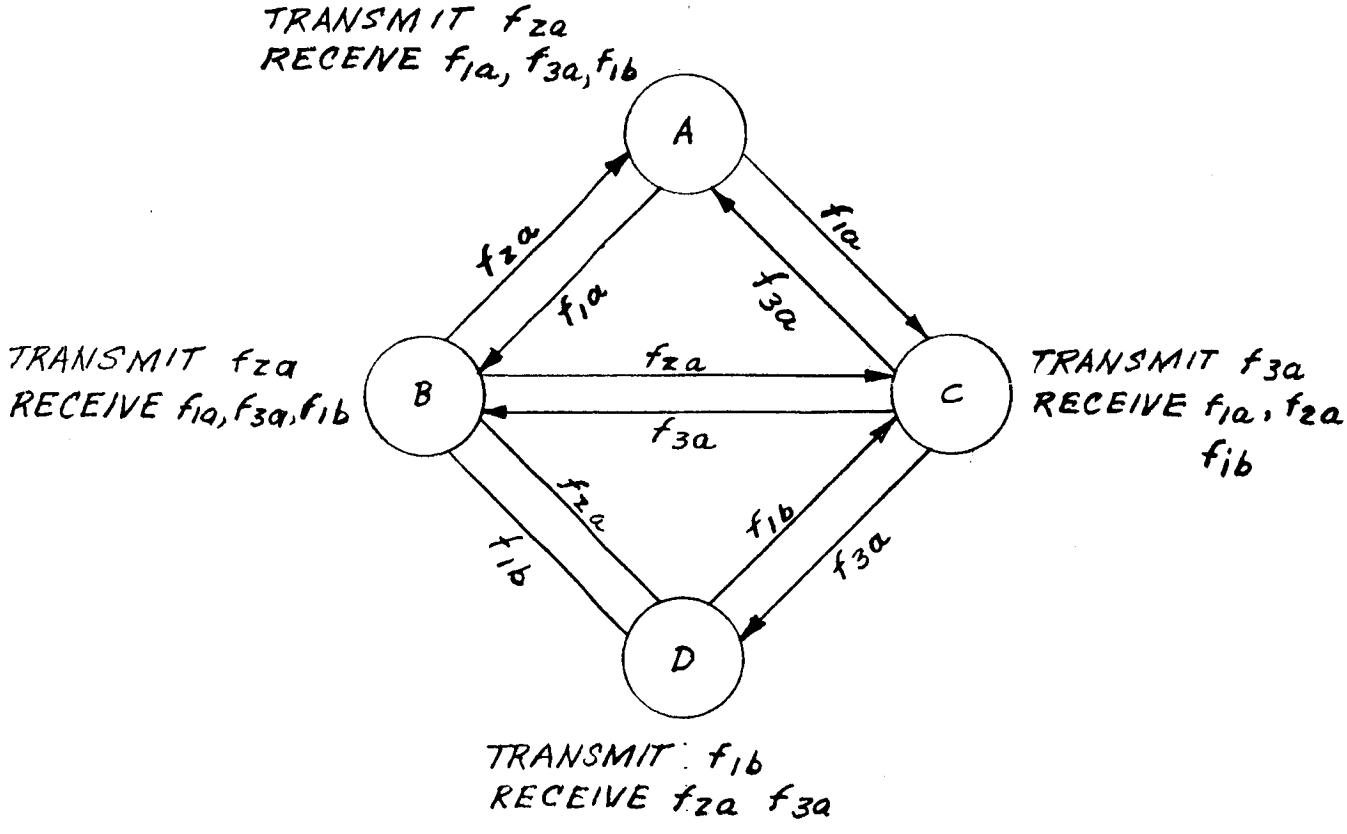


FIGURE C-5 NET CONTROL STATION METHOD C

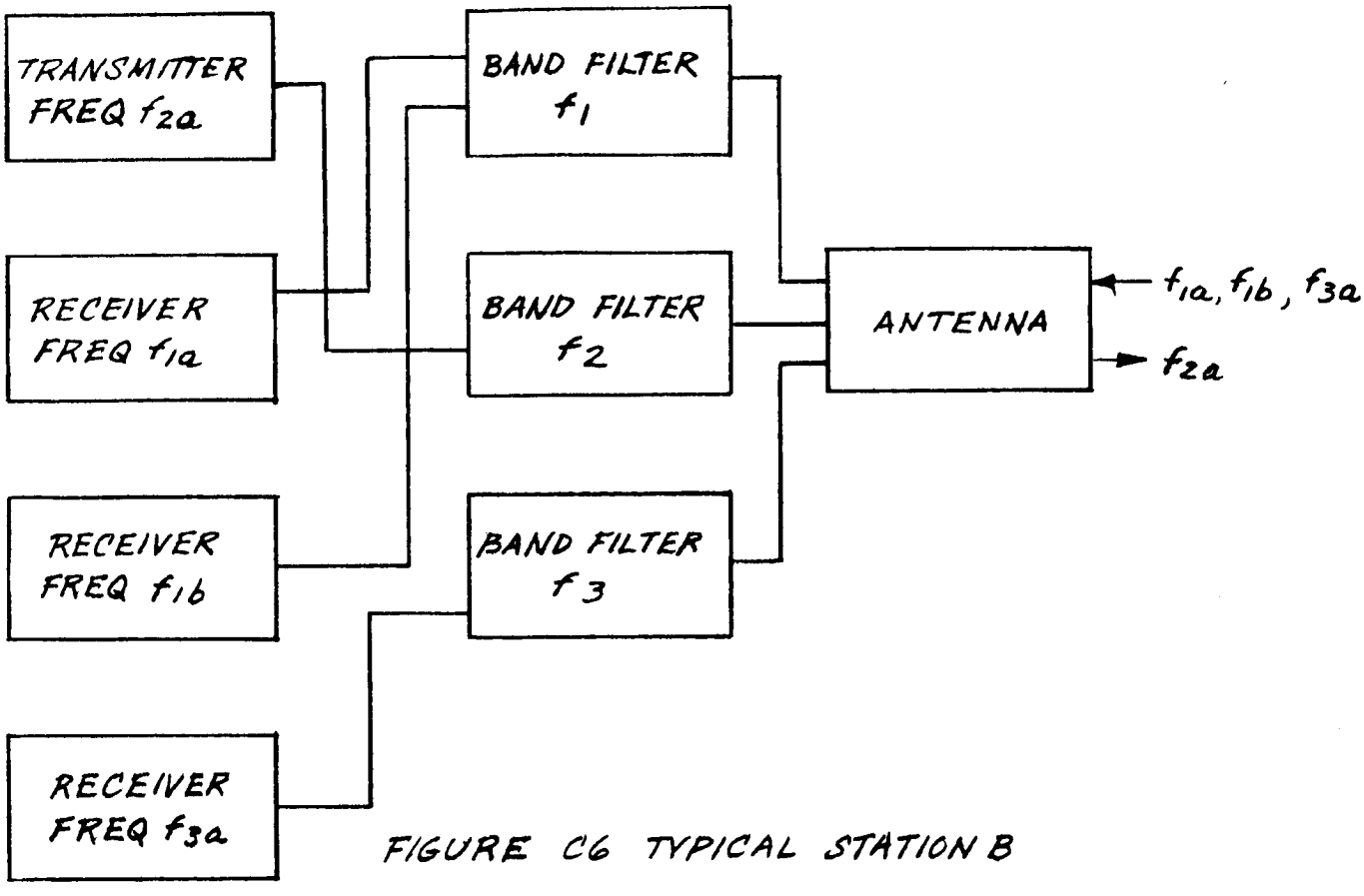


FIGURE C6 TYPICAL STATION B

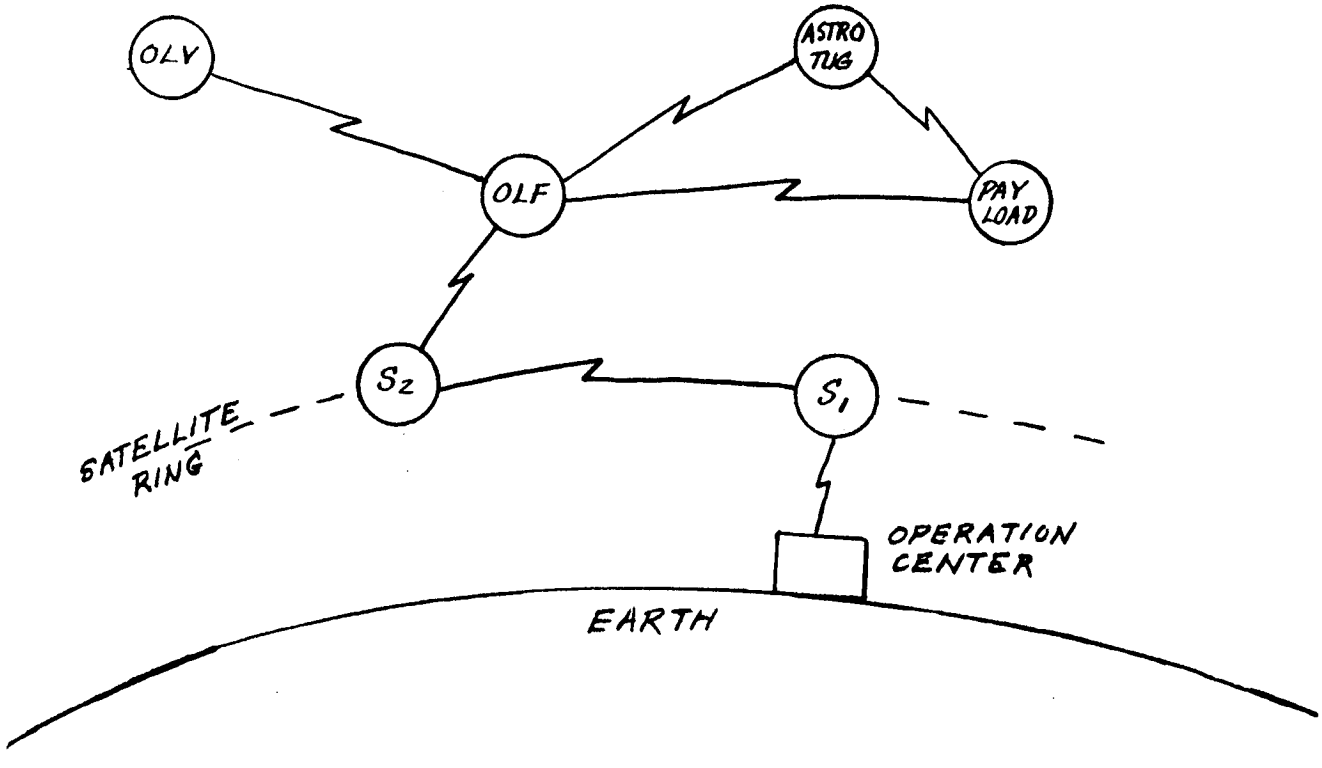


FIGURE C-7 FREQUENCY PLAN FOR OLF COMMUNICATION LINK

is carried to completion for the station under consideration. If the band filters are designed to be able to fulfill the transmit or receive requirements, complete patching flexibility may be achieved. The only difference between station B and station C in figure C-5 would be the interconnections between the transmitters, receivers and band filters and the frequencies of the transmitters and receivers.

Such a frequency plan provides for the OLF communication links as shown in figure C-7. Although there is continuous motion of the OLF complex with respect to the operations center, there is always a possible relay path between the operations center and any vehicle in space through the satellite ring. Other possible paths may be established upon demand. The paths suggested in figure C-7 are a few of the possible requirements.

A sequence of possible required paths are shown in figure C-8A through F. These examples are by no means restrictive. Many other possible combinations may be suggested by the operational requirements. Figure C-8A-D shows a possible required system just after the astro tug and OLF become operational. The payload is in communication with the ground just after launch in figure C-8a. If operation remains overhead it may communicate with the ground until it is in range of the OLF and then the OLF directly. (An example of this operation is a sounding rocket.) Figure C-8C illustrates the inclusion of the payload into the OLF to prepare for a docking operation of figure C-8D. Figure C-8E illustrates a possible communication complex later in the program. Note that figure C-8E and figure C-8F illustrate the flexibility of the system. In this case a payload is launched at some distance from the operations center. Communications may then be required with the OLF and operations center immediately after launch. This becomes possible in figure C-8F through the satellite relay ring.

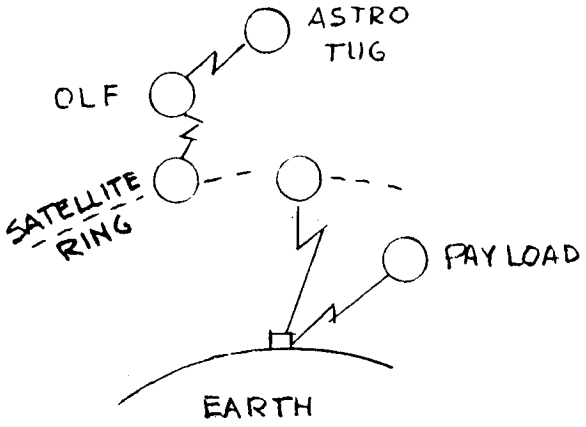


FIGURE 8a

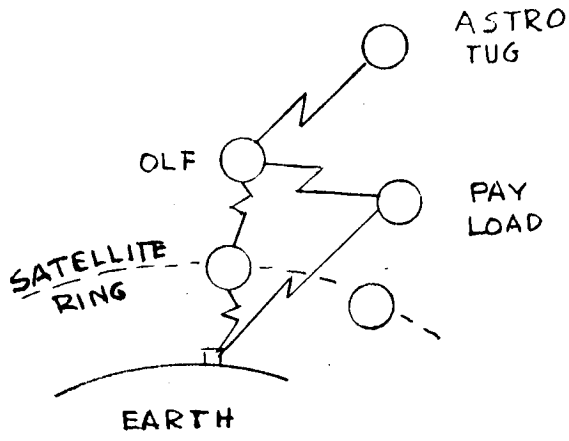


FIGURE 8b

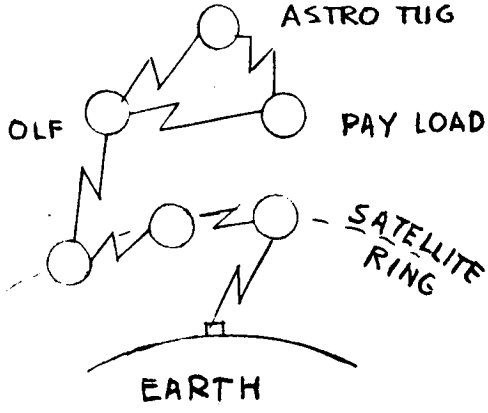


FIGURE 8c

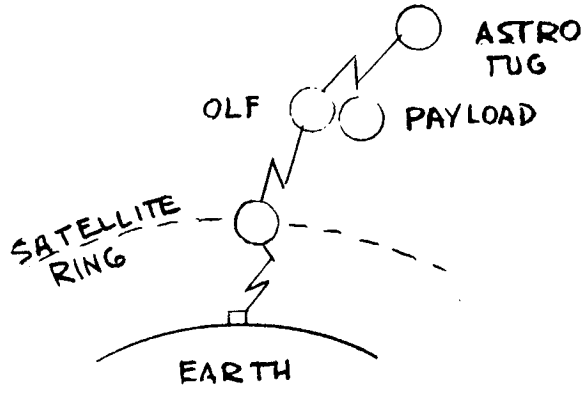


FIGURE 8d

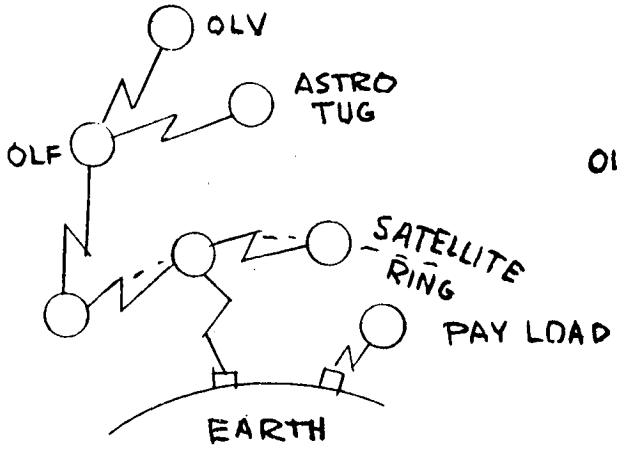


FIGURE 8e

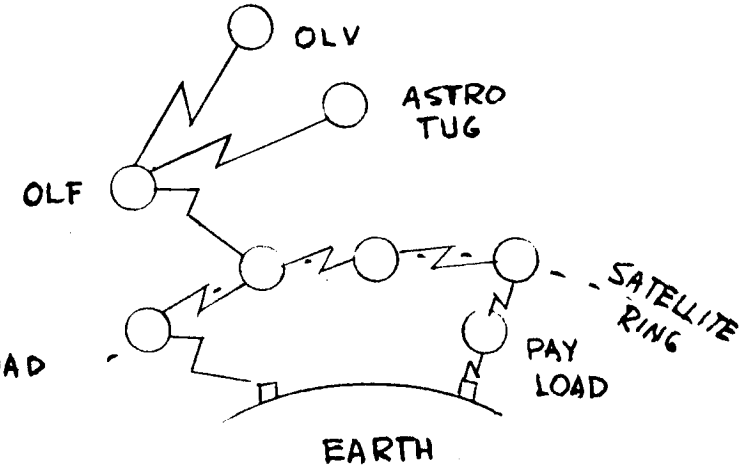


FIGURE 8f

FIGURE C-8 SEQUENCE OF POSSIBLE COMMUNICATION LINKS

APPENDIX D

THE FENCE

In the system proposed, the fence has the assignment of receiving and transmitting to and from the satellite ring and then relaying this information to the center of operations. The first part of this assignment is limited in capability (bandwidth and signal-to-noise ratio) within all practical limits by the power or weight of the space borne equipment.

The immediate solution of the second half, that of utilizing HF radio along the fence, introduces a limitation on capability that is somewhat independent of equipment. The assignment of wide channels in the HF spectrum and the vagaries of the ionosphere as a transmission medium, have a considerable effect on what may be done by means of HF radio. None the less the HF solution has been pursued. Overall comparisons may then be made with other means. Cables, or ground-to-ground communication via random or 24 hour satellites should be reexamined as candidates for improvement of the system. Unfortunately time has not permitted the evaluation of these aspects of the problem and they must be relegated to the list of items for further study.

Figure D-1 is a block diagram of a typical fence relay station. Each station may be divided into three sections, the equipment to and from the ring, the equipment for up fence operation and the equipment for down fence operation. The last two portions are identical. All three of these sections are never used simultaneously. The dotted arrows in the

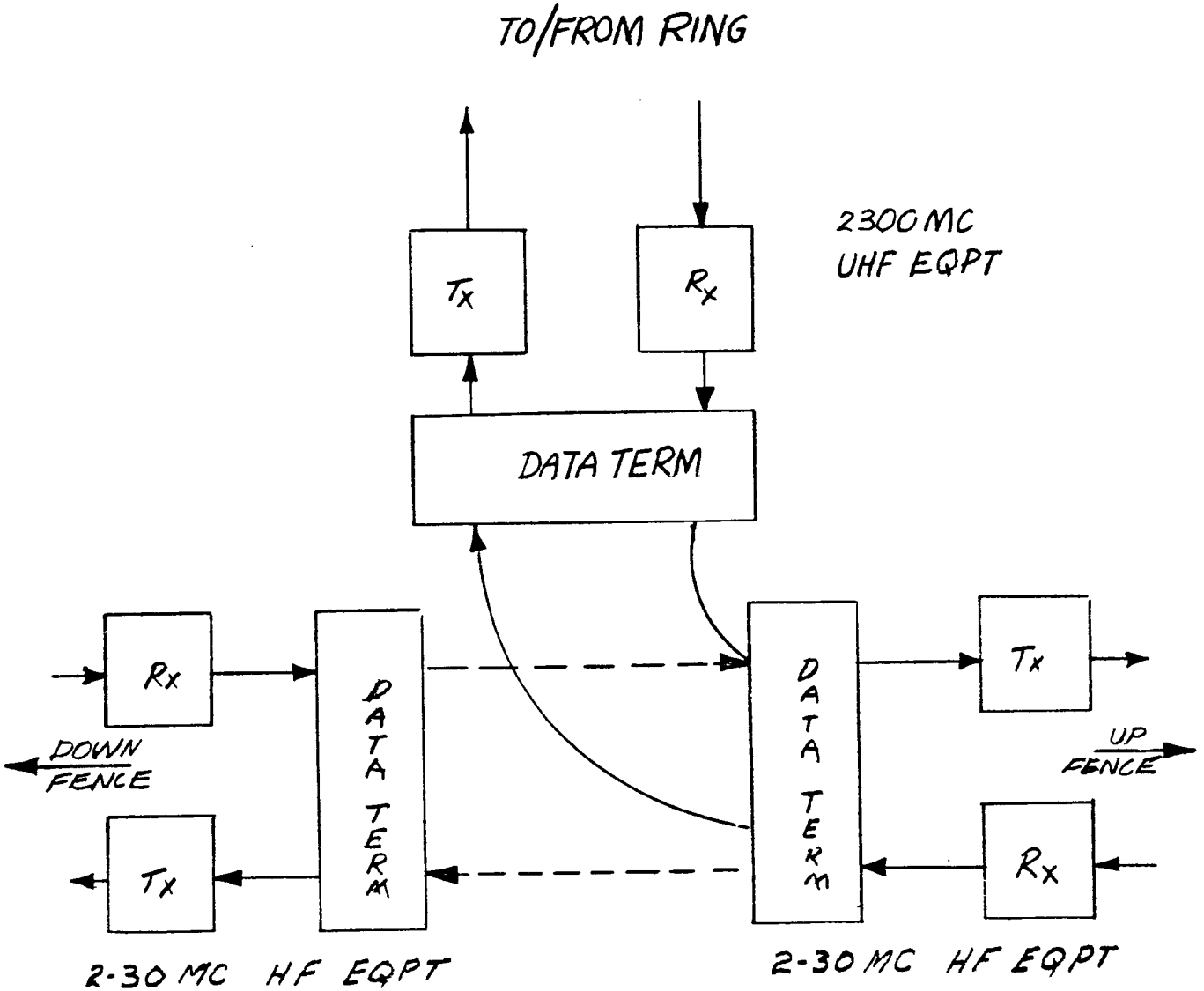


FIGURE D-1 FENCE RELAY STATION

center of the diagram indicate the connections that are used when the station is used as an HF relay only with the UHF pick up from the ring being done farther down fence. These sections are made up of transmitters, receivers and data terminals. Voice paths through the equipment have been omitted for simplicity. Further details of each section will be given after a short discussion of the use of this chain of station.

The path from ring to Cape includes one station utilizing its UHF equipment to and from the ring plus one or more stations utilizing both halves of the HF equipment up and down fence as a relay. The most northerly and most southerly station require only one half the HF equipment. The fence stations shown in figure 4-3 of the main text were selected to coincide with stations used for other programs in so far as possible. They are closer together than necessary but this is an advantage since it provides some built in alternate routes so important to overall reliability. Each hop is on the order of 750 statute miles and therefore there are many combinations that might be used in returning to the Cape. Although so far as managing the chain is concerned, it would be simple to always relay through the next northerly station, however, this would not be the most desirable from a frequency assignment basis. Single hops should be utilized when propagation is good and multiple hops when propagation is bad. In this regard, it is interesting to note that north-south paths are to be preferred to east-west paths for reliability on HF circuits.

Returning to further details of the stations, figure D-2 shows the UHF section. It includes a tracking antenna, a diplexer, a preamplifier, a receiver, a data terminal and a transmitter. The antenna gain is selected at 48 db in order to reduce the equipment in the satellite to a minimum weight.

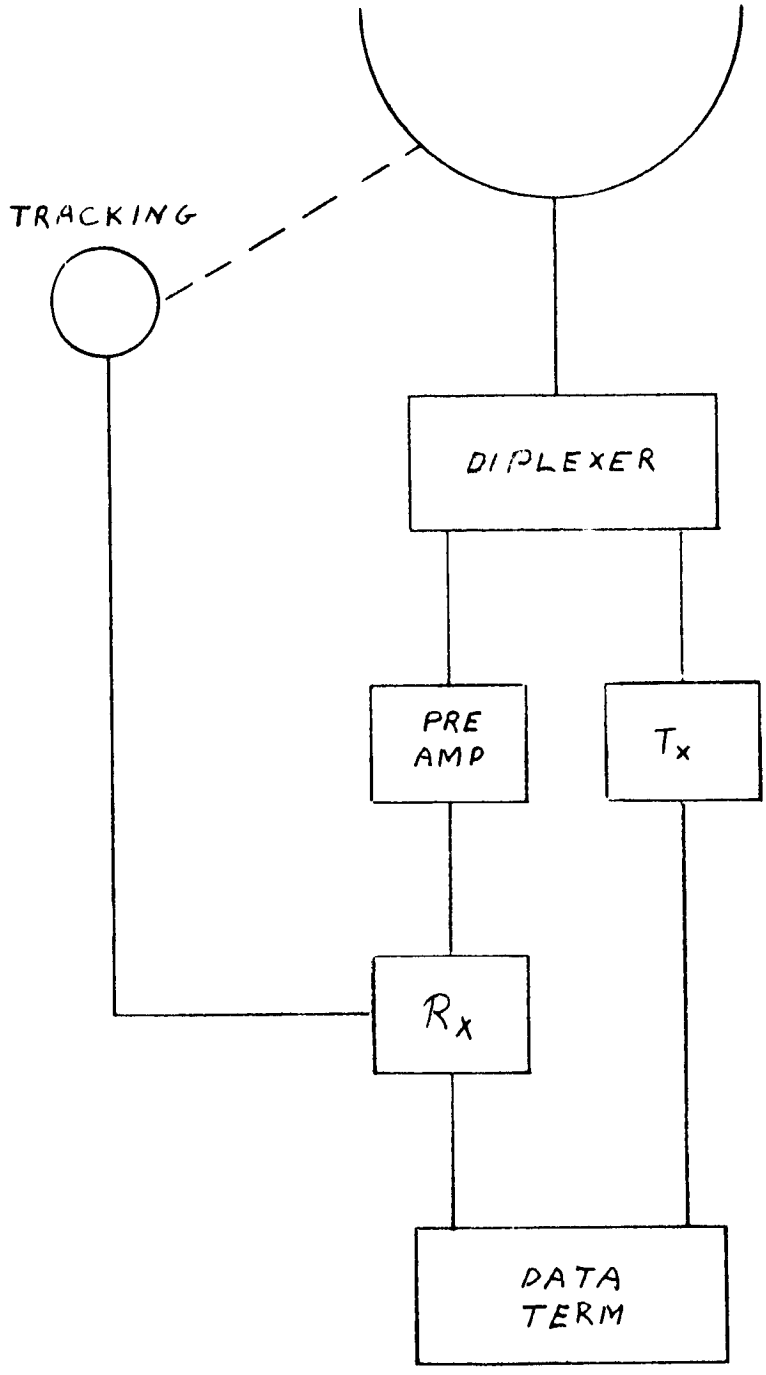


FIGURE D-2 2300 MC SECTION OF FENCE RELAY STATION

This gives a 0.5 degree beam width which requires accurate tracking. The receiver is of advanced design with a 200^{o} k noise temperature. The transmitter is quide modest, requiring only ten watts. The data terminal has been included since it is believed that the most flexible system is that in which the data is demodulated to binary form and coverted to multi-channel binary form at a slower individual rate to meet the requirements of propagation via the HF medium. Even if the proper composite form already exists as transmitted from the space craft, relaying through several HF hops requires re-generation so that terminal equipment is required at each station. In addition, there are probably some channels of data that have higher priorities than others contained in the composite signal from the UHF circuit. Demodulation would be required to be sure to forward the priority portions even under conditions of reduced capacity up the fence.

The HF section of the fence station consists of several HF systems in parallel each having a capability of 16-kc as shown in figure D-3. Separated transmitting and receiving antennas are shown. Three transmitters are combined on two single antennas. However, the filters are almost the same price as large directive antennas and if land is no problem, separate antennas might be used for each transmitter. The transmitter line up includes the antenna, the filter, the power amplifier, the exciter, and the data terminal. The antenna suggested is a vertical logarithmically periodic antenna covering 2-30 mc having a gain of 7 db over a quarter wave monopole. A bandpass filter tuned to the transmitting frequency having an attenuation of 20 db for the signal from the adjacent transmitter is required. The PA's are manually tuned 10 kw PEP 2-30 mc linear amplifiers. The exciters are manually adjustable in 1-kc steps from 2-30 mc. The data

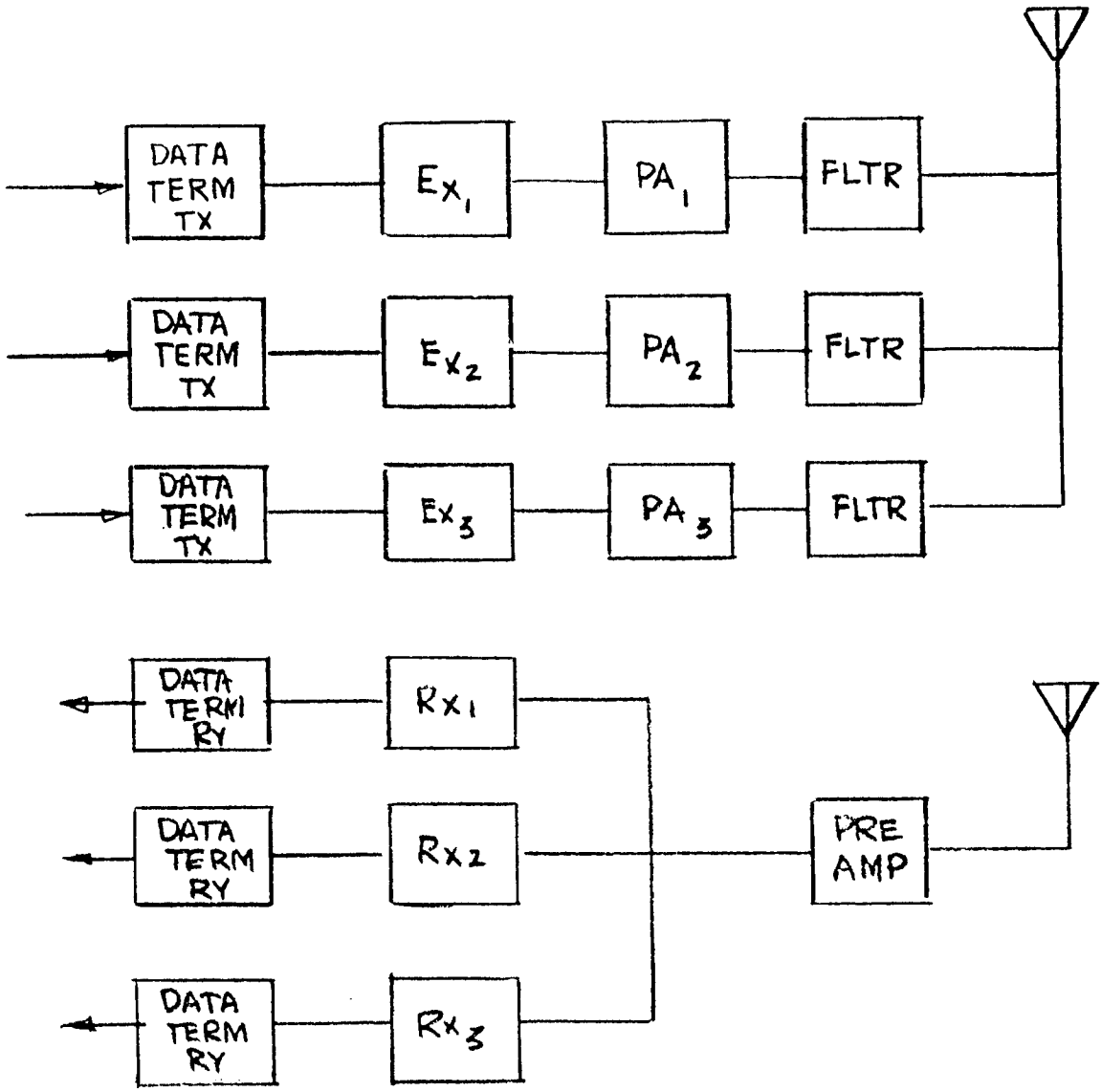


FIGURE D-3 HALF OF THE HF SECTION OF THE FENCE RELAY STATION

terminals are four phase state, frequency multiplexed, bit synchronous equipment giving about 1 bit per second per cycle of bandwidth (Collins Kineplex). Figure D-3 shows 3 x 16-kc or 48-kc of bandwidth. The limitation here is not equipment but 48-kc appears to be near an upper limit of the possible frequency assignments. Of course such a system could be implemented in steps starting with a single 16-kc band or less.

Figure D-4 is a table showing fence costs, $26,000,000 for ten years of operation.

ITEM COST $	BLDG 100K	HF ANT 7 db 2-30 mc 20K	FILTER 20K	PA 10 kw 15K	EX 2-30 mc 10K	Pre Amp 2-30 mc 10K	RX 2-30 mc 10K	TERM KPX 10/B/S (BW = B/S)	2300 mc RF 200K	DATA TERM 500K
STN										
1	1	2	3	3	3	1	3	48K	1	1
2	1	4	6	6	6	2	6	96K	1	1
3	1	4	6	6	6	2	6	96K	1	1
4	1	4	6	6	6	2	6	96K	1	1
5	1	4	6	6	6	2	6	96K	1	1
6	1	2	3	3	3	1	3	48K	1	1
TOTAL	6	20	30	30	30	10	30	480K	6	6
	600K	400K	600K	450K	300K	100K	300K	4800K	1200K	3000K

Total radio equipment and buildings 11,750K

PWR 88K/STN/yr at 10¢ KWH 100 KW

Personnel 75K/STN/yr at 15K/M x 5 min

Repairs 1,300K/yr

For ten years - 26 Million

Figure D-4 - Fence Costs for 48 KC Bandwidth

APPENDIX E

THE SATELLITE RING

E-1. GENERAL

A ring of satellites as shown in figure E-1 must have certain features to allow flexible operation. This ring may be compared to a microwave relay link across country. The distance between stations becomes greater since the line of sight range is greater. A gap between satellites No. 1 and No. 12 in the ring prevents closed loop oscillations in the system. There is no break in coverage however since satellite No. 1 and satellite No. 12 overlap each other in coverage. Several entrances will be provided at each satellite for communication from the ground OLF, S/C, and payloads.

E-2. FREQUENCY PLAN

Employment of the previously mentioned principles will result in the development of a frequency plan. This plan is based upon multi-band filters for the OLF, S/C, astrotug and satellite ring. One of the many possible frequency plans is shown in figure E-2. This figure indicates the number of frequency bands which would be required. The specific system under final consideration will determine the number of band-filters, transmitters, receivers etc. that are necessary for the system. In figure E-2 f_1, f_2, etc. indicate the number of bands required in the communication plan. These bands determine the number of band-filters required.

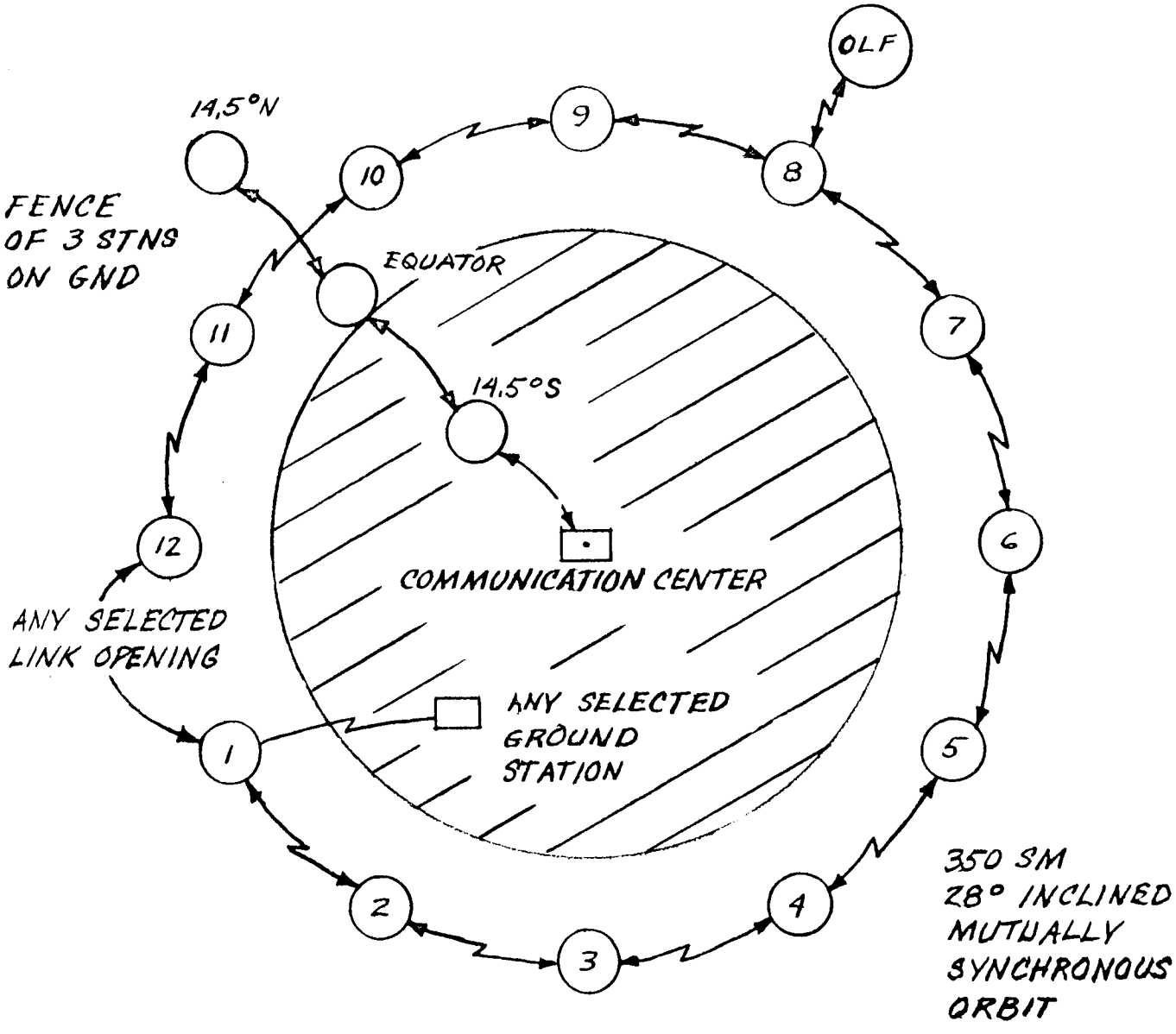


FIGURE E-1 OLF- SATELLITE RING-
GROUND FENCE- CONCEPT

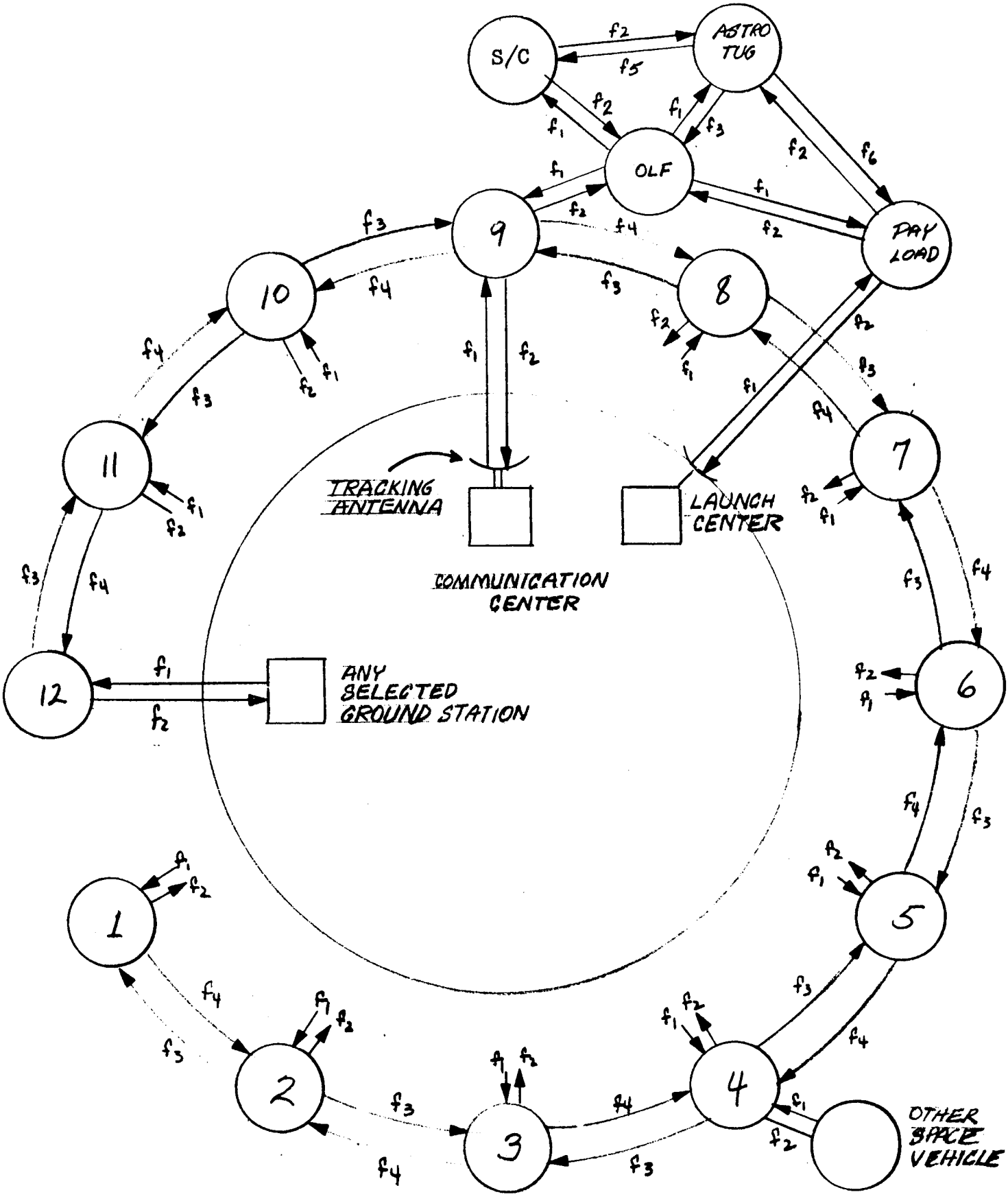


FIGURE E-2 RING FREQUENCY PLAN

Calculations of candidate hardware for the satellite ring are presented in the following paragraphs. They are based on a 40 db signal-to-noise ratio at each station or a minimum overall signal-to-noise ratio of 30 db. This provides excellent error rates and may be on the conservative side. Any conservatism may be converted directly into added bandwidth at double the bandwidth for each 3 db of signal-to-noise. The 200-kc is I-F bandwidth and for low deviation F-M system must be divided by about 3. Call this 75 kc of base bandwidth with 30 db signal-to-noise ratio from OLF to fence.

The calculations are based on the following concepts:

a. All satellite receivers and transmitters are of elementary variety. Receivers are crystal or traveling-wave tube (TWT) type. Power of the transmitters is held to a low level. Two watts at 2.2 kmc is attainable with all solid state devices. Powers above this require a tube amplifier. Higher solid state powers may be obtainable at the time of unit development.

b. The size of the antenna for the satellite to OLF link is restricted by the orbit of the OLF with respect to the satellite.

c. The size of the satellite antenna is restricted by the degree of orientation accuracy of the satellite.

d. Parametric receivers are allowed in the OLF and a maser receiver is allowed on the ground.

e. An orientable antenna is necessary on the OLF, S/C and astrotug. (mechanical or electronic orientation).

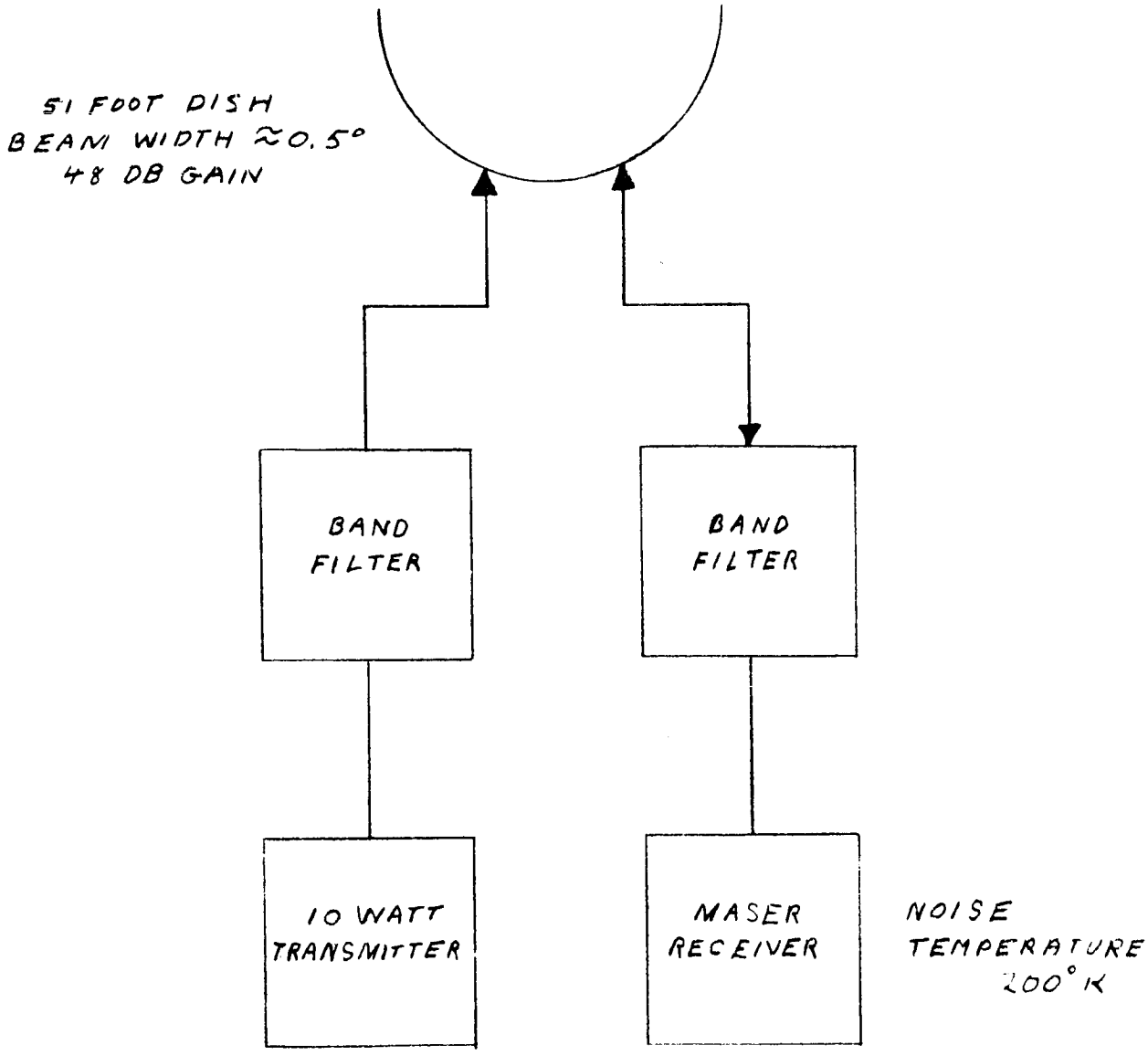


FIGURE E-3 GROUND STATION

Six sets of calculations follow for the six paths involved. Following these calculations is a summary of costs, weights and power requirements for the communication satellites.

E-3. 200-KC SYSTEM

a. Ground Station Range (350 to 1450 miles). Refer to figure E-3.

S/N 40 db = 10^4

Bandwidth 2×10^5

Receiver Noise Temperature 200°K

λ = 5 inches (2300 MC)

Gt = Gain of Transmitting Antenna 3 db (GAIN = 2)

Gr = Gain of Receiving Antenna

Pt = Power of Transmitter in Watts (2 WATTS)

SATELLITE TRANSMIT EARTH RECEIVE

$$Gr = \frac{S/N \; B \; 16\pi^2 \; R^2 \; k \; Tr}{\lambda^2 \; Gt \; Pt}$$

$$Gr = \frac{10^4 \times 2 \times 10^5 \times 16\pi^2 \; (1450)^2 \times (5280)^2 \times 1.38 \times 10^{-23} \times 200}{(5/12)^2 \times 2 \times 2}$$

$Gr = 7.35 \times 10^4$ or $Gr = 48.66$ db

$Gr = \frac{6.3A}{\lambda^2}$ $\quad A = \frac{\pi D^2}{4}$

$A = \frac{Gr\ \lambda^2}{6.3} = \frac{7.35 \times 10^4}{6.3} (5/12)^2$ $\quad D^2 = \frac{4A}{\pi}$

$= \frac{7.35 \times 10^4 \times 25}{6.3 \times 144}$ $\quad = \frac{4}{\pi} \times 2.03 \times 10^3$

$= 2.03 \times 10^3$ $\quad = 0.259 \times 10^4$

$D = 0.51 \times 10^2$

$D = 51$ ft DISH

b. Earth Transmit - Satellite Receive Ground Station Range (350 to 1450 miles)

S/N 40 DB = 10^4

Bandwidth 2×10^5

Receiver noise temperature 870°K (crystal receiver or twt)

$\lambda = 5$ inches (2300 MC)

Gt = Transmitter antenna gain 7.35×10^4

Gr = Gain of receiving antenna (3db) Gr = 2

Pt = Power of transmitter in watts

$$Pt = \frac{B\ S/N \times 16\pi^2\ R^2\ k\ Tr}{Gr\ Gt\ \lambda^2}$$

$$= \frac{2 \times 10^5 \times 10^4 \times 16\pi^2 \times (1450)^2 \times (5280)^2 \times 1.38 \times 10^{-23} \times 870}{2 \times 7.35 \times 10^4 \times (5/12)^2}$$

$= 6.23$ watts

Pt assigned 10 watts

c. OLF → Satellite

OLF → Satellite Range Maximum 2400 miles. OLF has tracking antenna.

S/N 40 db = 10^4

Bandwidth 2 x 10^5

Receiver noise temperature 870°K

λ = 5 inches (2300 MC)

Gt = Transmitting antenna gain 8 ft dish 32.6 db 1825 gain

Gr = Receiving antenna gain = 124 (20.94 db) 2 ft dish

$$Pt = \frac{B \times S/N \times 16\pi^2 \times R^2 \times k\ Tr}{Gr\ Gt\ \lambda^2}$$

$$= \frac{2\times10^5 \times 10^4 \times 16\pi^2 \times (2400)^2 \times (5280)^2 \times 1.38 \times 10^{-23} \times 870}{124 \times (5/12)^2 \times 1825}$$

= 7.95 watts

Use 8 watt transmitter on OLF

d. Satellite → OLF

S/N 40 db = 10^4

Bandwidth 2 x 10^5

Range 2400 MILES

B = 200KC

λ = 5 inches (2300 MC)

Gt = Gain of receiving antenna 8 ft dish (32.6 db 1825 gain)

Gr = Gain of Transmitting antenna 2 ft dish (20.93 db) 124 gain

Tr = 340°K (Parametric Receiver)

$$Pt = \frac{B \times S/N \times 16\pi^2 \times R^2 \; k \; Tr}{Gr \; Gt \; \lambda^2}$$

$$= \frac{2x10^5 x10^4 x16\pi^2 x(2400)^2 x(5280)^2 x1.38x10^{-23} x340}{124 \times 1825 \; (5/12)^2}$$

Pt = 3.1 watts

use 4 watt transmitter

e. Satellite to Satellite

S/N 40 db = 10^4

Bandwidth 2 x 10^5

Range 2400 miles

k = 1.38 x 10^{-23}

Tr = 870°K (crystal or twt receiver)

λ = 5 inches (2300 MC)

Gt = Gain of transmitting antenna 8 ft dish (32.6db) 1825 gain

Gr = Gain of receiving antenna 8 ft dish (32.6db) 1825 gain

Pt = Power of transmitter watts

$$Pt = \frac{13 \; S/N \; 16\pi^2 \; R^2 \; k \; Tr}{Gr \; Gt \; \lambda^2}$$

$$Pt = \frac{2x10^5 x10^4 x1.579x10^2 x1.60\%x10^{14} x1.38x10^{-23} x.87x10^3 x1.44x10^2}{(1.6[illegible])^2 \times 2.5 \times 10^7}$$

Pt = 1.04 watts

$$Gt = \frac{6.3A}{\lambda^2} = \frac{6.3(4)^2\,\pi}{(5/12)^2}$$

$$= 1825$$

$$= 32.6 \text{ db}$$

$$\text{Beam } \alpha = \frac{60\lambda}{D} = \frac{60 \times 5}{8 \times 12}$$

$$= \frac{120}{72} = 1.67$$

$$= 3.340$$

f. OLF to Astrotug

Range in miles desirable = 300

$$Gr\ Gt = \frac{B\ S/N \times 16\pi^2 \times R^2\ k\ Tr}{\lambda^2}$$

$$= \frac{2\text{x}10^5\text{x}10^4\text{x}16\ ^2(300)^2 \times (5280)^2\text{x}1.38\text{x}10^{-23}\text{x}870}{(5/12)^2}$$

$$= \frac{2\text{x}10^9\text{x}1.579\text{x}10^2\text{x}2.51\text{x}10^{12}\text{x}1.38\text{x}10^{-23}\text{x}.87\text{x}16^3\text{x}1.44\text{x}10^2}{25}$$

$$= \frac{2\text{x}1.579\text{x}2.51\text{x}1.38\text{x}.87\text{x}1.44\text{x}10^3\text{x}4}{.25}$$

$$= 55 \times 10^3$$

$$= 5.5 \times 10^4$$

$$Gr = Gt = 2.34 \times 10^2$$

$$= 234$$

$$Gt = \frac{6.3\pi D^2}{\lambda^2 4}$$

$$4Gt\ \lambda^2 = 6.3\pi D^2$$

$$D^2 = \frac{4Gt\ \lambda^2}{6.3\pi}$$

$$D^2 = \frac{4 \times 234 \times (5/12)^2}{6.3}$$

$$D^2 = \frac{4 \times 234 \times 25}{6.3 \times 144} = 8.22$$

D = 2.87 ft dish with Beam

E-4. SATELLITE EQUIPMENT COSTS

		Wt (lbs)	Watts Input
4 Receivers	30,000	32	40
2 1 watt transmitters	24,000	8	60
1 2 watt transmitter	12,000	4	35
1 4 watt transmitter	15,000	5	50
1 Duplex coupler	15,000	3	5
8 Band pass filters	24,000	8	-
2 8 ft dishes	6,000	60	-
1 2 ft dish	3,000	10	-
1 Slot antenna	800	5	-
	$129,800	135	190
Power Supply System Batteries and Solar Cell	90,000	120	
Control Equipment, Electronic Control Equipment	12,000	20 20	
	231,800	295	
Frame	100,000	100	
	331,800	395	
Development	4,000,000		
Make 12 ft units	3,600,000		
12 spares, test, train	3,600,000		
	11,200,000		

APPENDIX F

CANDIDATE OLF TELEMETRY SYSTEM

F-1. CANDIDATE SYSTEMS

a. General

The initial requirements placed on the OLF communications system were discussed in Section VIII. In the selection of a system with the capability of meeting these requirements, a number of factors bearing on each requirement were considered. These factors include accuracy, reliability, checkout, repair, power requirements, weight, size, and interchangeability. It must be recognized that these relevant factors may vary in importance depending on the mission requirements of the system. The following discussion of condidate systems is based in part on the relative weighting of the system characteristics discussed.

In view of the fact that an analog system, a digital system, and an analog-digital combination provide a representative sampling of the numerous special combinations of telemeters in use, these systems were subjected to close evaluation. Specifically, a PAM/FM/FM, a PCM/FM, and a hybrid system employing PAM/FM, PCM/FM, and FM/FM were chosen for evaluation.

b. Evaluation of Candidate System. A program-orientated evaluation was performed on the candidate systems to determine OLO mission efficacy. The results of this evaluation are shown in figure F-1. The system rated best in each category is scored with a 1. If identical abilities exist for more

than one system, each is scored with a 1. The implications of the scoring in figure F-1 are treated in the following paragraphs.

(1) Modulation (carrier). The FM modulated carrier was chosen because of the FM signal-to-noise improvement and because of compatibility with projected ground facilities and other OLO communications.

FIGURE F-1

Comparison of Candidate Systems

	PCM/FM	PAM/FM/FM	HYBRID (PCM/FM/FM) (PAM/FM/FM) (FM/FM)
Optimum carrier modulation	1	1	1
Data format	1		1
Bandwidth (minimum)	1		
Transmitter power (minimum)	1		
Data storage	1		
Self-check	1	1	1
Calibration	1		
Accuracy	1		1
High frequency information		1	1
Analog information		1	1
Threshold signals		1	1
Flexibility	1		1
Reliability	1		

FIGURE F-1 (Cont)

Comparison of Candidate Systems

	PCM/FM	PAM/FM/FM	HYBRID (PCM/FM/FM) (PAM/FM/FM) (FM/FM)
Maintainability	1		
Logistics	1		
Compatibility	1		1
Power	1		
Size and weight	1		
Parallel data transmission			
Score	15	6	10

(2) Data format. Because a major portion of the data will be in digital form and because of required accuracy of certain data (e.g., guidance parameters), a system which will accept and transmit digital data was chosen. The PCM system lends itself to various code formats which provide synchronization and reduction in total RF power requirements. The PCM format further provides compatibility with digital computers, which eases the data reduction burden.

(3) Bandwidth and Transmitter Power. To achieve an optimum system with respect to minimum launch-vehicle boost requirements, a minimum communications bandwidth must be maintained for the reason that an increase in bandwidth requires an increase in operating power, which in turn prescribes an increase in the weight of the orbital equipment.

A PCM/FM system will require approximately 1/8 the bandwidth of the FM/FM or the hybrid system. Through the use of orthogonal coding, PCM will require approximately 1/20 as much energy as a conventional FM/FM telemetry system in transmitting equal amounts of information. Discrete pulse position modulation requires high peak powers, and sub-carrier frequency modulation requires the generation of a large number of stable frequencies.

(4) Data Storage. It is assumed that continuous data transmission will be required during initial orbit, rendezous, and launch. During other periods it is projected that the data will be transmitted only at preset periods to preselected stations. To meet these requirements, two magnetic storage units with minimum time correlation capabilities will be required, one to handle digital data and the other to record analog signals. Because of high bit rate encountered during playback, a PCM system is superior.

(5) Self-Check Features. Because of the tie-in with the ground complex, the communications checkout equipment on board the OLF can be considerably reduced if the major portion of the communications checkout burden is placed with the ground. This philosophy should be incorporated into the OLF communications system. Failure detection is easier in a PCM system because of the go no-go characteristics of the digital circuitry.

(6) Calibration. Because of the required long life of the communication system, techniques which depend on calibration for data correlation are not suitable. A PCM system, because of its digital characteristics, most nearly meets the data correlation requirement. PCM exhibits none of the

errors due to zero drift, distortion, and sensitivity change seen in the FM/FM, PDM, and PAM systems.

(7) Accuracy. It is projected that the required data accuracy will fall into two ranges, one covering from 7 to 2 percent and the other from 1 to 0.1 percent. Through quantizing, either the PCM or hybrid system has this capability. The PCM system provides variable accuracies which can be programmed into the system through the control of the bit rates. The FM/FM system cannot provide the 0.1 percent accuracy required by many of the anticipated guidance parameters.

(8) High Frequency Information. The complexity in programming, multiplexing and encoding sets a practical limit on the range of information rates and accuracy that can be covered by PCM. Low accuracy, high frequency data is best transmitted by FM/FM or the hybrid system.

(9) Analog Information. The PCM system has no capabilities for transmission of analog information. No requirement for analog data transmission is anticipated for the OLF.

(10) Threshold Signals. A PCM system exhibits a well defined threshold which occurs when random noise exceeds one-half of the peak-to-peak signal input to the decision circuit for any significant amount of the detection time. The other systems can provide data of degraded quality in a high noise environment. Threshold signals are not expected, however, in operations involving the OLF where adequate transmitted power will be available to provide working safety margins.

(11) Reliability. Reliability is dependent upon the techniques used in the development and manufacture of a system and upon the type of system chosen. PCM logic lends itself readily to solid state circuitry which provides better reliability. In addition, marginal operation is less frequent in digital circuits. A more detailed analysis of reliability is given in Appendix G.

(12) Maintainability. By use of self-check features for detection and isolation of subassembly failures and by use of the OLF checkout equipment for failure detection and isolation of module failure, it is projected that a PCM system will require less maintenance than either of the systems which have the capability of handling analog information. Detailed maintenance and logistics concepts are given in Appendix H.

(13) Logistics. By the use of PCM digital building blocks identical to those used in other OLF equipment and by repair and maintenance at the subassembly and module level, it is felt that the spares requirements can be reduced substantially.

(14) Compatibility. Coded data will be used in most OLO operations; therefore, a PCM system is most compatible.

(15) Size and Weight. Microminiature circuit techniques can be employed to a greater extent in the PCM system; it is estimated that a 25 percent size and weight reduction can be achieved in the PCM system over the other two.

(16) Parallel Data Transmission. The PCM system has no capabilities for parallel data transmission; however, no such requirement is projected for the OLF telemeter.

FIGURE F-1 (Cont)

Comparison of Candidate Systems

	PCM/FM	PAM/FM/FM	HYBRID (PCM/FM/FM) (PAM/FM/FM) (FM/FM)
Maintainability	1		
Logistics	1		
Compatibility	1		1
Power	1		
Size and weight	1		
Parallel data transmission			
Score	15	6	10

(2) Data format. Because a major portion of the data will be in digital form and because of required accuracy of certain data (e.g., guidance parameters), a system which will accept and transmit digital data was chosen. The PCM system lends itself to various code formats which provide synchronization and reduction in total RF power requirements. The PCM format further provides compatibility with digital computers, which eases the data reduction burden.

(3) Bandwidth and Transmitter Power. To achieve an optimum system with respect to minimum launch-vehicle boost requirements, a minimum communications bandwidth must be maintained for the reason that an increase in bandwidth requires an increase in operating power, which in turn prescribes an increase in the weight of the orbital equipment.

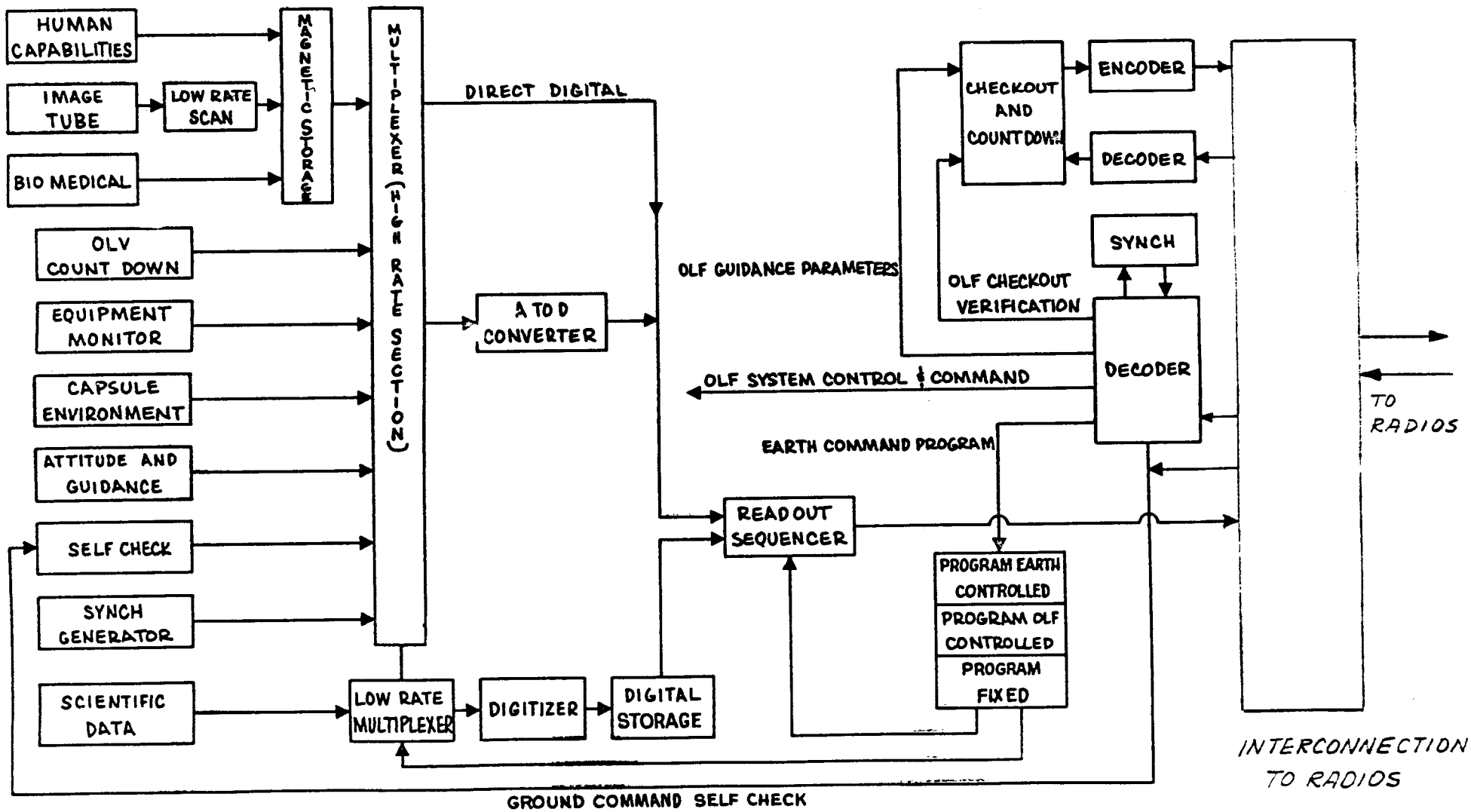


FIGURE F-2 PCM SYSTEM

FIGURE F-3

Projected Electrical Characteristics of
OLF Telemetry System

Storage	Total Bits	Record Speed	Play Back Speed		
Magnetic	700 K	200 B/Sec	5 KB/Sec		
Digital	600 K	15 B/Sec	40 KB/Sec		
A to D Converter	Signal Range	Rate	Max. Accuracy	Code Format	Bits per Word
	100/1	40 KB/Sec	0.1%	Binary	5 to 10
Multiplexer	Rate	Signal Range	Error		
	5 KC	100 to 1	0.1%		

FIGURE F-4

Physical Characteristics - Basic Telemetry System

Item	Assembly Number	Weight (Pounds)	Power (Watts)	Volume ($ft.^3$)
Signal Conditioning	1	15	50	0.18
Multiplexer	4	60	28	0.72
Sync Generator	1	15	7.2	0.18
Encoder	2	30	14.4	0.36
Programmer	1	15	10.2	0.18
Storage Unit (Electronics)	1	15	22.2	0.18

FIGURE F-4 (Cont)

Physical Characteristics - Basic Telemetry System

Item	Assembly Number	Weight (Pounds)	Power (Watts)	Volume ($ft.^3$)
Decoder	1	15	14.4	0.18
Checkout and Countdown	1	15	20	0.18
XMTR (Ground)	0.5	7.5	5	0.09
XMTR (S/C)	0.5	7.5	22	0.09
Receiver (Ground)	0.5	7.5	20	0.09
Receiver (S/C)	0.5	7.5	10	0.09
Power Supplies	2	30	80	0.36
Storage Unit (Mechanical)		3	4	0.10
Antennas		20		
Cables and Connectors		30		
Totals (Basic Tele System)	16.0	293.0	307.4	2.98

FIGURE F-5

Physical Characteristics
Additional Telemetry Equipment

Item	Assembly Number	Weight (Pounds)	Power (Watts)	Volume ($ft.^3$)
Sensors		40	40	2
Narro-band video (Transmit Only)	2	30	50	0.36
Control and display		20	20	2
Totals	2	90	110	4.36

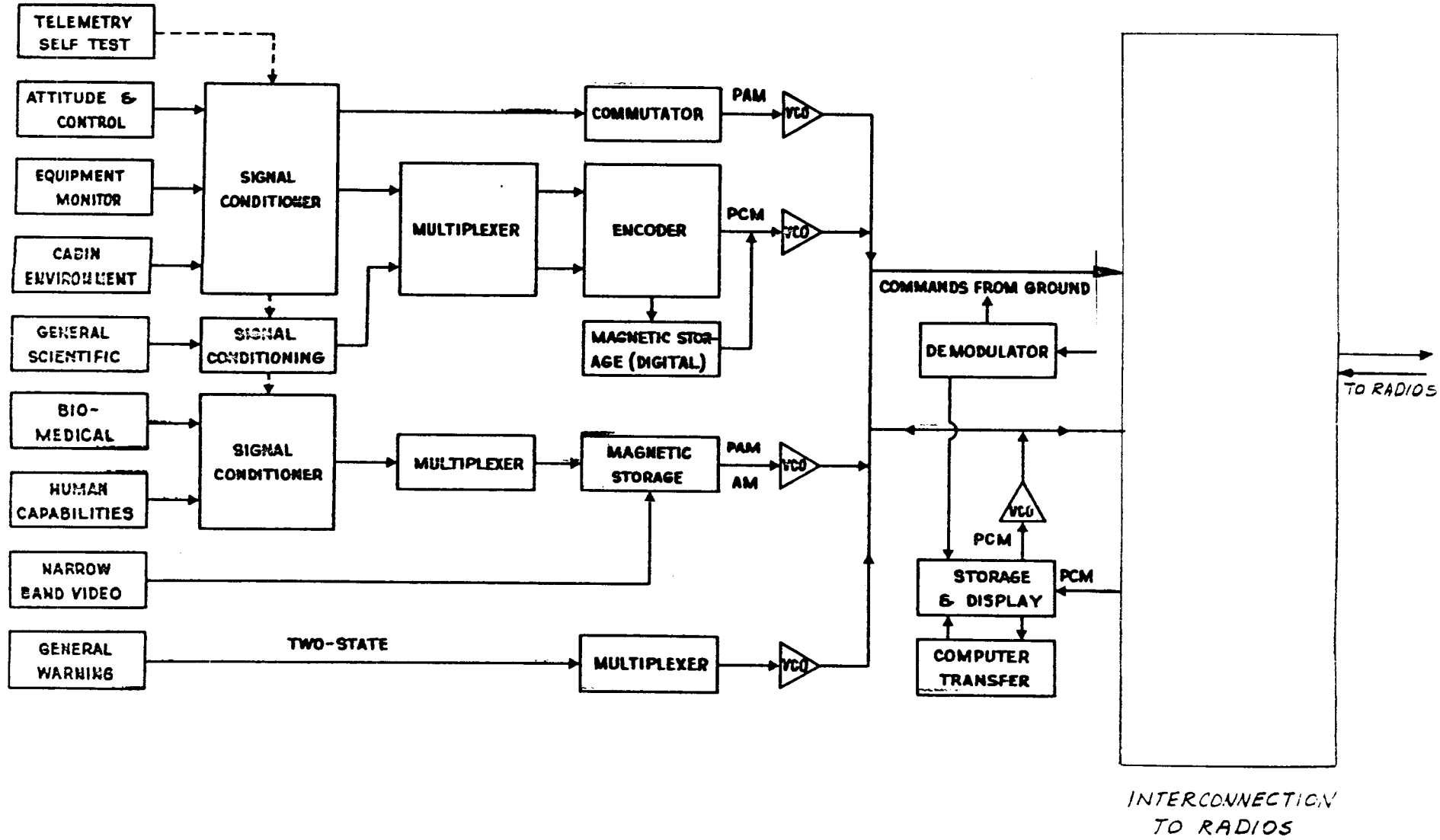


FIGURE F-6 ALTERNATE HYBRID SYSTEM

The estimated total values of weight, size, and power for the OLF communication system including spares are summarized below:

	Basic System	Spares
Weight	383 pounds	219 pounds
Size	7.34 ft.^{3}	3.99 ft.^{3}
Power	417 watts	75 watts

(1) Encoding. Straight binary coding will satisfy the OLF requirements, although the wide range of signal accuracies, signal levels, and signal information rates coupled with identification difficulties might make the use of redundant coding schemes mandatory. The binary format to be used will require a 14 bit word. Ten bits will be required for data identification, three bits for word identification, and one bit for parity. Coding of binary data to an accuracy of 0.1 percent requires a minimum 10 bit data word. The 10 bits used to identify the data signal amplitudes provide readout to 1/1024 part, which provides an accuracy of better than 0.1 percent for the high level signals. The threshold of the comparator amplifier in the encoder (see figure F-7) will limit the accuracy obtainable from the low-level, 0 to 50 mv signal. This threshold limitation makes only the first eight bits significant or limits identification of the data signal to 1/256 part. This provides an encoding accuracy of 0.4 percent for the low level signals.

The output of the multiplexer will be introduced into the encoder assembly (see figure F-7). The sample and hold circuit will look at this signal and store the voltage

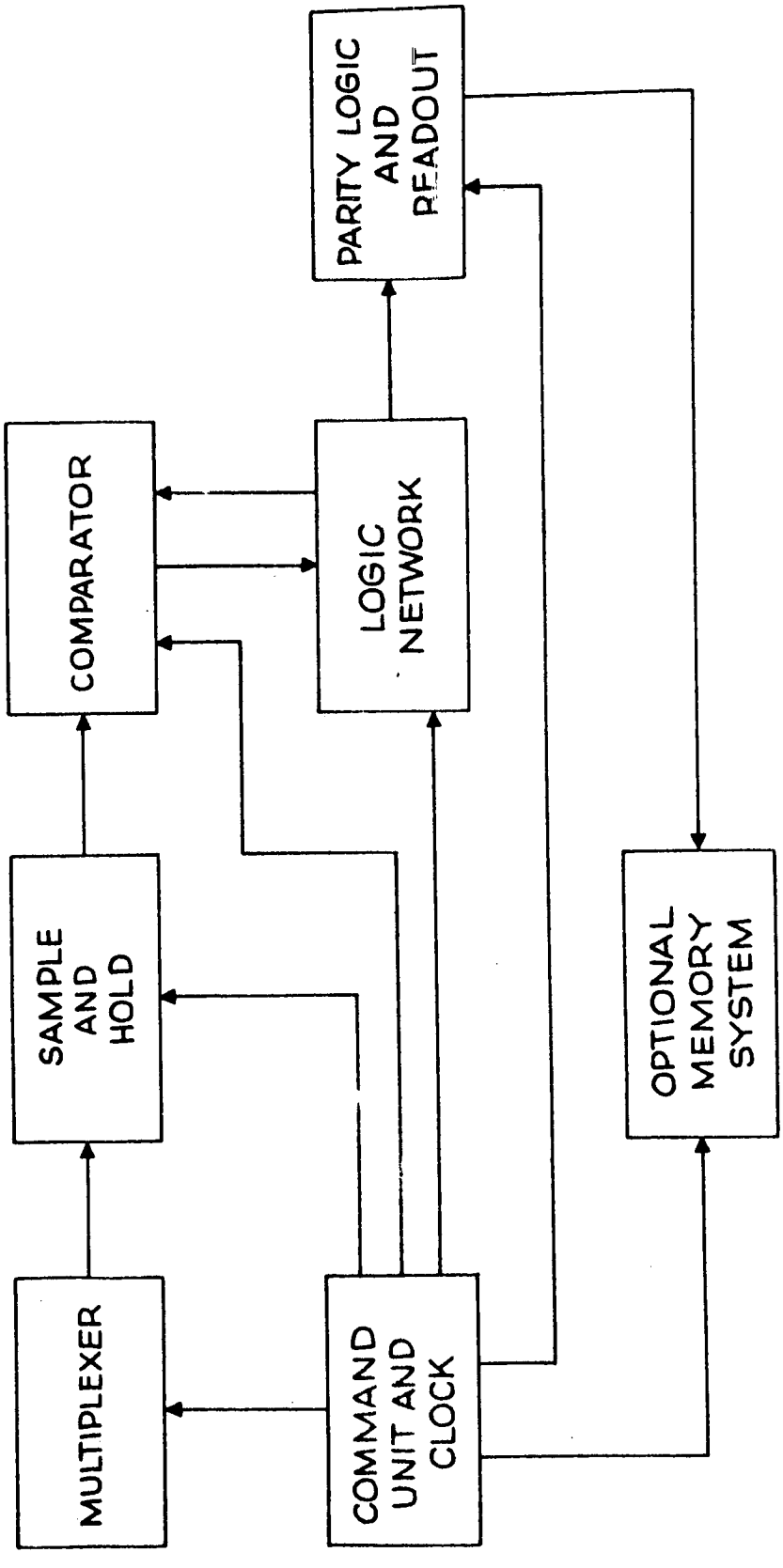


FIGURE F-7 DECODER SYSTEM

on a capacitor through a compensated emitter follower amplifier. At the end of the sampling period, the charge stored on the capacitor will set the level of the output on the hold amplifier, which will be fed into one side of the comparator amplifier. A reset pulse will activate the digitizer circuit in the logic network, the output of which will provide a reference input to the comparator amplifier. The high and low-level switches will be synchronized through the programmer to allow the comparator amplifier to operate either on a low level 50-mv full scale voltage or a high level 5-volt full scale voltage. This automatic level and accuracy ranging will eliminate the need for a precision dc voltage amplifier ahead of the analog-to-digital converter and will aid in minimizing over-all system errors. The comparator will have a regenerative type output circuit that decides which of two inputs is larger. If the input from the sample and hold is larger than the input from the digitizer circuit, the comparator will not provide an output pulse. The next most significant bit (1/4 full scale voltage) is then added to the most significant voltage. Should the input from the digitizer network now be larger than the input from the sample and hold, the comparator will produce a pulse output to the encoder to remove the 1/4 full scale voltage from the comparator. This process of comparing continues through the full scale of ranges: 1/8, 1/16, 1/32, and so on. Each time the comparator generates a pulse, the voltage introduced from the digitizer circuit discrete scale is removed from the comparator, and when the comparator does not generate a pulse, the discrete scale voltage is added to the input to the comparator. This is repeated for the 10 bit encoding of the analog sample, and a parity bit is added in the serial pulse train from the parity logic circuit. Frame synchronization codes will be generated

by a frame sync logic circuit. The frame sync will be started by a reset output from the commutators and will provide one word for frame sync after each of the data channels has been sampled.

(2) Multiplexing. It is expected that the multiplexing unit will be extremely flexible to fulfill system requirements, including those concerning a wide range of input signals, sampling rates, and operating modes. All solid-state devices will be used, and both super- and subcommutation techniques together with different rate sections will be employed.

The multiplexer will be built in sections. Each section will be programmed to operate on a predetermined sampling plan. This sampling plan may be changed on a time-base command or on an earth-to-vehicle command. The predetermined sampling plans will include a discrete number of individual channel sampling rates in any frame to make the sampling rate compatible with the frequency content of the sample. The sampling plan will take into account data priorities and available transmission bandwidth. The multiplexer will have a low level section (0 to 50 mv) and a high level section (9 to 5 V). Figure F-8 presents a block diagram of the multiplexer and the auxillary equipment associated with it.

(3) Electrical Characteristics. The multiplexer will use a basic solid-state switch consisting of two silicon surface alloy junction transistors with the collectors common. This permits the two emitters to be used as the input and output terminals. See figure F-9 for the basic switch configuration. E_c will be a square wave switching voltage supplied from a programmed matrix. When the switch is open, the polarity of the switching voltage E is as shown, and all

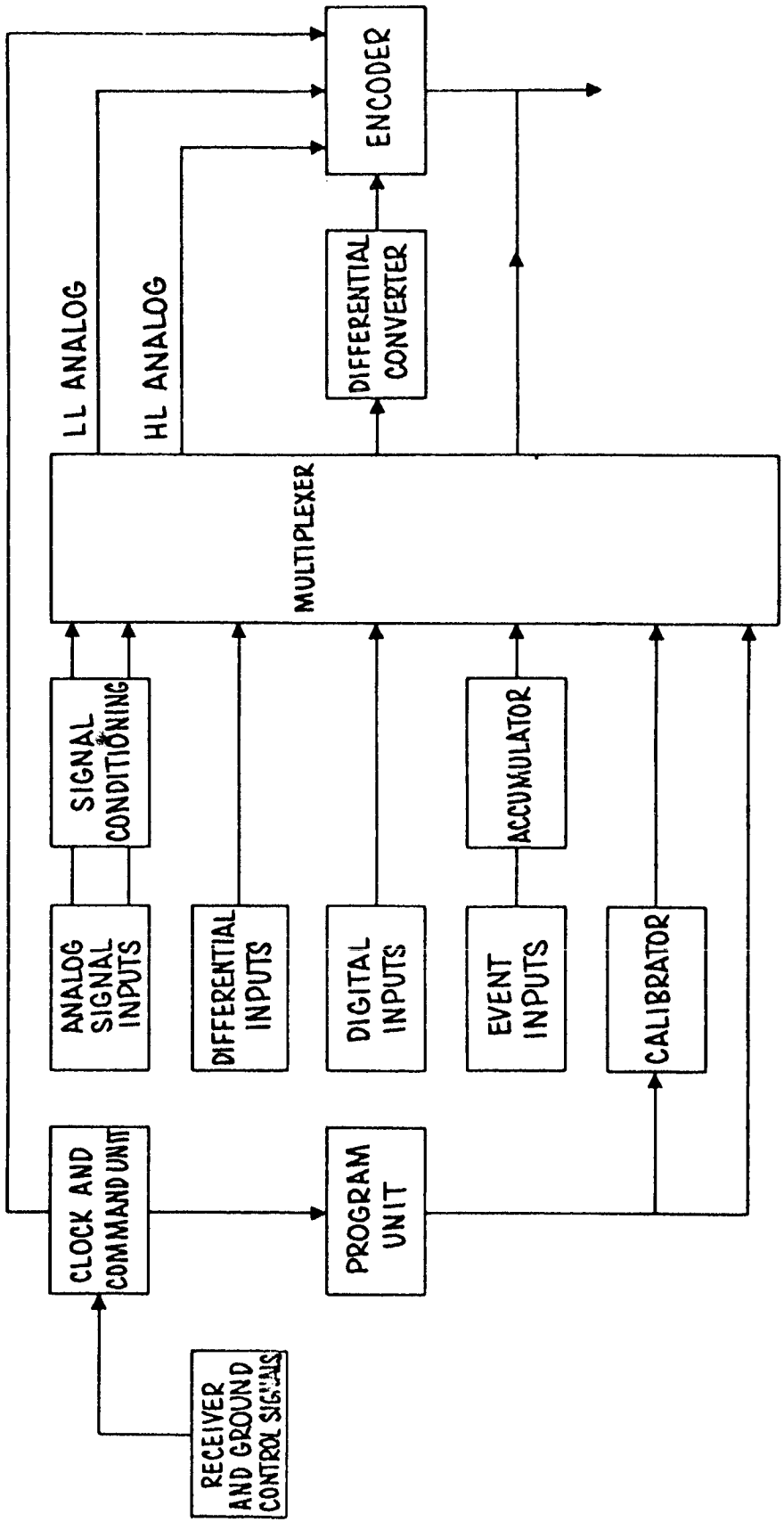


FIGURE F-8 PROGRAMABLE MUTIPLEXER SECTION

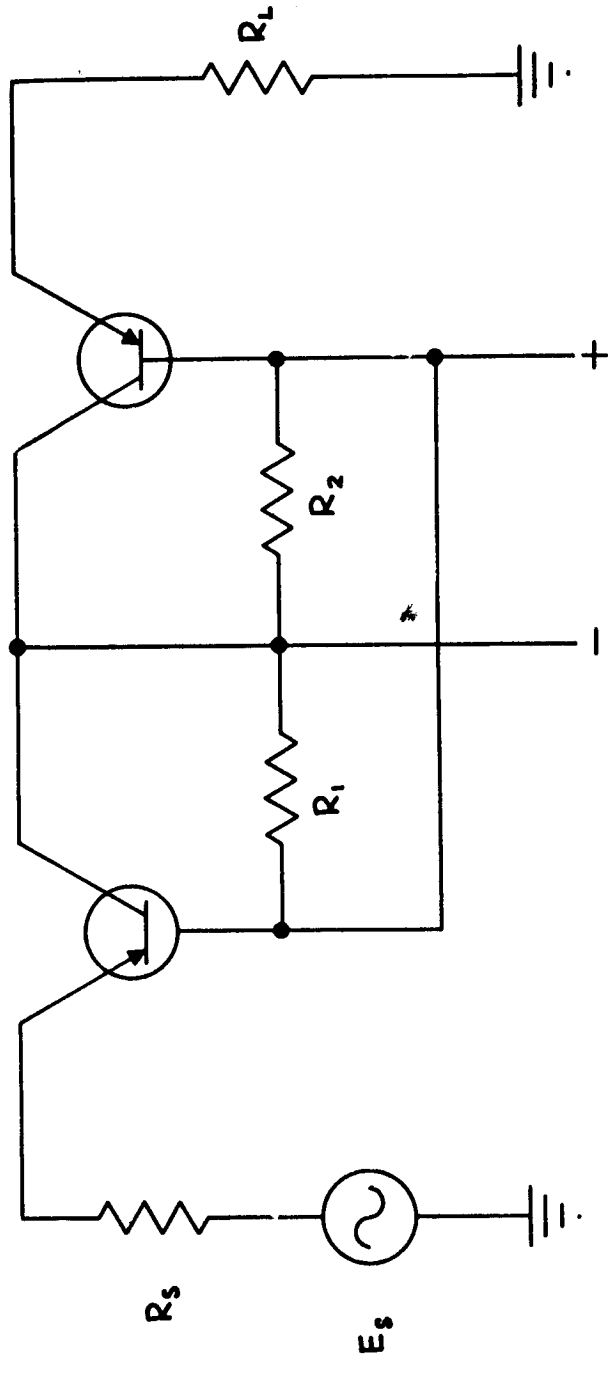


FIGURE F-9 BASIC SWITCH

junctions are reverse biased. The leakage of a back-biased collector junction is 20 times that of the emitter junction. Therefore, using the transistor with the emitter junction in series with the signal and load results in a resistance more than 20 times that of the collector junction. The magnitude of this resistance is greater than 10,000 megohms. When the switch is closed, a low impedance path will exist between the source voltage and load resistor through the saturated junctions of the two transistors. Matched transistors with minimum saturation voltages will be selected. The inverse connection will also result in a greater isolation during switch off time. The switch must also maintain voltage balance over large variations in temperature. This balance can be maintained since both transistors are made of the same material and the junctions will be located in close proximity to each other. The saturation voltage on the transistors will be nearly equal, and the difference will be in tenths of millivolts. Consequently, for a large change in temperature the difference signal and the error signal introduced will be on the order of microvolts.

Sequencing will be provided by a programmed multiplexer driver (see figure F-10). The driver is controlled by a series of gates and a control pulse. Figure F-11 shows the pulse waveform for synchronizing and driving. Complete dc isolation must exist between the sequencing circuitry and the signal path to eliminate the chance of cross coupling. This isolation will be provided through the use of transformer coupling.

(4) Signal Conditioning. For this system a minimum amount of signal conditioning equipment is anticipated. The majority of operations in this area will be performed by

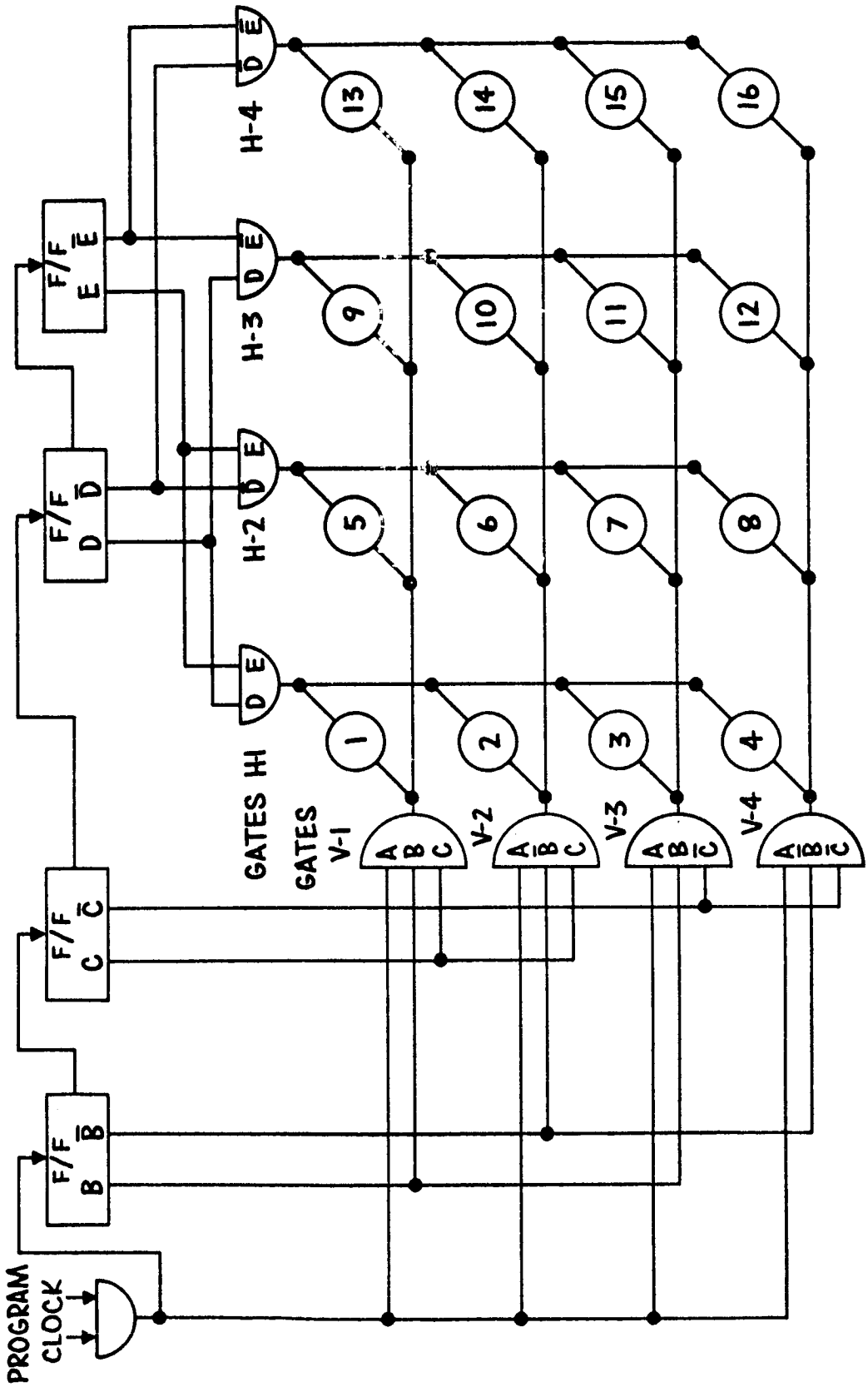


FIGURE F-10 PROGRAMMED MULTIPLEXER DRIVER

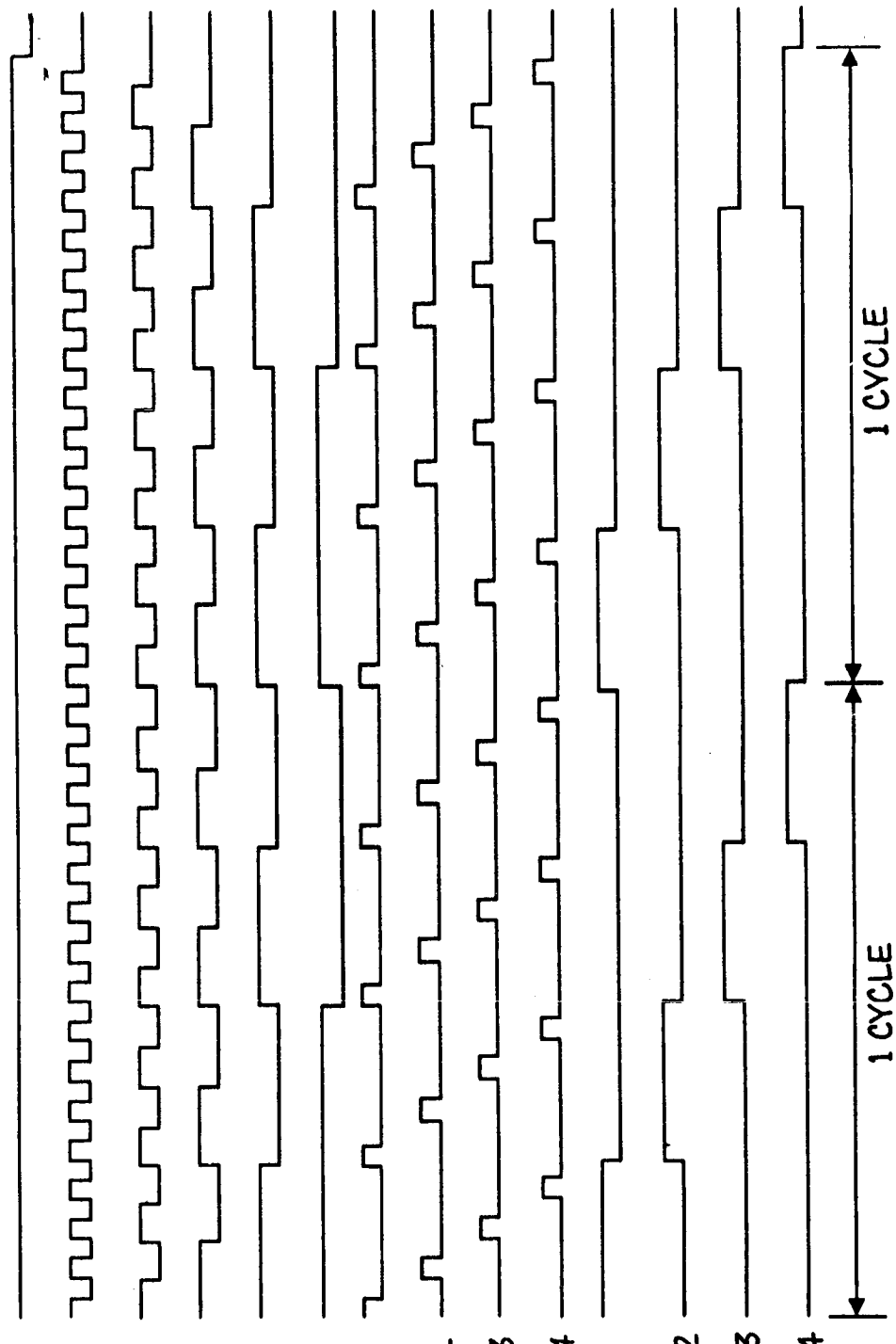


FIGURE F-11 WAVEFORMS OF DRIVER UNIT

automatic level and accuracy ranging in the encoder. Signal conditioning may include phase demodulators, resistive divider networks, magamps, matching networks, and that conditioning necessary to make any special sensor compatible with the telemetry measurement schedule.

Measurements with a differential output such as bridge networks will be processed through a differential converter. The converter will have a constant gain and will provide a high common mode rejection. The input to the normally ac coupled transmitter requires special conditioning to accept a random pulse train of a non-return-to-zero format. Conditioning is necessary to regenerate pulse train signals having varying average dc levels into pulse train signals with a constant dc reference level. The transmitter will then see the "go" or "no go" condition of the last bit, and no dc shift will be experienced by the transmitter when a nonsymmetrical pulse train is applied. This go no-go feature results in polarity synchronization of the transmitter with respect to the input signal from the encoder in the event of an interrupted pulse train or loss of power.

(5) Storage. The OLF will provide storage for four types of data: human capabilities and bio-medical information, picture images, scientific data and checkout and countdown information. It is anticipated that only periodical monitoring of human capabilities and bio-medical information will be required after Phase I in view of knowledge gained during Phase I and previous programs. The results from these periodical tests will be recorded on a controlled, continuous record, magnetic tape. The operator controlling the test will likewise control the record time and rate. The picture image data will also be recorded on this tape.

Scientific data, for the most part, will change slowly with respect to time. This data will be converted to binary format and stored on a magnetic tape with a stepping-type transport which advances upon command. Playback for both recorders will be programmed for relaying stored information to preselected ground stations. Maximum storage time will be 24 hours. Checkout and countdown information will be transferred directly into and from the OLF checkout computer.

(6) Self-check Features. Since a malfunction in the OLF telemetering equipment could generate extraneous signals which have the same characteristics as measured information, some means must be provided to allow the ground equipment to distinguish failures. By periodically transmitting a set of properly selected self-check signals, a ground station can detect a malfunction and isolate the fault to the subassembly level. Wherever possible, this feature will be incorporated into the OLF communications system to avoid excessive weight of checkout equipment in the OLF.

(7) Power Supply Requirements. Power for the telemetry system will be supplied through a dc to dc converter with a multiwinding output. Each output winding will supply the necessary voltage level and isolation for a given subsystem. Balanced diode quads will be used in rectification to ensure minimum offset flux being produced by unbalanced loading. Voltage regulator sections will be provided, where needed, for precision reference sources. Power supply filtering will be used to minimize RF interference (conducted or radiated) being fed to the primary power source.

APPENDIX G

RELIABILITY DEVELOPMENT PROGRAM

G-1. INTRODUCTION

To ensure mission success through equipment reliability, it is necessary to define and pursue a program which will optimize the overall reliability of the system.

G-2. OBJECTIVES

In order to adequately define the proposed reliability program, the following objectives were established and met. The reliability requirements for the communications equipment for each phase of the OLO program were determined. Problem areas associated with the reliability effort were defined, and solutions to these problems were proposed. Then the reliability program deemed necessary to achieve the reliability goals for each phase of the OLO program was determined.

G-3. APPROACH

a. General

By necessity, reliability is of prime importance for the OLO mission from considerations of mission success, human safety, and program cost. To achieve maximum overall

system reliability, both the inherent reliability and the operational reliability of the equipment must be optimized. Inherent reliability can be defined as the potential reliability associated with the basic equipment design. Design techniques that can enhance reliability of the equipment are circuit and/or component redundance, component derating, fail-safe circuitry, and self-healing circuitry. Operational reliability, which can be considered as an extension of the inherent reliability, can be defined as the actual or available reliability of the equipment after fabrication. Thus, the operational reliability is always less than the inherent reliability. Some of the factors contributing to this decrease in reliability experienced in the field are inadequate fabrication processes, errors in human judgement, and unpredictable equipment degradation with time.

To ensure mission success through equipment reliability, a program will be pursued which will optimize the overall system reliability. This will be effected by utilizing fabrication and design techniques that maximize the inherent reliability of the equipment and minimize the effects of inadequate fabrication processes, errors in human judgement, and the degradation effects associated with time.

Initially, the POMS for each phase of the OLO program will be determined. This probability will be a function of environmental stress levels, equipment operating time, and various equipment redundancies and will vary as man's role changes in the maintenance and logistics pattern. The operating reliability will then be the optimum compromise of each of these areas for each phase of the program.

b. Reliability Analysis

Basic to the discussion of reliability is that the total reliability, R_T, of any configuration is the products of the individual reliabilities, R_i, of its constituents; that is,

$$R_T = R_1 \cdot R_2 \cdots R_n = \prod_{i=1}^{n} R_i \qquad (1)$$

Implicit in equation (1) is the requirement that the basic elements R_i be independent in the statistical sense. Except in special cases where interaction can be shown to exist between the R_i's, independence is a defensible and commonly made assumption.

Simple systems with few elements have the possibility of much higher inherent reliabilities than highly complex systems with many parts. Unfortunately the requirements placed upon the systems under proposal demand that the systems be of severe complexity, resulting in the ultimate need for high component reliabilities.

For R_T to be maximized, it is not only necessary that all of the R_i's be large, but it is equally important that they have approximately the same value. For example, efforts to raise the total reliability by increasing some of the contributing reliabilities, for example from 0.99 to 0.995, are ineffectual when there are remaining reliabilities as low as 0.7 or 0.8 in the system. Although high, homogeneous reliabilities are desired, they are not generally realizeable simply because the contributing items are designed for such varied functions. To raise the reliability of an item above a given level is most costly and in some cases, with present or even proposed techniques, extremely difficult.

To overcome this problem it is common to have replacement parts for those whose reliability is low. This philosophy of redundance admits of two extremes:

Case I - The total configuration, R_T, considered and packaged as a single block, has a single spare R_T.

Case II - The total configuration, R_T, considered and packaged as n blocks, has a single spare for each of the n blocks. In each case the total spares are equal to R_T. From reliability considerations alone, what degree of redundancy should be

R_T $\qquad\qquad$ R_1 $\quad R_2 \ldots\ldots\ldots R_n$ $\quad R_T$

R_T $\quad$ Spare $\quad$ Spares $\quad$ R_1 $\quad R_2 \ldots\ldots\ldots R_n$ $\quad R_T$

Case I $\qquad\qquad$ Case II

adopted with the multiplicity of combination for grouping that exists?

To answer this question it is necessary to restate the problem in mathematical terms. A basic block of equipment has a reliability P. It is desired to achieve a higher reliability R by dividing the block into N arbitrary sections and by having replacements for n of the modules which occupy these sections. What is the optimum number of sections, N, and replacements, n, required to achieve this goal?

1 2 3 .. x_i .. n n+1 ..... N Basic block P

1 2 3 .. x_i .. n Spares R

Let P = reliability of basic block

N = total number of sections

n = number of modules which have replacements

N-n = number of modules without replacements

x_i = reliability of module in ith section

P_R = overall reliability of redundant sections

P_N = overall reliability of non redundant sections

R = reliability revised block with redundance.

Since it is assumed that a failure in any one of the modules in the basic block will cause the equipment to fail, the following equation holds

$$P = \prod_{i=1}^{N} x_i \qquad (2)$$

Now consider the ith section; assume it has a spare module. The module for this section and its replacement both have a reliability x_i. For this section to fail it is necessary that both modules fail. The probability of this event is given by $(1 - x_i)^2$. Therefore, the probability of success of this section is $1 - (1 - x_i)^2$. This holds for each section which has a spare. Therefore,

$$P_R = \prod_{i=1}^{n} \left[1 - (1 - x_i)^2\right] \tag{3}$$

The reliability of the non redundant sections is given by

$$P_N = \prod_{i=n+1}^{N} x_i = \prod_{i=1}^{N-n} x_i \tag{4}$$

The overall reliability of the revised block is then

$$R = (P_N)\,(P_R) = \prod_{i=1}^{N-n} x_i \prod_{i=1}^{n} \left[1 - (1-x_i)^2\right] \tag{5}$$

In order to make calculations possible and to get an insight into the problem, assume $x_i = x_j = x$ for all i, j. Then

$$R = x^{N-n} \left[1 - (1 - x)^2\right]^N \tag{6}$$

where

$$x = P^{1/N} \tag{7}$$

c. Results

Figures G-1a, G-1b, G-2a, and G-2b illustrate the achieved increase in reliability as a function of N for basic reliabilities of 0.5, 0.9, 0.95, and 0.99, respectively. Each figure has curves for 25, 50, 75, and 100 percent redundancy. These curves show immediately that reliability increases rapidly with increasing redundancy and rather slowly with increasing N. For example, with a basic reliability of 0.9, division into 4 sections and 100 percent redundancy increases the

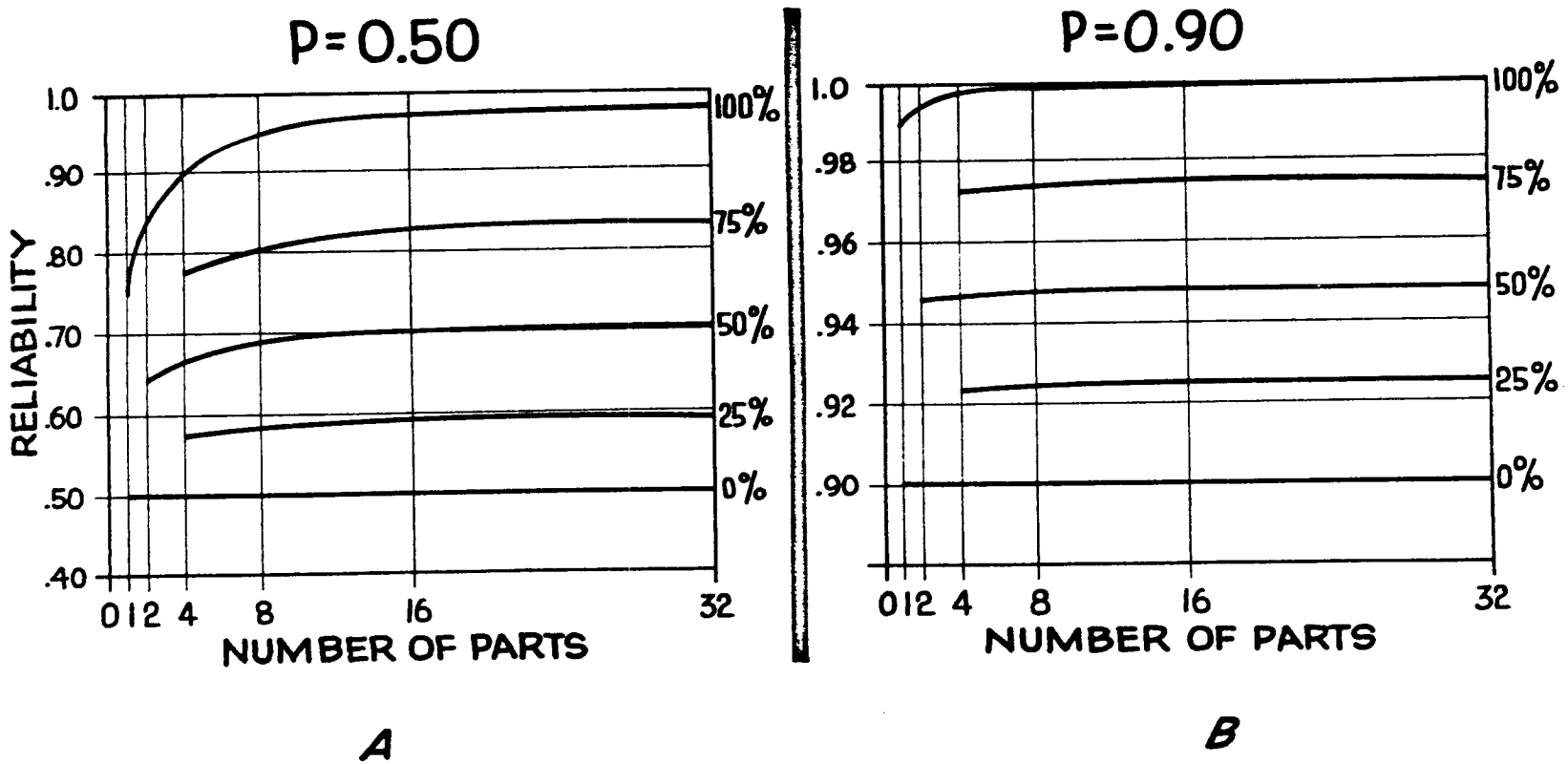


FIGURE G1 BASIC RELIABILITY P=0.50 AND P=0.90

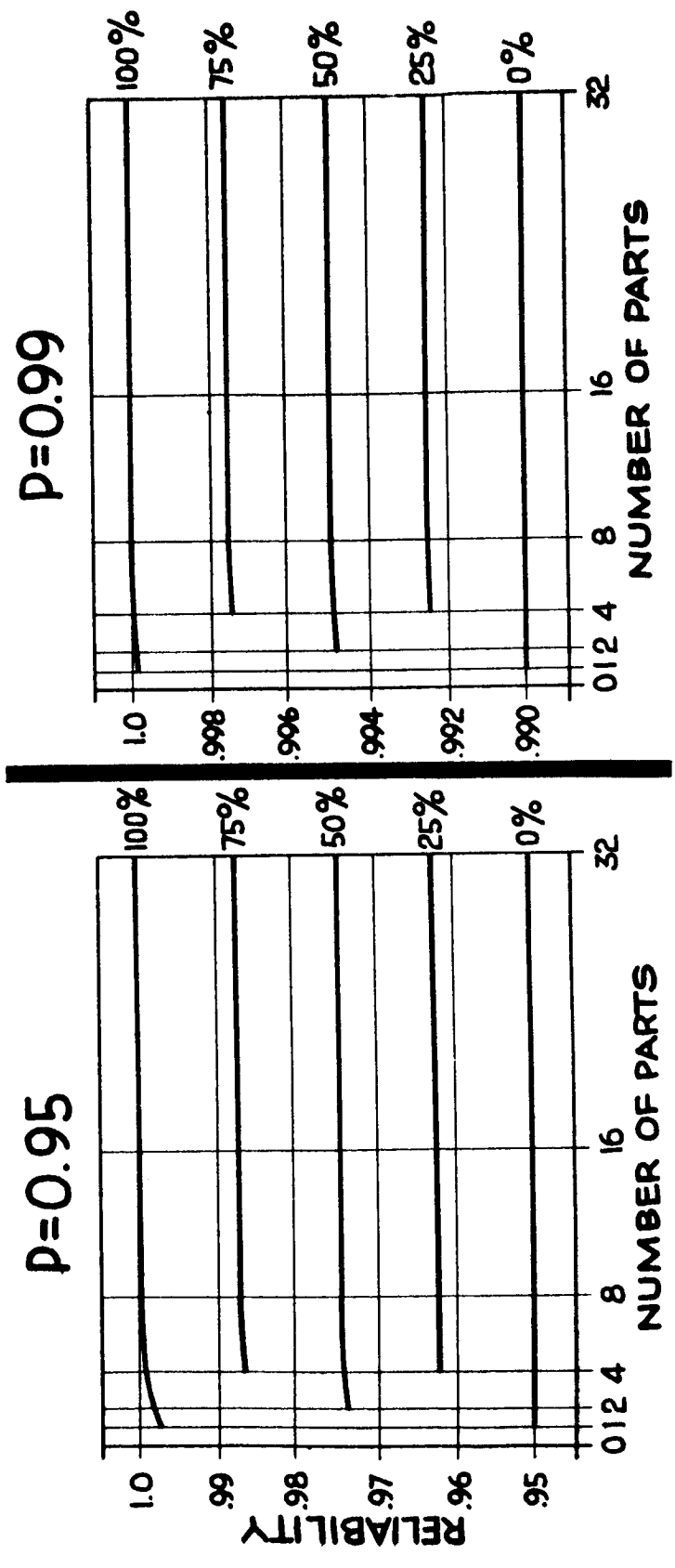


FIGURE G-2 BASIC RELIABILITY
P=0.95 AND P=0.99

reliability of 0.997, whereas division into 32 sections with 50-percent redundancy increases the reliability to only 0.948.

Figure G-3 shows the variation of reliability with weight for prescribed initial reliabilities and percent of redundancy. It is important to observe that 100-percent redundancy incorporated into the design renders the initial reliability of the configuration relatively insensitive to the final reliability. Because the percent of redundancy is such a critical parameter and the weight w for the telemetry system is comparatively small, at least 100-percent redundancy is proposed to achieve the desired reliabilities.

Consider a block of equipment with a basic reliability P. Let this block consist of N identical modules. Let the number of spares be n. Since the spares are completely interchangeable, the new reliability is given by

$$R = 1 - \sum_{s=n+1}^{N} \binom{N}{s} p^{N-s} q^{s} \qquad p = p^{1/N} \tag{8}$$

where p is the reliability of each module
and q is the probability of failure for each module (1 - p)

In contrast to this let another block consist of N different modules, each of which has the same reliability p. Then the reliability becomes

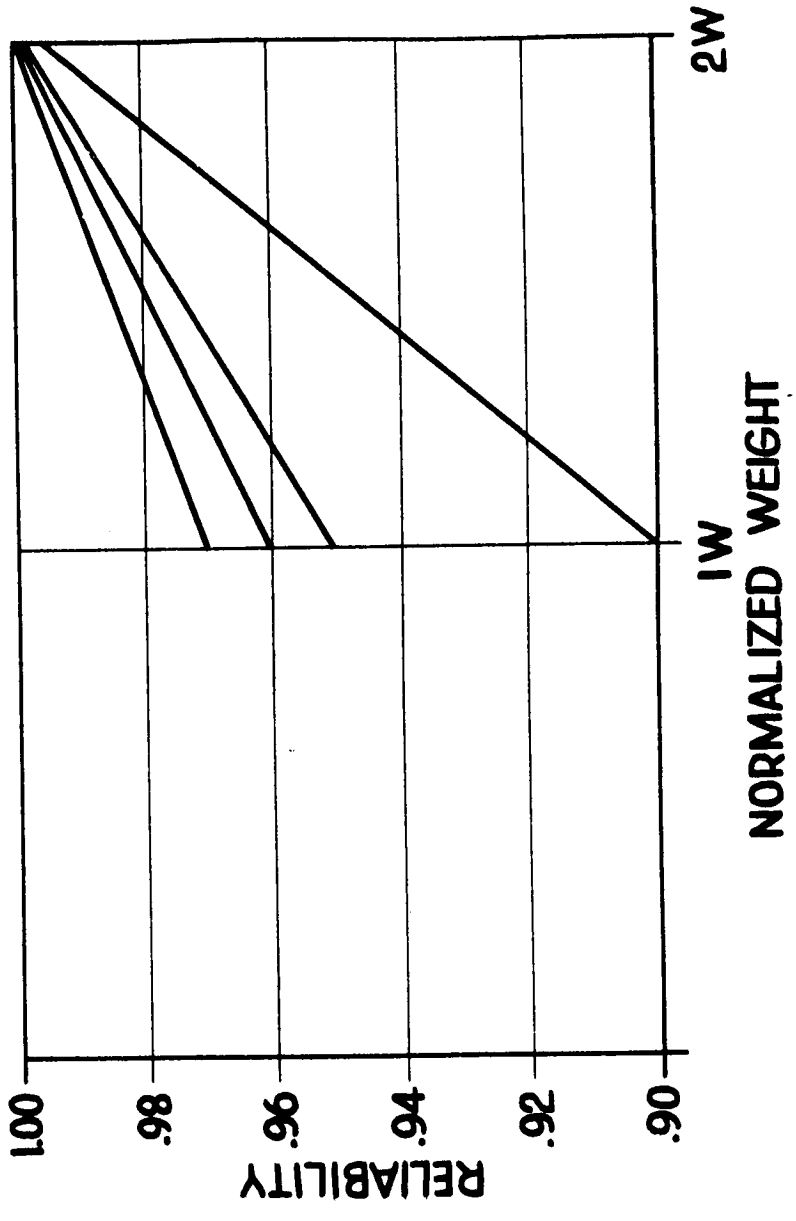


FIGURE G-3 RELIABILITY

$$R = p^{N-n} \left[1 - (q^2)\right]^n$$

where p and q are defined above and p is still given by

$$p = p^{1/N}$$

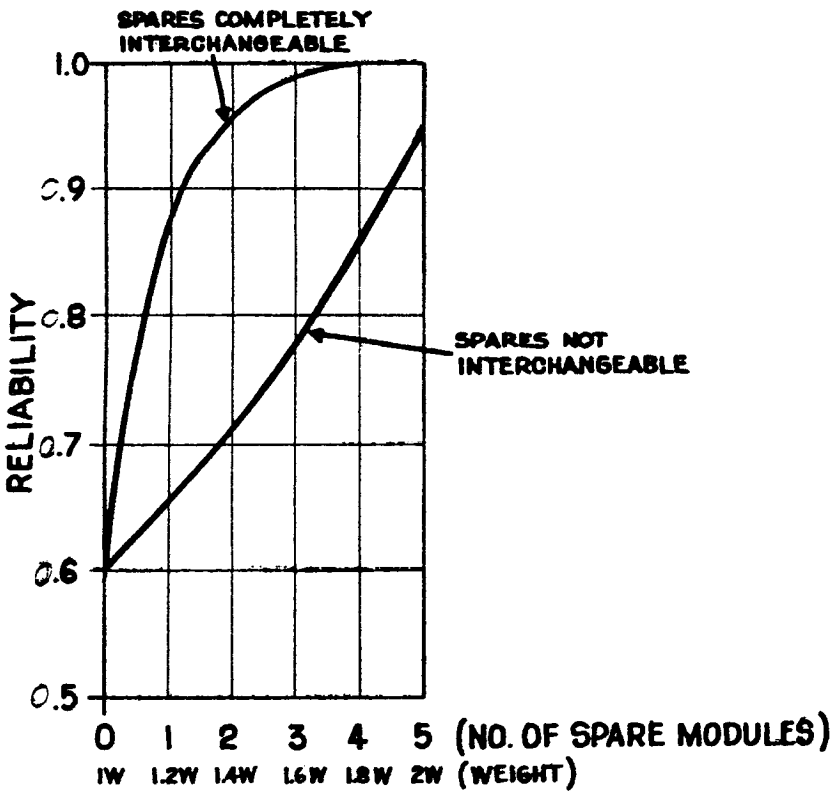
The advantages of using identical modules wherever possible are shown very dramatically in figure G-4a and G-4b. For a block of five identical modules with a basic reliability of 0.9, a redundancy of 20 percent gives a reliability of 0.994, and a redundancy of 40 percent yields a reliability of 0.9997. Even when the basic reliability is 0.59, less than 100-percent redundancy brings the final reliability very close to unity. It is thus apparent that the use of identical circuits whenever possible allows a minimum number of spares and thus minimizes weight and space.

Consider now the case of multiple redundancy with noninterchangeable items. The reliability of a block consisting of N items is given by

$$R = \prod_{i=1}^{N} P_i \tag{9}$$

and the weight of the block by

$$W = \sum_{i=1}^{N} w_i \tag{10}$$



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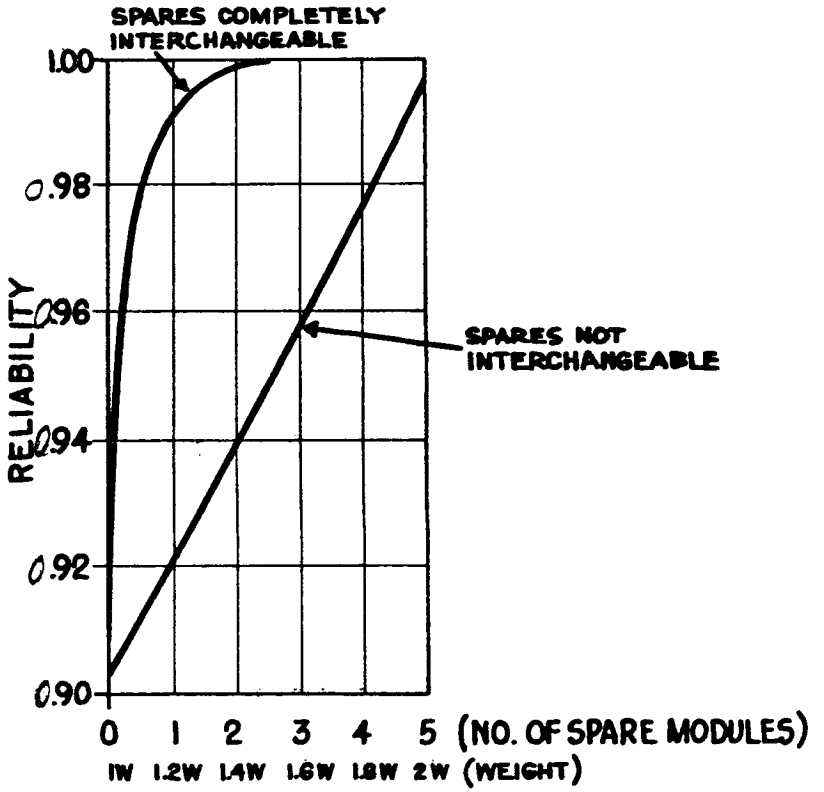


B

FIGURE G-4 RELIABILITY-SPARES INTERCHANGEABILITY

where p_i is the reliability of the ith item and w_i the weight of the ith item. If there is a spare for n of the items, then the total reliability is

$$R_T = \prod_{i=1}^{N-n} P_i \prod_{j=1}^{n} \left[1 - (1-P_j)^2\right] \qquad (11)$$

and the weight is

$$W_T = \sum_{i=1}^{N-n} w_i \sum_{j=1}^{n} 2w_j \qquad (12)$$

where p_i is the reliability of the ith item which has no spare, p_j is the reliability of the jth item which has a spare, and w_i and w_j are the corresponding weights.

If each of the n redundant items has L spares, then equation (11) would become

$$R_T = \prod_{i=1}^{N-n} P_i \prod_{j=1}^{n} \left[1 - (1 - P_j)\, L + 1\right] \qquad (13)$$

and

$$W_T = \sum_{i=1}^{N-n} w_i + \sum_{j=1}^{n} (L + 1)\, w_j \qquad (14)$$

Since $p_j < 1$, it is obvious that as L increases $(1 - P_j)^{L+1} \rightarrow 0$ and equation (13) would become, in the limit,

$$R_T = \prod_{i=1}^{N-n} p_i \qquad (15)$$

but (14) remains

$$W_T = \sum_{i=1}^{N-n} w_i + \sum_{j=1}^{n} (L+1)\, w_j \qquad (16)$$

Although R_T approaches a limit as L increases, W_T does not but rather increases without bound.

For convenience, consider the case where each item has the same reliability and the same weight:

$$P_i = P_j = P \quad i = 1, \ldots, N-n \quad j = 1, \ldots, n$$

and

$$w_i = w_j = w \quad i = 1, \ldots, N-n \quad j = 1, \ldots, n$$

then equation (13) is

$$R_T = p^{N-n} \left[1 - (1 - p)^{L+1}\right]^n \qquad (17)$$

and equation (14) is

$$W_T = (N-n)\, w + n\, (L + 1)\, w$$

$$W_T = (N + nL)\, w \qquad (18)$$

Let p = 0.974 be the reliability and W_o the weight of each of four items in a block. Then without redundancy

$$R_T = P^4 = 0.9$$

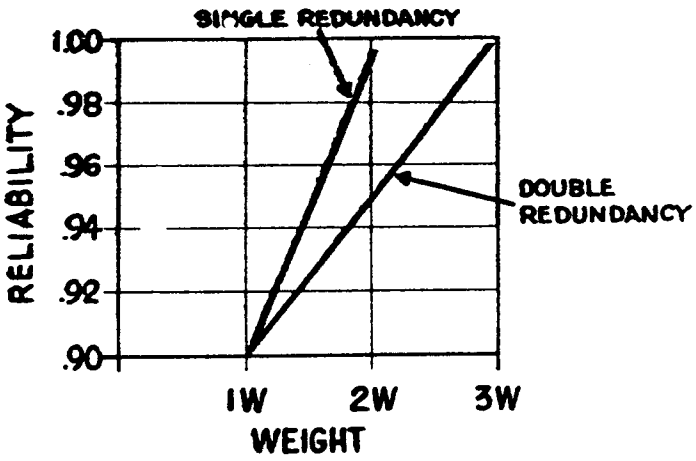
$$W_T = 4W_o$$

The increase in R_T and W_T for 0, 25, 50, 75, and 100-percent single and double redundancy is shown in figure G-5 and figures G-6a and G-6b.

Figure G-5

Single Redundancy			Double Redundancy		
% Redundance	R_T	W_T/W_o	% Redundance	R_T	W_T/W_o
0	0.9	4	0	0.9	4
25	0.923	5	25	0.924	6
50	0.947	6	50	0.949	8
75	0.972	7	75	0.974	10
100	0.997	8	100	0.999	12

Figure G-5 shows that the greatest reliability is afforded by 100-percent redundancy. There is a slight increase in reliability by offering double redundancy rather than single redundancy. The initial weight is doubled by 100-percent single redundancy and tripled by 100-percent double redundancy. Obviously, the number of spares for each item is limited by weight and volume considerations, so the maximum reliability for the minimum weight is desired.



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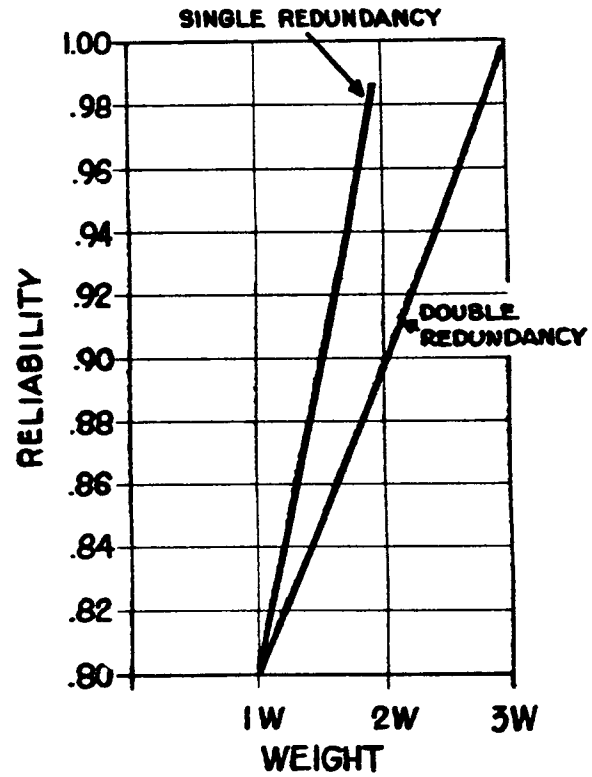

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FIGURE 8-6 RELIABILITY REDUNDANCY

Redundancy can be specified in two ways. The equipment may be designed in such a manner that circuit logic locates and replaces the failed unit with its spare. On the other hand, the failure may be detected by the equipment through self tests and the defective module removed and replaced by man.

It is suggested that approximately 15-percent redundancy be built into the critical circuits themselves and approximately 100-percent redundancy be carried out by modular replacement.

Figure G-7 shows the basic reliability blocks for the telemetry-communications system. The improvement of reliability with man in the loop is based upon 100-percent modular replacement with the defective modules being repaired where possible during noncritical periods.

d. Reliability Program

In order to achieve the required reliability for the telemetry equipment, a program encompassing six basic points is required. These points are delineated below in anticipated chronological order.

(1) A reliability specification for all equipment will be defined.

(2) Reliability units (POMS) will be allocated for the conceptual hardware and for man (see paragraph c.3b.

(3) The reliability unit allocations will be reviewed with considerations of redundancy and preventive maintenance.

Figure G-7 - Communications System

Phase	Signal Conditioning	Decoder	Receivers	Trans-mitters	Storage	Antenna	Total Reliability	Critical Period
2	0.9896 0.9999	0.9886 0.9999	0.9886 0.9999	0.9886 0.9999	0.9896 0.9999	0.9935 0.9999	0.9400 0.9998	12 hours
3	0.9913 0.9999	0.9904 0.9999	0.9904 0.9999	0.9904 0.9999	0.9904 0.9999	0.9953 0.9999	0.9500 0.9998	8 hours
4	0.9931 0.9999	0.9921 0.9999	0.9921 0.9999	0.9921 0.9999	0.9931 0.9999	0.9970 0.9999	0.9600 0.9999	6 hours
5	0.9949 0.9999	0.9939 0.9999	0.9939 0.9999	0.9939 0.9999	0.9949 0.9999	0.9987 0.9999	0.9700 0.9999	2 hours

Phase 4

X.XX basic equipment reliability

X.XX reliability with 100 percent redundancy

(4) Assistance will be given during the process of equipment design with respect to the points outlined in(2) and (3). In addition, fabrication controls that will guarantee the optimum operational reliability will be established.

(5) The program and design will be reviewed and appropriate changes will be effected in the points outlined in (1), (2),(3) and (4). This effort will be a continuous activity throughout each phase of this program. Considerations will be given to the optimum compromise with respect to equipment redundance, preventive maintenance, and in-flight maintenance and repair.

The detailed program necessary to achieve the desired reliability for the checkout and telemetry equipment is delineated in the succedding paragraphs and exemplified in figure G-8.

e. Step I - Establishment of a Reliability Specification

Initially, the entire system and system requirements will be reviewed for specific reliability requirements. This information, together with the conceptual design and the expected environmental levels, will be utilized to generate a reliability specification. This specification will be different for each phase with revisions reflecting the most recent definition of environmental conditions and system requirements. For Phase I, the environmental conditions will be estimated from existing data on orbital flight environments. In addition to environmental conditions, this specification will also include, but not be limited to, mean-time-between-failure requirements, safety margins, and maintenance philosophies.

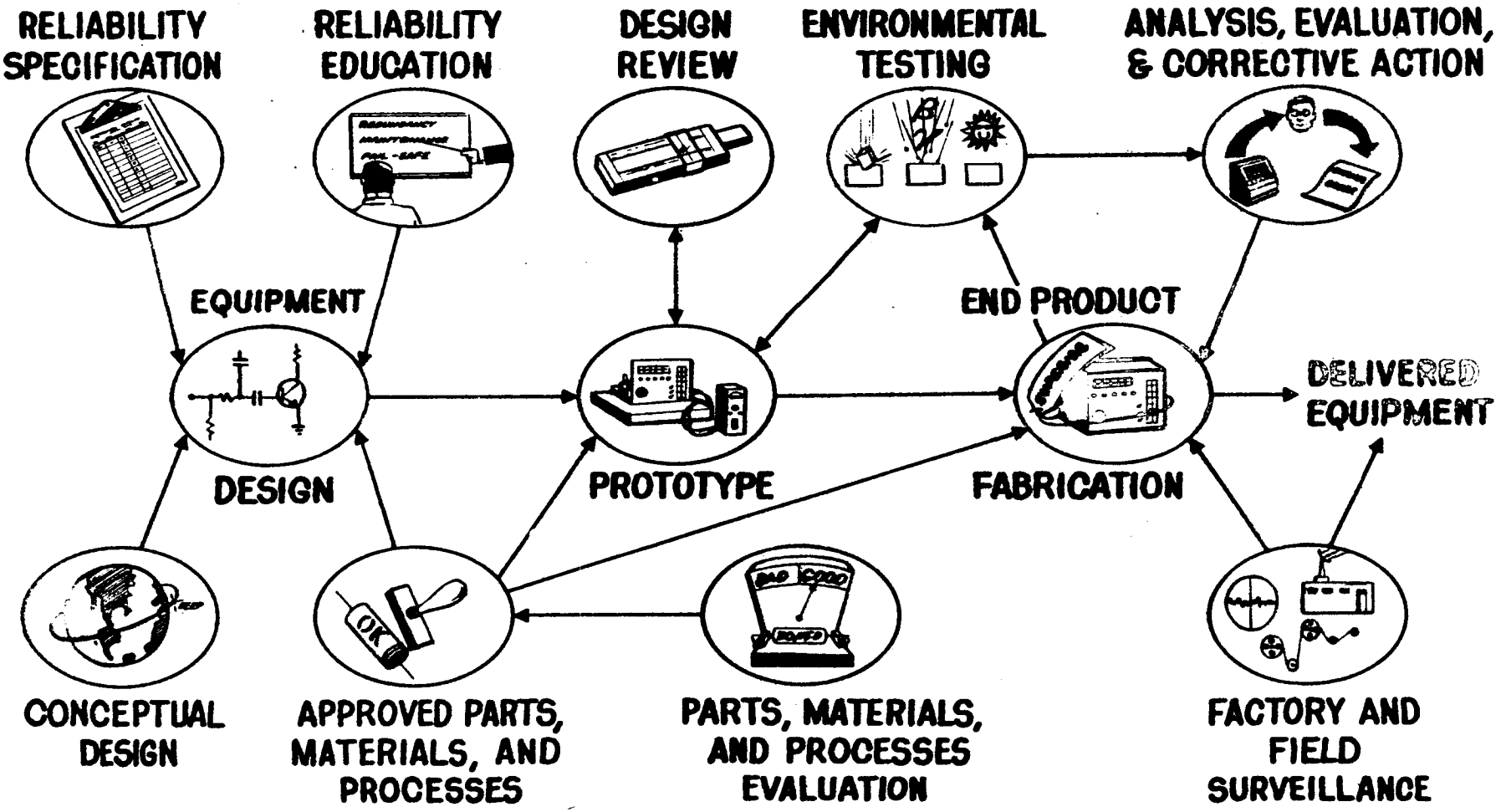


FIGURE G-8 RELIABILITY FOR CHECKOUT TELEMETRY EQUIPMENT

f. Step II - Reliability Education

It is intended to employ reliability education as a means of introducing reliability into this program at the earliest possible date. Initially, the reliability education program will be directed toward the equipment design personnel. It is felt that equipment design personnel versed in reliability techniques and appreciative of the role of reliability techniques in ensuring mission success represent the most desirable foundation for a reliability program. This philosophy coincides with the reasoning that reliability must be built into a product and cannot be tested into it. Practically, this approach will mean a maximum probability of mission success during the early program phases and a minimum amount of design changes throughout the program.

Equipment design personnel will be educated in design techniques stressing the merits of redundancy, self-sensing fail-safe circuitry, self-healing circuitry, and maintenance and considering component application data such as derating, standardization, and failure rates. In addition, reliability prediction techniques will be taught to the equipment design personnel. This phase of the education will allow the designer, to a limited degree, to statistically evaluate problem areas and effect appropriate decisions while compromising equipment redundancies and maintenance without degrading the overall system reliability. While the design personnel have this statistical capability, this does not relieve the reliability personnel from the responsibility of reliability unit allocation.

After equipment configuration has been established, the reliability education of fabrication and test personnel

will be effected. These personnel will indoctrinated into the proper use of a comprehensive failure reporting system and in any special assembly techniques, processes, and tests which are currently nonstandard.

g. Step III - Approved Components, Materials, and Processes

Components, materials, and processes representing the respective state-of-the-art will be assessed relative to applicability, reliability, and availability. To aid in this assessment, information generated from current and future defense and scientific programs will be analyzed; if information is not readily available, special exploratory tests will be performed to obtain preliminary design information. From this industry-wide assessment, a tentative list of reliability approved components, materials, and processes will be generated. Subsequently, equipment design personnel will select their basic design items from this list. Those items selected will then undergo extensive accelerated qualification testing to ascertain mode of failure, mean-time-between-failure, upper and lower bounds of reliable operation, parameter distributions as items are received from the vendor, and parameter distribution shifts with respect to time. The information gathered will be available in sufficient time for the design changes deemed necessary as a result of this program to be incorporated along with the changes deemed necessary as a result of prototype equipment test.

In the event components, materials and processes which are necessary to satisfy the pre-defined reliability requirements are not available, an accelerated developmental program to advance the state-of-the-art will be proposed. If this approach is deemed necessary, results from it will not be available for system integration until Phase II.

Once approved components, materials, and processes have been selected, it will be necessary to ensure that the respective product vendors maintain high quality standards. This will be accomplished by continual requalification testing.

Special treatment will be given to electronic component parts procured for this program. Every electronic component part will be serial numbered so that a detailed history of each part is readily available. The history will be maintained on automatic data handling cards and will include such information as date of manufacture, date of shipment, date placed in stock, date of fabrication into equipment, and date of failure if applicable, and will reflect the parametric values obtained at each point where electrical parameters are measured. These cards will be particularly useful in analysis of data from qualification and re-qualification testing; particularly, a statistical prediction of expected electrical parameter variations with respect to time will be made. Subsequently, the parameter variation versus time studies will be used to establish subassembly failure criteria with respect to operating time and tolerance buildup. In addition, all electrical components will undergo an electrical aging period prior to fabrication into the equipment. The components will be electrically operated for a predetermined period of time to permit detection of all latent manufacturing defects prior to the component's being installed in the equipment. It is necessary that component failures must be held to the absolute minimum in field usage, it is also important that rework (repair) resulting from component failure prior to equipment delivery must be minimized.

This rework, though it makes the equipment operable, increases the probability of a fabrication error being introduced into the equipment and is therefore required to be minimized. Consequently, the electrical aging process is deemed mandatory.

All components, materials, and processes which exhibit a failure at any stage in the program (this includes component manufacture) will be thoroughly analyzed to determine cause of failure. This information will be used to detect potential problem areas and to determine suitable corrective action to prevent the failure recurrence whatever the cause may be

All fabrication processes whether they be standard industrial practices or advancements in the state-of-the-art will be reviewed and evaluated for reliability adequacy with respect to this program. Those fabrication processes found superior will be selected and made mandatory for use on this program.

Finally, by utilizing a selected group of components, materials, and processes and by limiting the equipment design and build to this group, standarization of the program will be effected.

h. Step IV - Design Reviews

A design review program will be utilized to promote design excellence and will be implemented at the initial design phase (where it has the greatest effectivity) at the completion of prototype tests prior to fabrication of deliverable hardware, and to a lesser degree at any point where a design modification is effected. The design

review effort is not designed to optimize design reliability, maintainability, or standardization, for these factors will be an integral part of the equipment design as a result of the reliability education program described previously. Rather, the design review effort will be used to detect problem areas and effect suitable corrective action; such areas of activity might be where the conceptual design was based on an unknown (assumed) parameter, where an initial design requirement is no longer valid, or where the use of a component, material, or process is no longer acceptable. In addition, design review personnel will consider tolerance limits and correlate these limits with the expected component parameter variations with respect to time as established in paragraph c-3g Step III - Approval Components, Materials, and Processes.

The design review activity will be a continuous effort through all phases of this program; however, it will have its greatest contribution for Phases I, II, and III.

i. Step V - Reliability Testing

A reliability test program to verify quantitatively the ability of the system and/or systems to satisfy the number one design requirement, reliability will be conducted. To accomplish this verification, the reliability tests will be designed and conducted at the breadboard and prototype equipment stages. These tests will be effected under multiple environmental levels at the subassembly, assembly, and system level as well as operating life tests under the critical environment. Test plans will encompass Latin Square repeated-test-to-failure techniques to obtain maximum information for minimum cost. This step in the reliability

program will be emphasized during Phases II and III and de-emphasized in subsequent phases as the configuration of the equipment has been established and proven.

Test techniques and philosophies are discussed in detail in paragraph G-4 and G-5.

h. Step VI - Processes and Fabrication Controls

The success of any reliability program is reflected in the operating reliability of the delivered equipment. In order to maximize the ratio of operating reliability to inherent reliability, it is necessary to ensure high quality workmanship, thereby minimizing the equipment degradation introduced during fabrication. This will be accomplished by a detailed program designed to define and effect controls with respect to the fabrication process. Specifically, the following areas will be considered:

(1) Operator Training

As stated previously, high quality workmanship in all stages of the fabrication process is mandatory for achievement of the reliability goals. It cannot be over emphasized that quality must be established and maintained, beginning with the assembly operator or technician who must be thoroughly aware of acceptable quality standards and then be continuously critical of his work. This awareness will be achieved through a training program designed to inform the trainee of necessary fabrication techniques and acceptable fabrication principles. The training program will be culminated when it is certified the trainee is conscious of the quality level required and has developed

proficiency in performing high quality fabrication. Operator or technician self appraisal will be achieved by including in the aforementioned training program a course psychologically designed to instill pride of workmanship and a sense of responsibility through the demonstration of the role played by the system, assembly, sub-assembly, and component part and the effect any failure may have. Subsequent to this initial training and certification program, the control process will consist of operator progress monitoring, period recertification, and supplementary training as processes or fabrication standards are revised. This training activity will be conducted as a phase of Step II, Reliability Education paragraph c-3f.

(2) Operation Instructions and Test Specifications

It is envisioned that some assemblies and subassemblies will require detailed operation instructions for fabrication and test specifications for inspection reliability will review these documents for adequacy with respect to critical processes and/or controls. The "Operation Sheet" procedure as presently employed will provide a means of re-emphasizing special and unique reliability requirements; that is, reliability standards and bulletins for special fabrication processes will be referenced in the Operation Sheets. Test specifications will naturally be reviewed for measurement requirements of the critical parameters. In addition, parameter tolerances will be reviewed for consistency with the results of the serialized component parts control system (Step III) and the design review effort (Step IV).

(3) Special Processes.

In a program where extremely high levels of quality and reliability are established goals, it may be necessary to employ special processes for all or portions of the equipment. An example of such a special process is the aging of all electrical components as outlined in paragraph G-3g Step III, Approved Components, Materials, and Processes. In addition, it is feasible that the equipment design may require that a component, sub-assembly, or assembly be kept absolutely free of microscopic foreign particles; such a requirement would dictate clean-room operations and a special sequence of fabrication.

The equipment will be reviewed during the design and "breadboard" testing phases for areas that require special processes. The results of this review will be incorporated into the prototype fabrication phase. This review will continue to a lesser degree throughout all phases of this program and special processes will be initiated as they are deemed necessary.

(4) Process Controls.

To maintain a high degree of operating reliability, the finished product must conform to its design specifications and standards. This conformation can be accomplished only through rigid process control. The method of control will vary with the process requirements. Some of the process controls which will be utilized are in-process visual inspection, 100 percent critical parameter measurement, continual requalification of component parts, subassembly tests-to-failure on a sampling basis, and redundant inspection. The exact definition of the process controls required is limited at this time by the insufficient information concerning the equipment design. All in-process controls will be defined prior to

the prototype fabrication and will be continually reviewed for adequacy throughout all phases of this program.

(5) Special Handling Requirements.

Component parts, sub-assemblies, assemblies, and complete systems will require special handling to ensure that no degradation in reliability is introduced through antiquated handling techniques. All equipment will be reviewed to determine the optimum handling techniques. Physically delicate component parts and assemblies will be protected from physical damage not only during fabrication but also while they are being shipped, stored, tested, handled, and disassembled for inspection and repair. A complete definition of special handling requirements will not be available until all components have been selected, tested, and evaluated and the equipment design has been defined. Carriers, racks, holders, stands, vehicles, and rooms are some equipment handling aspects that will be considered.

(6) Factory and Field Surveillance.

To ensure that all processes and fabrication controls are rigidly adhered to and to verify that a high level of operating reliability is attained, it is necessary to establish and maintain a factory and field surveillance program. Factory surveillance will be accomplished simply by monitoring the fabrication and test processes to verify that all established reliability principles are discriminately employed. Field surveillance will be employed to verify that the equipment performs its intended function under actual environmental conditions; an observer at all ground based installations where operations are being directed will accomplish this task.

G-4. EVALUATION CRITERIA

At the component, circuit, and subassembly levels, the reliability will be predicted by determining the failure distributions with resulting confidence bounds placed upon the mean-time-to-failure (MTTF) and mean-time-between-failure (MTBF). From the distributions the required burn-in periods and estimates for wear-out time will be established. It will be necessary also to establish criteria by which tolerance limits may be set. These limits must reflect the degradation expected with time, drift in component values, changes due to severe environment, etc. Components and circuits differ in the effect they have upon the output of an assembly or subassembly. This makes it necessary to establish separate tolerance limits for each item. This requirement is further complicated because of the possible interaction of correlation between contributors.

Suppose the output X_1 of a circuit or subassembly is a function of several items X_2 ..... X_n, where the X's may or may not be correlated. The fundamental task is to obtain the optimum weighting to be assigned to the variables X_2 ..... X_n in predicting or explaining variation in the output X_1. In the majority of cases, the problem can be linearized. That is, values for B_1, B_2, B_3 ..... B_n can be selected in equation (19) to give the best estimate of X_1.

$$X_1 = B_1 + \sum_{i=2}^{n} B_i X_i \qquad (19)$$

where

$\widehat{X_1}$ = best estimate of X_1

The coefficients B_i in equation (19) usually involve different units of measurement. To avoid this difficulty the X's are converted to a common standard by the transformation

$$z_j = \frac{X_j - \bar{X}_j}{\sigma_j}$$

where

$\bar{X}_j$ = mean of the distribution of the X_j's

σ_j = standard deviation of the X_j's

N = no. of X's for any X_j

Equation (19) then becomes

$$z_1 = \sum_{i=z}^{n} \beta_i z_i \qquad (20)$$

where, under this transformation, $B_1 = 0$

The function $\frac{\sum(Z_1 - \hat{Z}_1)^2}{N}$ is to be minimized by the proper choice of the B's. Taking the partial derivative of this junction with respect to the B's and solving the resulting set of equations for the B's give

$$\beta_j = (-1)^j \frac{D_{1j}}{D_{11}} \qquad (21)$$

where

$$D = \begin{vmatrix} 1 & r_{12} & r_{13} & \cdots & r_{1n} \\ r_{12} & 1 & r_{23} & & r_{2n} \\ r_{13} & r_{23} & 1 & \cdots & \\ \cdot & & & & \\ \cdot & & & & \\ \cdot & & & & \\ \cdot & & & & \\ r_{1n} & r_{2n} & & & 1 \end{vmatrix}$$

D_{11} = minor which remains when the first row and first column are deleted.

D_{1j} = minor which remains when the first row and the jth column are deleted.

r_{ij} = correlation coefficient for z_i, z_j

The B coefficient in equation (19) cannot be used to indicate the relative importance of the variables $X_2 \ldots X_{n1}$ as contributors to the variation of X_1. It can be shown however, that the

$$\text{variance of } \hat{z}_1 = \beta_2^2 + \beta_3^2 + \ldots \beta_n^2 + 2 \sum_{\substack{i=2 \\ j=2 \\ i \neq j}}^{n} \beta_i \beta_j r_{ij}$$

The importance of the variable $X_1 \ldots X_n$ can be judged by the magnitude of the squares of the β coefficients and the correlation between the variables; hence the tolerance limits can also be established upon the same basis. The correlation coefficients r_{ij} in equation (21) will be established by changing the X's in the breadboard designs by appropriate sampling techniques.

At the assembly level the reliability predictions will be based upon the following Latin Square design. The size of the square will vary with the assembly being tested, ranging from a 3x3 to a 5x5 design. This prediction technique has been successfully demonstrated in present test-to-failure-programs.

Consider a sample of m pieces of equipment (m items) available for testing. Let each item be tested at each of m different stress levels. At each stress level, the item is repaired and retested until r failures have occurred, thus making a total of rm failures for each item. Let j = the number assigned to an item and let S_L be the Lth stress level, where lower subscripts represent less severe levels. Let the order to which the stress levels are applied to the jth item be according to the jth row of an m x m Latin Square. To be specific, let m = 3 and consider the arrangement given in figure G-9.

Figure G-9
Latin Square - Sample of Three Items

Item No.	Order of Stress Application		
	1	2	3
1	S_1	S_2	S_3
2	S_3	S_1	S_2
3	S_2	S_3	S_1

Item No. 3 is subjected to stress level No. 2 (S_2) until r failures are observed, then to stress level No. 3 (S_3) for r failures, and finally to level No. 1 for r S_1 failures, the item being repaired and returned to service after each failure. The number of failures which can be sustained by a single item will depend both on the severity of the environment and the type of equipment being tested.

The equipment has a different mean life, θ_L^K, for each stress level, S_L, at which it is tested. Let

θ_{jLq}^K = the estimate of θ_L^K for assembly j, where the subscript q indicates the order in which the Lth stress is applied to the item.

An estimate for the mean life can be written

$$\overset{K}{\underset{jLq}{}} = \frac{T_{jq}^K - T_{jq-1}^K}{2} \qquad (22)$$

where T_{jq} = the total time for which the jth has been stressed at the qth stress level applied, and at all previous stress levels.

If the mean life is estimated correctly for each stress level (i.e., if K is chosen properly), then the estimated mean life for a given stress level will be approximately the same whether or not the item had been previously tested. If K is too large, θ_L will be overestimated; if K is too small, θ_L will be underestimated. Where age or wear-out

effects have been successfully compensated for, there will be no significant differences between the average values for each of the columns in figure G-10.

The proper value of K can be determined by an analysis of variance, figure G-10, in which the natural logarithm of the mean life is used in the Latin Square.

Figure G-10
Analysis of Variance

Source of Variation	Degrees of Freedom	Statistical Test
*S.S. due to age effects = SS_A	2	$\frac{SS_A}{SS_E}$ F (2, 2)
S.S. due to stress effects = SS_S	2	$\frac{SS_A}{SS_E}$ F (2, 2)
S.S. due to difference between items = SS_D	2	$\frac{SS_D}{SS_E}$ F (2, 2)
Error SS = SS_E	2	
Total SS = SS_T		

*S.S. = Sum of squares

The mean life can be approximated by

$$\theta_L^K = (as + b) \qquad (23)$$

where s = stress level

a, b = constants

Independent estimates for a and b can be obtained by standard linear regression theory or by expanding equation (22) in a Taylor's series about estimates for a and b. These estimates are corrected by the resulting values and the expansion repeated. Iteration, continued until the process converges, results in the final estimates.

Once $\hat{a}$ and $\hat{b}$ are determined the reliability is given by

$$\text{reliability} = \epsilon^{-T^{K-(\hat{a}s+\hat{b})}} \quad \text{for } T > 0 \qquad (24)$$

The average strength is defined as that stress level, s, for which the reliability in equation (24) is 0.5, and is given implicitly by

$$0.5 = \epsilon^{-T^{K-(\hat{a}s+\hat{b})}} \qquad (25)$$

The reliability boundary, B, is the extreme stress level sustained by the equipment under service conditions. The safety margin, defined as the number of standard deviations which separate the average strength from the reliability boundary, is

$$\text{safety margin} = \frac{K \, \text{LnT} - \text{Ln}\,(-\text{Ln}\ 0.5) - \hat{b}}{\hat{a}} \Big/ \frac{\sqrt{3}\,\sigma}{\hat{a}} \qquad (26)$$

where σ^2 = residual variance arising from the estimates, $\hat{a}$ and $\hat{b}$.

The maximum safe operating level (MSOL), defined as that stress level for which the reliability of an item tested for a period of time, T, is say 0.95, is given implicitly by

$$0.95 = \epsilon^{-T^K \epsilon^{-(\hat{a}s + \hat{b})}} \qquad (27)$$

The 95 percent lower confidence bound for the MSOL is given by

$$T = \left\{ (-\text{Ln}\ 0.95)\ \epsilon^{\left[\hat{a}\,s + \hat{b} - t_{0.95}\,h\,(s)\right]} \right\}^{1/K} \qquad (28)$$

where

$t_{0.95}$ = value in the "t" tables at the 0.95 level with one degree of freedom

$$h\,(s) = \left[\sigma^2 \,(C_o + C1\ S_L + C_2\ S_L^2) \right]^{1/2} \qquad (29)$$

where

$$C_o = \frac{\sum_{L=1}^{3} S_L^2}{3 \sum_{L=1}^{3} (S_L - \bar{S})^2}$$

$$C_1 = \frac{-2\bar{S}}{\sum_{L=1}^{3} (S_L - \bar{S})^2}$$

$$C_2 = \sum_{L=1}^{3} (S_L - \bar{S})^2$$

If the stresses during flight, ψ, are expressed as a function of time, then reliability (flight) =

$$\int_{t=0}^{T} e^{-(\hat{a}\psi + \hat{b})} kt^{k-1} dt \qquad (30)$$

where T = time of flight.

Estimates for the stresses, ψ, will be determined by preliminary studies, tests and telemetry data. With present information it is felt that the temperature, vibration, and radiation environments can be treated independently in the Latin Square analysis. The combination of the results of all these environments will then constitute a basis for the flight reliability of the checkout and telemetry systems.

G-5. PROBLEM AREAS

a. Man's Maintenance Capability

One problem area, the extent of which is still undefined, is man's effectivity in performing maintenance activity in a space environment. Man's capabilities and limitations in a space environment will only be defined after the completion of a space mission. There are, however, certain basic assumptions that can be made at the present time. The assumptions are delineated below:

(1) Man will be able to perform certain maintenance and operating functions on the OLF and S/C.

(2) In the absence of a gravitational environment and mechanical restraints, the average maximum continuous hand force which a man can exert is equivalent to a torque of 2.5 foot-pounds. With mechanical restraints this capability is considerably improved.

(3) Man's (unit)proficiency index will improve with time and experience in a fashion similar to that depicted graphically in figure G-11.

(4) Man's ability to perform maintenance while being a member of a three man crew will not be decreased by the loss of one crew member.

FIGURE G-11 UNIT PROFICIENCY GROWTH

The proposed approach to optimizing reliability is to have a built-in circuit redundancy of approximately 15 percent (see figure G-3) and 100 percent spare modules. The proposed approach to compromise for a loss in man's effective maintenance is to have small, light-weight, modular replacement units, and the capability of a shift upwards in the built-in circuit redundancy requirements. Thus, the unanswered question of the degree of man's effectivity in space imposes an additional design requirement: namely, the basic design of the telemetry and communications equipment must be flexible to the degree that built-in redundancy can be increased if required. Consequently, it is proposed that a continuous study program for evaluating and optimizing man's effectivity in space be included as a portion of the reliability effort.

b. Long-Term Component Stability. For many of the components which are to be used in this operation, long-term stability characteristics as functions of environment are undefined. In cases where long-term component stability information is presently available, it is felt this information is not compatible with the anticipated environments for the OLF and S/C. Since many of the environmental factors contributing to long-term component stability requirements are undefined for the OLO mission, it is proposed that a study program on this problem be effected. This study program would define critical component stability, and in conjunction with Step III, Approved Components, Materials, and Processes, paragraph C-3g, approved components will be evaluated for stability characteristics. In addition, this study program will investigate accelerated methods of duplicating long-term environmental

exposure. The end result of this program will be revised maintenance techniques designed to replace equipment before degradation failures are experienced.

c. Optimizing Reliability Techniques

A study program to develop and optimize reliability test techniques must be initiated at the inception and continued throughout all phases of this program. Some predictable factors necessitating this study program are design changes in the equipment, improvements in the state-of-the-art in environmental testing, and continual experience gained in testing. In particular, some of the study areas to be considered are

(1) Formulation of statistical models for all equipment operating modes with emphasis on equipment degradation. These models will consider man as an integral part of the equipment.

(2) Establishment of a comprehensive equipment analysis and tolerance study (Step IV, Design Reviews, paragraph C-3h) to optimize reliability unit allocation as the design activity progresses.

(3) Revisions of the data acquisition system to permit effective use of computer-program failure and test information on such a time scale to allow resultant corrective action to be integrated into the equipment within the over-all program schedule.

(4) Modifications to existing reliability test techniques to employ improvements in the state-of-the-art of environmental testing and to utilize cumulative test experience gained during this program.

G-6. CONCLUSIONS

It has been concluded that the reliability requirements for telemetry and communications can be met consistent with the program schedule. The reliability goals can be achieved without a significant state-of-the-art improvement by the use of critical circuit redundancy, one complete set of spares for maintenance, and the use of man to perform maintenance.

a. The use of man as a maintenance agent is the optimum approach for attaining high reliability. Complete redundancy (100 percent spares) is proposed because of the low weight of the conceptual design and a balance of yield with cost-weight factors.

b. Any improvement in components and circuitry techniques enhancing reliability should be utilized to effect a longer mean-time-between-failure rather than to reduce the number of spare modules.

c. The anticipated vibration and temperature environment does not present any special problems with present design techniques and components. The radiation environment and its effect on long-term component stability cannot be defined; however, it is assumed that if man can survive, the equipment will survive though its operating capability may be degraded.

d. Component development does not appear to be an area for concentrated reliability improvement; the normal component state-of-the-art improvements are commensurate with the over-all program goals.

G-7. ASSUMPTIONS

The assumptions used in the definition of the OLO reliability program are as follows:

a. The basic design of the ground, OLF, and S/C telemetry and communication equipment is compatible. Advantages resulting from this compatibility are standardization and increased confidence from a larger sample size.

b. Man will be capable of performing certain maintenance and operating functions on the S/C and OLF.

c. Man's ability to perform maintenance while being a member of a three man crew will not be decreased by the loss of one crew member.

d. For reliability test and analysis purposes, the environments are considered to be independent.

e. As a compromise between reliability and maintenance requirements, "Meca" packaging concepts and welded circuits will be used.

f. The anticipated vibration and temperature environments do not present any special problems with present design techniques and components.

g. Prototype equipment will be available for reliability testing.

G-8. RECOMMENDATIONS

It is recommended that additional study be conducted in the areas delineated and that the reliability program as outlined in paragraph G-3, Steps I through VI, be utilized to meet the reliability requirements.

APPENDIX H

CONCEPT OF MAINTENANCE AND LOGISTICS

H-1. INTRODUCTION

The present state-of-the-art of component technology and system design for reliable operation is not sufficiently advanced to field a system capable of executing the OLO mission without a system failure. Therefore, in order to achieve the desired probability of mission success, methods of supplementing the reliability effort must be examined. One logical method is provided by appropriate maintenance or repair of the equipment used in the operation. Maintenance will result in an optimum system when a trade-off between reliability of mission success and costs is considered.

Maintenance, however, implies replacement of malfunctioning items and creates a logistics or re-supply problem. Both maintenance and logistics as applied to OLO will be discussed in this Appendix.

H-2. OBJECTIVES

The objectives of this Appendix are to examine the maintenance and logistics problems associated with the OLO program. Included in the discussions will be:

a. Definitions of maintenance and logistics

b. A discussion of maintenance requirements

c. A discussion of maintenance methods

d. A discussion of design concepts for maintenance

e. A discussion of man's role in maintenance

f. A discussion of stockage and re-supply of spare parts.

H-3. APPROACH

In essence, "maintainability" denotes the amenability of a system to repair or maintenance which will ensure that a system has a high probability of successfully completing its mission. In-flight or orbital maintenance may be divided into two areas:

a. Testing and failure diagnosis

b. Corrective or other actions (includes alignment, calibration, repair, or interchange of equipment).

The denoted functions can be performed manually or with automatic test equipment, but in either case equipment, personnel, facilities, procedures, and time to accomplish the desired result are required.

Maintainability remains a basic necessity because equipment used in space missions must operate reliably with exacting precision during long-term mission under severe environmental conditions. These requirements suggest a highly integrated, complex system that must have a mean time to failure (MTF) much greater than the time required

to complete the mission. It is significant to note that maintenance must be considered as a basic design criterion from the beginning of the program. If it is considered early it will create a system which may be maintained easily and reliably. If maintenance requirements are ignored until late in the program, the penalties of cost and schedule delays may prohibit the inclusion of an optimum maintenance capability and result in a decrease in probability of mission success.

a. Maintenance Methods

(1) Maintenance Model. A model of the steps required for the performance of maintenance is illustrated in the figure H-1:

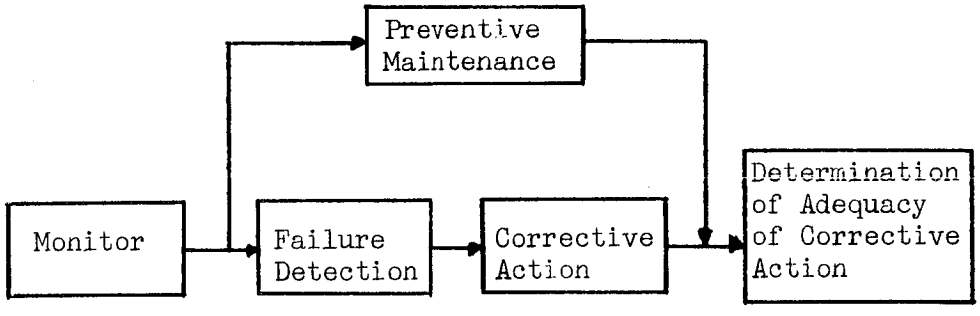


Figure H-1. Maintenance Model

The model holds true for automatic, semiautomatic, or manual maintenance. The fully automatic system would automatically test itself and then monitor the functions in the rest of the system. An alternate method would bypass the self-test feature and use a majority-of-dissenting-vote technique for failure detection. After a failure is detected, an automatic repair or self-healing technique would be used. This

will consist of switching to an alternate signal flow path or some equivalent operation. After the new flow path has been established, a verification test will be made to determine if the equipment has been restored to a normal mode of operation. Preventive maintenance may be performed by apparatus sensing the operating hours or rate of degradation and switching to an alternate path.

Manual maintenance is accomplished by reading of meters, scopes, and other test equipment by a human operator. Corrective action is taken by the operator's manually switching to alternate equipment or replacing of assemblies. The operator may also perform preventive maintenance by replacing equipment where degradation is indicated or when the MTTF of the equipment has been reached.

The semiautomatic maintenance scheme would combine elements of the automatic and manual maintenance schemes in an optimum manner: full automation is required in areas where repair time must be held to a minimum, but manual operation is used where flexibility is required and time permits. This system is the most practical of those presented and will be adopted if further study substantiates the advantages given above.

(2) Repair Techniques. Once a failure is detected, either an adjustment or a repair by a replacement module is required. The use of digital equipment will do much to decrease the number of adjustments, but some of the analog equipment may require adjustment. Due to a logistics consideration, an adjustment is preferable to a replacement and scrapping of the item and will therefore be used as a repair technique.

Replacement of a malfunctioning part may be accomplished by modular or piece part replacement. Piece part replacement is preferable from the standpoint of minimum weight and cost. However, the complexity of test equipment for isolation to piece part level will impose severe weight problems. The time required for piece part replacement and the reduced reliability as a result of semi-skilled operators in an alien environment are factors which prohibit piece part replacement. Dense packaging of electronics may also prevent reliable replacement of piece parts.

Repair will therefore be accomplished by modular replacement. However, the basic tools for piece part replacement will be furnished for use in case of extreme emergencies only. For example, if all operating equipment and spares of one type are inoperative, man may attempt to cannibalize piece parts from one unit to repair another.

Any replacement of piece parts will be done by welding techniques, and attempts will not be made to rebond components to their boards or to replace potting compounds because of problems associated with such operations in a space environment.

b. Design Concepts for Maintenance

(1) Optimum Modular Size. The reliability studies in Appendix G indicate that at least 8 to 16 replaceable assemblies are required for each subsystem to achieve the degree of reliability required for mission success. Maintenance and logistics requirements, however, dictate that a larger number of replaceable modules be used. Additional study will be required to determine the optimum number of

modules and hence the optimum size. Trade-offs in this area will be maintenance capability, logistics requirements, reliability, weight, volume, and cost.

(2) Accessibility. In order to permit repair by replacement in a minimal time, a convenient design must be established to give the operator maximum access to the equipment. This dictates that the equipment must be housed in a "shirt sleeve" environment. Figure H-2 illustrates a method of packaging for optimum modular replacement. This scheme has merit from the standpoint of packaging density, maximum accessibility, and maximum troubleshooting capability. Many other schemes are possible and must be evaluated, however.

(3) Updating of Equipment. During the evolutionary OLO program many changes will be required to portions of the equipment which will have an influence on system communications. To avoid scrapping of equipment already in orbit, a maximum flexibility is required of the equipment. A packaging technique such as shown in figure H-2 will give this flexibility. Modules and subassemblies may be replaced or interchanged to effect a new configuration. It should be noted that actual packaging densities will be greater than those illustrated in figure H-2.

(4) Levels of Maintenance. Two levels of maintenance are possible with the figure H-2 packaging. Subassembly replacement will be accomplished should a failure occur during a mission. Modular replacement will be performed during down time between missions.

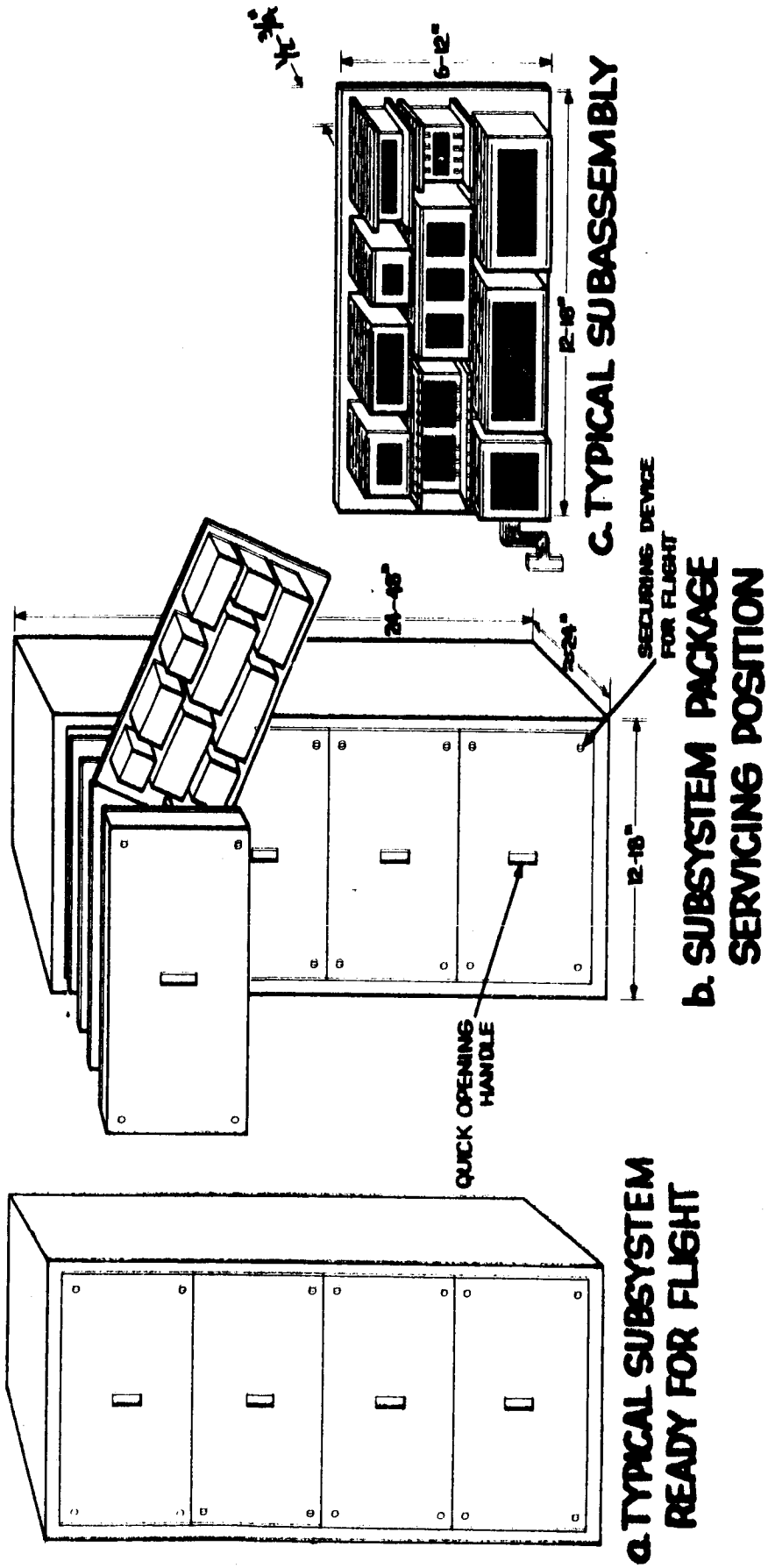


FIGURE H-2 PRELIMINARY PACKAGING CONCEPT

(5) Standardization. Due to weight limitations a concerted effort must be made to standardize in the areas of tools and test equipment. As an example, the same wrench should be used for as many applications as possible, a multimeter may serve many functions, and so on. Close coordination between all contributors to the OLO program must be maintained to effect this discipline.

(6) Maintenance Aids. Since maintenance will be required of the crew, aids must be furnished such as tee holds, handles on equipment, or a strap down seat to enable the operator to exert a sufficient torque or push to complete the maintenance job. If these are not furnished, countermotion of the man's body may prevent completion of the job.

c. Man's Role In Maintenance.

(1) General. It has been pointed out that the more sophisticated maintenance scheme that excludes man will become extremely complex and that the equipment will be prohibitively heavy. The use of man in the maintenance loop is essential, since his capabilities can be substituted for mechanical sophistication.

(2) Problem Areas. The most significant problem concerning the use of man in the maintenance loop is how efficient and stable he will remain in the space environment. Studies have been carried out to determine effects of weightlessness, absence of friction, and spatial disorientation on sensory processes and basic psycho-motor performance, manual application of forces, and handling of basic tools in a space environment. However, additional research must be completed

before man may be used to his best advantage in the OLO program. These studies indicate that man is capable of completing all basic maintenance functions.

Other studies have indicated that the most significant function of man is not necessarily manipulation of hardware, but his intellectual performance as a decision-maker. These studies also point out that human operators, if supplied with a priori reliability information, do "converge upon an optimum or ideal mathematical solution. This would indicate that the human operator trained in decision making could perform adequate maintenance." The problem then is how shall the operator be furnished with the requisite a priori information. This may be accomplished with maintenance aids and proper training.

(3) Operator Maintenance Aids. Two methods of providing the operator with the requisite information are through a microfilm library and through contact with a technical group on the earth. The microfilm projector may be used for recreational reading purposes. The microfilm library should include such items as procedures, functional block diagrams, schematics, spares, lists, and decision tables. This information will be used for unusual problems or in cases where a failure area is not isolated to a specific subassembly or module.

The ground support complex may also aid the operator by serving as a braintrust and directing the trouble-shooting activity when required.

(4) Training of Operators. To be effective the operator must know his equipment well enough to "field strip it while blind folded". To accomplish the required training an operator

must enter the project during the initial design phases and work with design engineers from the program's inception. Provisions will be made to train the operators at the factory and in the field during the development phase.

H-4. LOGISTICS

a. Definition. Logistics may be defined as the methods employed in the details of transport, storage, and supply of maintenance equipment. Logistics requirements are a direct function of maintenance requirements which in turn are dependent on failure rates.

b. Stocking of Repair Parts. A study is required to determine optimum types and levels of spares to be stocked at various points such as the OLF, launch pad, factory, and lunar base. This basic equipment reliability and repair philosophy must be considered in this study. The best present estimate of minimum spare requirements are 15 to 30 percent for the OLF, 100 percent for both the launch pad and the factory. This study must also include an investigation of storage, indexing, and inventory techniques, access requirements, etc. Storage in racks similar to the one in figure H-2 will satisfy most of the storage access problems.

c. Re-Supply of Spare Parts. A basic assumption used in the reliability studies is that an adequate supply of spares will be available when required, which in turn presents a re-supply problem. The method of replacing expended spares presents a severe economic problem and is an area which

deserves further study. One possible method of replacing critical parts on the OLF at a somewhat reduced cost is to use a smaller, unmanned missile system such as the Blue Scout for small shipments (100 to 200 pounds).

d. Disposal. Disposal of spent parts represents a problem in space. To eject them into orbit would jeopardize future space operations. To send them back to earth requires an unwarranted consumption of fuel. The most feasible solution is space garbage bags suspended outside the OLF for the period of its useful life.

H-5. EVALUATION CRITERIA

At this point consider some of the factors which have an effect on the maintainability of a system for the purpose of defining a figure of merit or predictor of maintainability. This predictor may be useful in evaluating proposed maintenance systems.

Factors which influence maintainability are as follows:

a. MTTF - Mean Time To Failure - A time dependent function of the reliability of the systems normalized with respect to the time required to complete mission.

b. MTTR - Mean Time to Repair - Normalized with respect to mission time allocated for repair.

c. CTR - Cost to Repair - Normalized with respect to mission cost. Cost of repair includes part and transportation costs.

d. POMS - Probability of Mission Success - The required probability for completing the mission successfully.

e. CORF - Cost of Repair Facility - Includes tools, test equipment, etc. Normalized with respect to mission cost.

f. RATRS - Ratio of Available Spares to Required Spares - One method of combining these factors to yield a figure of merit for maintainability (M) is

$$M = \frac{(MTTF)\ \ (POMS)\ \ (RATRS)}{(MTTR)\ \ (CTR)\ \ (CORF)}$$

The multiplicative mode was chosen on rational grounds because it appeared that a simple additive relationship would not give adequate weighting to any one factor when that factor approached an extreme. It was also assumed that a more complex or second order relationship was not justified.

Perhaps the two most important factors in a space mission are probability of success and cost. Figure H3a illustrates POMS versus mission cost for various values of M. These curves represent trends and, as such, were not generated with actual data.

Figure H3b illustrates the degree to which maintenance can increase the POMS with a resultant program cost savings. This savings is made possible by an initiation of maintenance when the reliability program has entered the asymptotic region. A study will be required to determine where the optimum point (1) on figure H-3b is located. This study must include results of initial reliability and cost studies.

H-6. PROBLEM AREAS

A number of problem areas requiring resolution have been discussed in previous sections. An outline of maintenance and logistics problems is given in figure H-4.

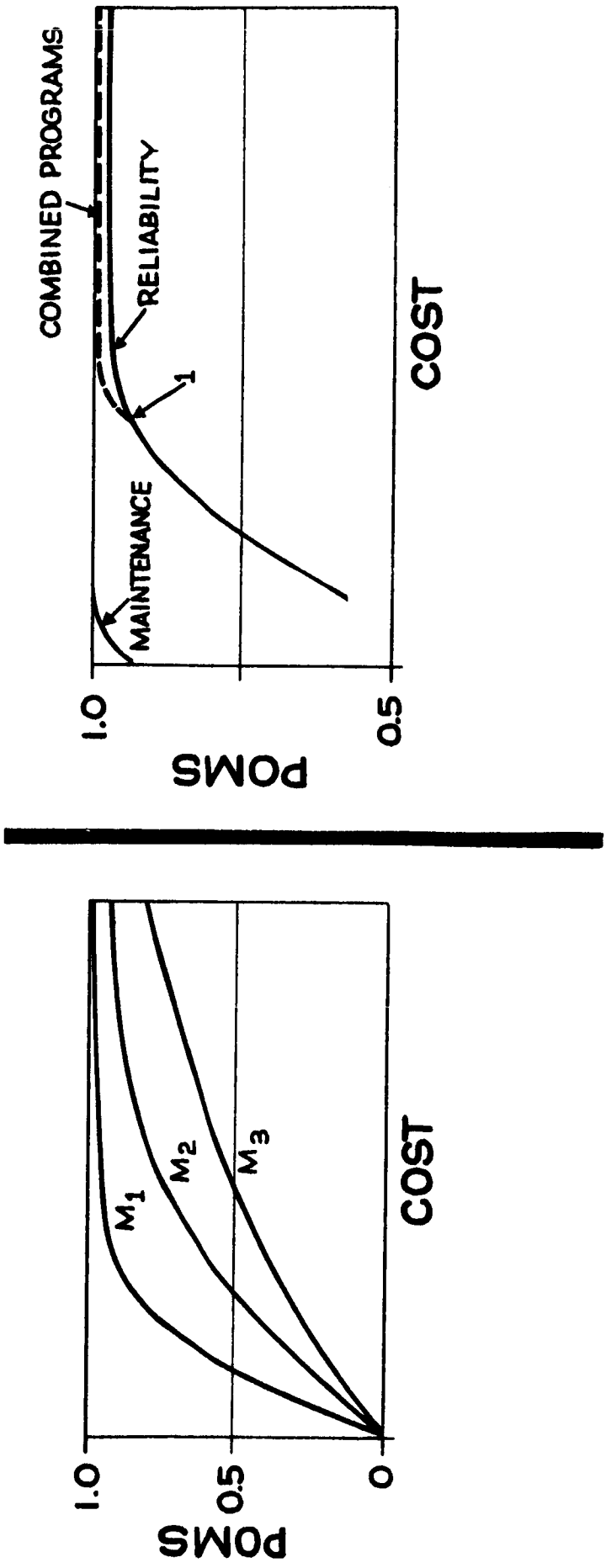


FIGURE H-3 EFFECT OF M& MAINTENANCE ON POMS & COST

FIGURE H-4

Maintenance and Logistics Problems

Problem	Phase	Statement of Problem	Proposed Solution(s)	Proposed Studies
Repair Techniques	All	Methods are required to replace malfunctioning modules, cables, etc., in a space environment.	Welding and Modular replacement.	Study is required to determine optimum techniques.
Accessibility	All	Packaging and layout of equipment must be such that access for maintenance is provided.	Include maintenance as a primary design criteria.	Determine optimum packaging and layout criteria.
Updating of Equipment	All	Provisions for updating techniques must be devised to prevent obsolescence of equipment.	Include updating provisions in original design.	Study of updating requirements and techniques.
Levels of Maintenance	All	The extent of maintenance required under varying conditions must be established. This in turn will influence test equipment requirements, tools, and workload, on the human operator.	Establish trade-offs as required.	Define required levels of maintenance.

Figure H-4 (Cont)

Problem	Phase	Statement of Problem	Proposed Solution(s)	Proposed Studies
Standardization	All	A major effort is required to standardize test equipment, tools, electronic modules, etc., to create flexibility and reliability and to reduce weight.	Set up program to define requirements and direct action in this area.	Study to determine optimum standardization program.
Maintenance Aids	All	The operator must be furnished with aids to adapt him to a space environment and enable him to perform normally. He must also be furnished with data concerning troubleshooting and maintenance.	Define and design aids which will assist the operator in maintenance. Examples are toe and hand holds, microfilm reader with microfilmed procedures, etc.	Study to determine requirements and design requisite equipment.
Operator Training	All	Extensive training of operators in the maintenance area must be accomplished.	Set up a program to train the operators.	Study and outline a training program.
Re-Supply	All	Maintenance and repair operations require the use of spares. These spares must be replenished occasionally.	Define primary and secondary modes of re-supply.	Investigate methods and cases of re-supply.

Figure H-4 (Cont)

Problem	Phase	Statement of Problem	Proposed Solution(s)	Proposed Studies
Parts Storage	All	An optimum system must be set up for storage, inventory, and indexing of repair parts.	Many varied systems may be used. A study is required to determine which is optimum.	Determine optimum system.

H-7. CONCLUSIONS

Maintenance in space will remain a necessity until the reliability of present equipment is increased by several orders of magnitude. It now appears that man will be capable of performing the required maintenance provided he is furnished with proper maintenance aids. Additional studies are required to determine man's limitations and the maintenance aids he will require. Studies are also required to determine techniques of maintenance in space.

H-8. RECOMMENDATIONS

It is recommended that man be used in the maintenance function through all phases of the OLO program. It is further recommended that a series of studies be conducted to determine man's capabilities for maintenance and which aids are required for his maintenance mission.

VOLUME V

ORBITAL LAUNCH OPERATIONS

PART II

ATTITUDE CONTROL

TABLE OF CONTENTS

Paragraph		Page
	SECTION I	
	SUMMARY	1-1
	SECTION II	
	INTRODUCTION	2-1
	SECTION III	
	INITIAL CONSIDERATIONS	3-1
3-1.	Vehicle Characteristics, Missions, and Attitude Control Requirements	3-1
3-2.	Comparison of Alternate Orbital Launch Operation Configurations	3-7
3-3.	Vehicle Attitude Control System Summary	3-11
	SECTION IV	
	RECOMMENDED SYSTEMS AND TRADE-OFF CONSIDERATIONS	4-1
4-1.	Orbital Launch Facility	4-1
4-2.	Unmanned Payload without Tug	4-13
4-3.	Orbital Launch Vehicle (No Tug)	4-22
4-4.	Manned Astrotug	4-22
4-5.	Unmanned Payload (With Tug)	4-28

TABLE OF CONTENTS (Cont)

Paragraph		Page
4-6.	Orbital Launch Vehicle (With Tug)	4-31
	SECTION V	
	COST AND SCHEDULE	5-1
5-1.	General	5-1
5-2.	Basis for Estimate	5-1
5-3.	Estimated Costs and Schedule	5-4
	SECTION VI	
6-1.	Problem Areas	6-1
6-2.	Program Recommendations	6-5
6-3.	Reliability Program	6-6

LIST OF ILLUSTRATIONS

Figure		Following Page
2-1	Alternate Configuration Matrix	2-3
3-1	Orbital Launch Facility Requirements	3-2A
3-2	Unmanned Payload Requirements - No Tug	3-2B
3-3	Astrotug Requirements for Payload Retrieval	3-4A
3-4	Unmanned Payload Requirements - With Tug	3-6A
3-5	Orbital Launch Vehicle Requirements	3-6
4-1	Stabilization and Control - Pitch Axis - Block Diagram	4-3
4-2	Stablization and Control - Roll Axis - Block Diagram	4-4B
4-3	Stabilization and Control - Yaw Axis - Block Diagram	4-6
4-4	OLF Cold-Nitrogen System	4-10A
4-5	Solenoid-Staged System	4-12
4-6	Storable Bi-Propellant System Schematic	4-12A
4-7	Storable Bi-Propellant Reaction Control System	4-12B

LIST OF ILLUSTRATIONS (Cont)

Figure		Following Page
4-8	Unmanned Payload Attitude & Vernier Velocity Control System - No Tug	4-14A
4-9	OLF Equipment for Automatic Control of Unmanned Payloads	4-14B
4-10	Driven Displacement Control System with Integrating Rate Gyro vs. Electromechanical Integrator	4-16
4-11	Schematic Arrangement of Lateral Accelerometers for Measuring Angular and Linear Accelerations	4-16B
4-12	OLV Schematic	4-22A
4-13	Minimum Attitude-Control System for Manned Astrotug	4-24
4-14	Augmented Attitude and Vernier-Velocity Control System for Manned Astrotug	4-24A
4-15	Unmanned Payload Attitude-Control System (with Astrotug)	4-28A

LIST OF TABLES

Table	Title	Page
3-1	Fuel Requirements	3-9
3-2	Orbital Launch Facility Component Weight, Volume, and Power Estimates	3-14
3-3	Unmanned Payload (No Tug) Component Weight, Volume, and Power Estimates	3-15
3-4	Manned Astrotug (Minimum System) Component Weight, Volume, and Power Estimates	3-16
3-5	Unmanned Payload (With Tug) Component Weight, Volume, and Power Estimates	3-17
3-6	Component Failure Rates	3-19
3-7	Single-Axis Attitude Control System Reliability	3-19
5-1	Summary of Attitude Control Equipment Costs for All OLO Vehicles in Thousands of Dollars	5-6

SECTION I

SUMMARY

The Sperry Rand Corporation is supporting the Vought Astronautics study of Orbital Launch Operations (Contract NAS 8-853) by investigating three Astrionics areas. The subject of this volume is Attitude Control of OLO vehicles, which covers the work accomplished by the Air Armament Division of Sperry Gyroscope Company and by Vickers, Inc., both Divisions of Sperry Rand Corporation.

The principal conclusions of this work are:

a. The attitude control aspects of the OLO program are within the time scale for state-of-the-art development as conceived by Sperry Rand. No major technological breakthroughs are necessary to successfully establish orbital launch operations.

b. Certain follow-on study work is absolutely necessary to firmly establish plans for successful orbital launch operations. In the area of attitude control these are:

(1) To establish the relative-attitude sensors and method of relative-attitude control required to operate two OLO vehicles in close proximity to one another.

(2) To prepare specifications for attitude control equipment to be used in the initial orbital-launch facility, the orbital-launch vehicle, and unmanned payloads.

Since the above work is well within the capabilities of the Air Armament Division of Sperry Gyroscope Company and

Vickers, Inc., it is requested that assignment of this work to Sperry Rand for the next funding period be considered.

c. Other recommended follow-on work is as follows. These items, while not placed in the absolutely necessary category, are still considered necessary for well-balanced planning and the proper initiation of work programs necessary to meet the objectives of the national OLO program on schedule.

(1) To proceed with further development of storable bi-propellant attitude controls.

(2) To establish the optimum attitude-control technique for use with a coupled tug/payload combination.

(3) To prepare specifications for the attitude control equipment required in the intermediate orbital-launch facility, the intermediate unmanned payload, and the astro-tug.

These programs are also recommended for Sperry Rand support of the Vought Astronautics OLO Development Program.

d. Some of the major technical conclusions, of interest to Vought Astronautics as an aid in planning the above work, are as follows:

(1) Development of special relative-attitude sensors will be required for use during docking or joining of two vehicles. Otherwise, component requirements are well within the state-of-the-art.

(2) It appears that electrical as well as mechanical interconnection of the astro-tug and the incoming payload will be advantageous from the standpoint of fuel economy and equipment weight savings.

(3) When operating unmanned payloads, it appears that the attitude command function can be performed remotely from the orbiting facility. This should allow a significant reduction in the weight of payload attitude-control equipment.

SECTION II

INTRODUCTION

This report presents the results of a preliminary study of attitude-control requirements for the vehicles to be used in Orbital Launch Operations (OLO). The study was conducted for Vought Astronautics by the Research and Advanced Development Directorate of the Air Armament Division of Sperry Gyroscope Company with the assistance of Vickers, Inc. The objectives of the study were to synthesize candidate attitude-control systems for each OLO vehicle, to isolate the basic attitude-control problems peculiar to Orbital Launch Operations, and to recommend programs that will resolve the problems and produce equipment designs suitable for the applications.

The Orbital Launch Operation will consist of placing two or more payloads in an earth orbit, combining all or part of the individual payloads into a space vehicle, and launching this vehicle from orbit. This operation can be performed in several ways. The simplest type of Orbital Launch Operation would proceed in the following manner: unmanned payloads would be boosted into orbit by large, non-manrated boosters and the crew would follow in a smaller, more reliable vehicle. A launch vehicle would be assembled from the several vehicles and injected into a lunar or deep-space trajectory. Nothing of importance would remain in orbit after the launch vehicle departed.

A more sophisticated procedure would be to orbit a large permanent facility, sending the facility crew up after

a satisfactory orbit is assured. Several payloads, both manned and unmanned, would follow and the facility crew would then assemble and/or fuel a launch vehicle. The permanent facility and its crew would remain on station for later operations. An astro-tug might be required for payload retrieval and launch vehicle orientation in the course of this orbital launch operation.

Since the type of Orbital Launch Operation to be employed has not yet been finally established, three basic OLO complexes must be considered: systems without a permanent orbiting facility, systems with a permanent facility and no astro-tug, and systems with both a permanent facility and an astro-tug. An effort was made in this study to compare the overall attitude-control requirements for these three alternate configurations, as well as to synthesize tentative attitude-control systems for the individual vehicles in each complex.

Figure 2-1 shows a matrix of the vehicles required in each of the alternate configurations. The first column of figure 2-1 summarizes the several attitude-control schemes considered for each vehicle. The control methods are manual control of remote vehicles from the permanent facility, automatic control of remote vehicles from the permanent facility, self-contained manual control of manned vehicles, and self-contained automatic control of unmanned vehicles. The applicability of each method varies, of course, with the vehicle and with the phase of operations in which the vehicle is employed.

The remaining columns of figure 2-1 list phases of the Orbital Launch Operation in which attitude control of one or more vehicles is required. These include boost to earth

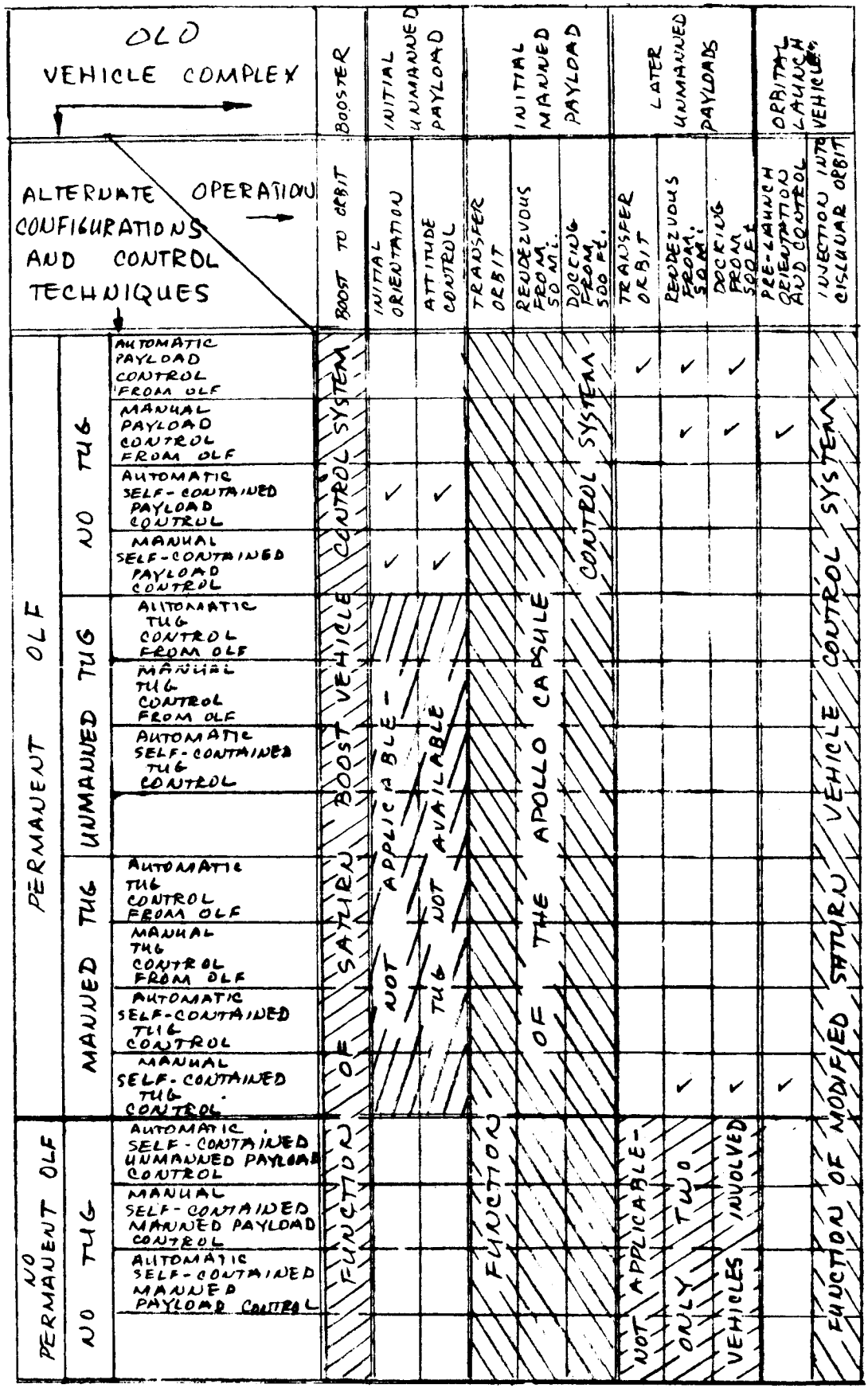


FIGURE 2-1 ALTERNATE CONFIGURATION MATRIX

orbit, rendezvous of various vehicles, and cislunar injection of a launch vehicle. As noted in the figure, boost to orbit, rendezvous by manned payloads, and cislunar injection will be performed by modified Saturn or Apollo vehicles. Attitude control systems are already available for these vehicles, and were therefore not considered in this study. For the purposes of this discussion, Orbital Launch Operations begin when the last booster stage separates from an unmanned payload and end when the main propulsion rocket on the orbital launch vehicle is ignited.

The program time schedule made investigation of all control schemes in every vehicle impractical. Accordingly, the checked combinations of Figure 2-1 were selected for detailed study. At least one example of every control scheme under consideration is included.

The areas investigated in synthesizing attitude control systems for these vehicles were:

- Manual and/or automatic attitude and vernier-velocity controls
- Stability augmentation
- Reaction-control torquers and vernier-velocity thrusters.

Guidance and navigation equipments were considered only to the extent necessary to define guidance/control-system interface requirements. Particular attention was given to the role of the human in performing attitude control, and an effort was made to establish which functions are best suited to manual control. Detailed study of such items as manual override and authority limits for automatic systems, automatic trimming of manual systems, and synchronization of

the automatic and manual systems were reserved for follow-on programs.

In performing trade-offs, reliability was considered of primary importance and, subject to performance limitations, was given highest priority. Weight was also given considerable attention, particularly when equipment was to be duplicated in every payload boosted up from earth. An attempt was made to centralize common equipments in the permanent facility whenever possible.

The following sections of this report summarize the study and its findings. Section III describes the vehicles, their missions, and their attitude control requirements. Comparisons of the permanent facility/no permanent facility and tug/no tug configurations are drawn directly from this basic information. Section IV discusses the control systems recommended for each vehicle, together with alternate systems, and trade-off considerations. Section V summarizes the program cast and schedule through 1966, while Section VI is devoted to detailed discussion of the program proposed for calendar year 1962. Finally, Section VII summarizes the recommended action to be taken in developing each of the vehicle attitude-control systems.

SECTION III

INITIAL CONSIDERATIONS

3-1. VEHICLE CHARACTERISTICS, MISSIONS, AND ATTITUDE CONTROL REQUIREMENTS

The vehicles described below are a composite of those being considered by Vought Astronautics at the time this study was initiated. The quoted performance specifications are based on documents supplied by Vought Astronautics, extractions from the literature, and data obtained from other space programs underway at Sperry. Accordingly, the vehicle and mission characteristics, and attitude control requirements, are of a preliminary nature and will be updated as more definitive information becomes available in follow-on programs.

a. Orbital Launch Facility

A sketch of the Orbital Launch Facility (OLF) and a tabulation of its attitude control requirements are shown in figure 3-1. The vehicle consists of a hangar, living quarters, airlock with attached escape vehicle and Astrotug, and a nuclear reactor. The attitude control functions include initial alignment, coarse attitude hold, orientation for docking, and fine attitude hold.

The OLF is considered to be boosted into orbit with its longitudinal axis approximately horizontal. The first function of the attitude control system is to stop

any initial tumbling and rotate the vehicle until its longitudinal axis is vertical. Vertical orientation was selected to take advantage of the restoring torque due to the gravity gradient. The time allotted to damp a 1-rpm tumbling rate is 9 minutes if the tumbling is around the yaw axis, and 90 minutes if the tumbling appears in pitch or roll. In reality, pitch tumbling rates will be much lower than yaw tumbling rates because of the different moments of inertia.

The attitude control system must maintain the longitudinal axis of the OLF within 5° of the local vertical when the coarse-attitude-hold mode is engaged. In addition, the OLF may have to be reoriented for docking of an incoming payload, and will be required to maintain the selected docking orientation very accurately. A docking reorientation maneuver, consisting of ±180° rotation about the local vertical and up to 30° departure from the local vertical, has been specified. The 30° limit allows some simplifications in the attitude control system. A fine-attitude-hold capability of ±0.5° has been specified for use during docking operations.

The positioning-accuracy and threshold-rate specifications for this vehicle were obtained from documents supplied by Vought Astronautics. The remaining quantities could then be computed. As an aside, some adjustments are acceptable in the quoted torque specifications. Generally speaking, the maximum torque level can be varied if the torque-time product remains constant. However, the control system must always meet the threshold-impulse and minimum-pulse-width specifications.

REQUIREMENT \ FUNCTION	INITIAL ALIGNMENT		ATTITUDE HOLD		ORIENTATION FOR DOCKING		DOCKING ATTITUDE HOLD		
ATTITUDE CONTROL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	
1. POSITIONING ACCURACY	±5°	±5°	±5°	±5°	-	-	±.5	±.5	DEG
2. MAX. ACCELERATION	10^{-3}	10^{-2}	-	-	10^{-3}	10^{-2}	-	-	DEG./SEC.^{2}
3. THRESHOLD RATE	.05	.05	.05	.05	-	-	.05	.05	DEG./SEC.
4. STEADY DISTURBANCES	-	-	.045	-	-	-	5	-	FT.-LBS.
5. VARIABLE DISTURBANCES	-	-	.045	-	-	-	.045	-	FT.-LBS.
6. MAX. DEMAND, STEADY	58	52	-	-	52	52	-	-	FT.-LBS.
7. DURATION MAX. DEMAND	270	540	-	-	270	540	-	-	SEC. (REPEATED)
8. THRESHOLD IMPULSE	$3x10^3$	$3x10^2$	$3x10^3$	$3x10^2$	-	-	$3x10^3$	$3x10^2$	FT.-LB-SEC.
9. MINIMUM PULSE-WIDTH	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	SEC.
THRUST CONTROL									
10. MAX. FORE/AFT. ΔV	—		—		—		—		
11. MAX. LATERAL ΔV	—		—		—		—		
12. THRESHOLD ΔV	—		—		—		—		
13. MAX. DEMAND, STEADY	—		—		—		—		
14. DURATION MAX. DEMAND	—		—		—		—		
15. THRESHOLD IMPULSE	—		—		—		—		

INITIAL ALIGNMENT

DAMPING IRPM INITIAL TUMBLING RATE: 90 MINUTES
ROTATE 90° IN PITCH: 9 MINUTES IF NO TUMBLING

ATTITUDE HOLD

SALVE TO LOCAL VERTICAL

ORIENTATION FOR DOCKING

ROTATION OF 360° MAX AROUND LOCAL VERTICAL
±30° DISPLACEMENT FROM LOCAL VERTICAL

DOCKING ATTITUDE HOLD

MAINTAIN SELECTED ATTITUDE FOR 20 MIN. DOCKING MANEUVER

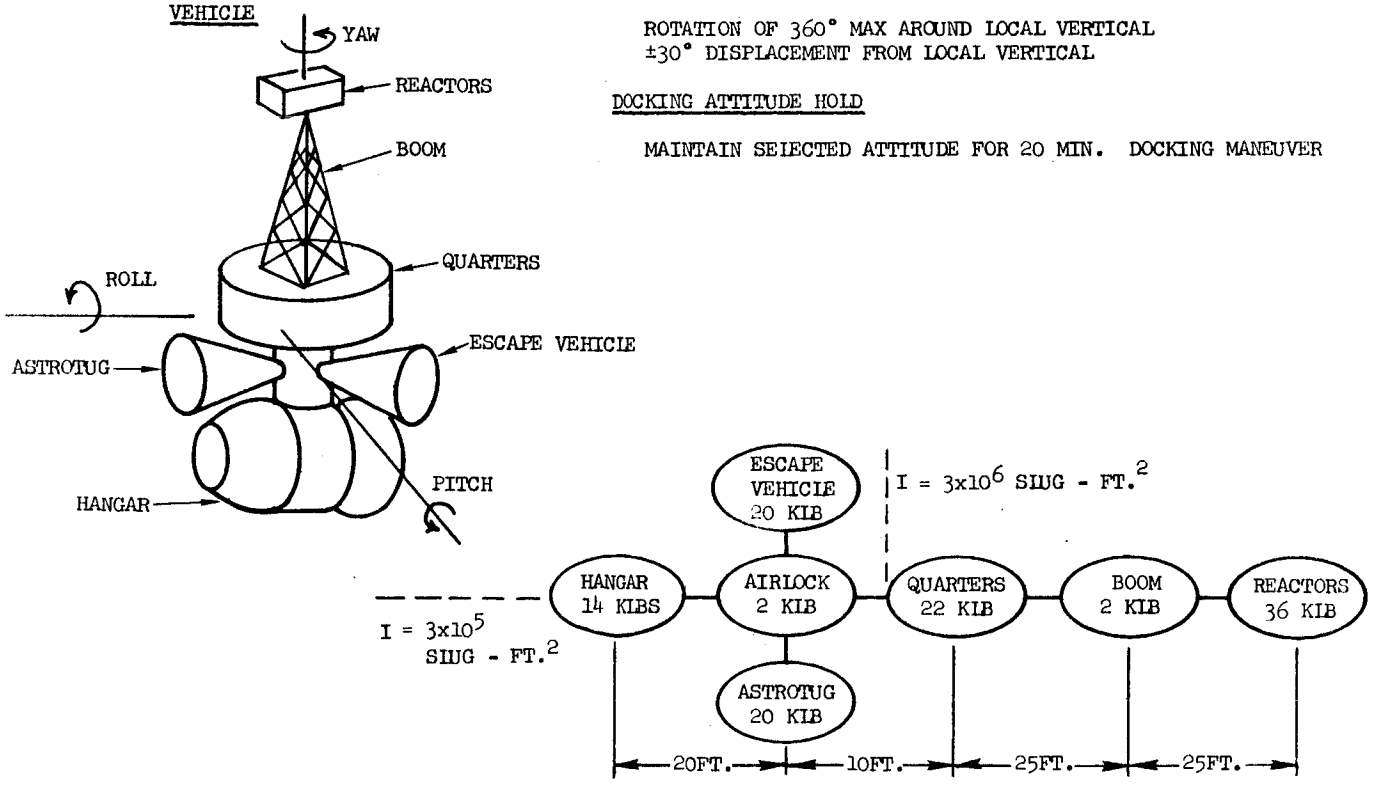


FIGURE 3-1 ORBITAL LAUNCH FACILITY REQUIREMENTS

REQUIREMENT \ FUNCTION	RENDEZVOUS		DOCKING		
	PITCH YAW	ROLL	PITCH YAW	ROLL	
ATTITUDE CONTROL					
1. POSITIONING ACCURACY	±.5°	±.5°	±.5°	±.5°	DEG.
2. MAX. ACCELERATION	.15	.30	.03	.03	DEG./SEC.^{2}
3. THRESHOLD RATE	.05	.05	.05	.05	DEG./SEC.
4. STEADY DISTURBANCES	-	-	-	-	
5. VARIABLE DISTURBANCES	-	-	-	-	FT.-LBS.
6. MAX. DEMAND, STEADY	$4x10^3$	800	800	80	FT.-LBS.
7. DURATION MAX. DEMAND	24	24	-	-	SEC. (REPEATED)
8. THRESHOLD IMPULSE	$1.3x10^3$	$1.3x10^2$	$1.3x10^3$	$1.3x10^2$	FT.-LB.-SEC.
9. MINIMUM PULSE-WIDTH	.5	.5	.5	.5	SEC.
THRUST CONTROL					
10. MAX. FORE/AFT. ΔV	900		1.0		FT./SEC.
11. MAX. LATERAL ΔV	-		1.0		FT./SEC.
12. THRESHOLD ΔV	1.0		.01		FT./SEC.
13. MAX. DEMAND, STEADY	4000		25		LBS.
14. DURATION MAX. DEMAND	560		100		SEC. (REPEATED)
15. THRESHOLD IMPULSE	$2.5x10^3$		25		LB.-SEC.

ORBIT TRANSFER

REQUIREMENTS NOT ESTABLISHED

RENDEZVOUS

INITIAL RANGE 50 MILES MAX. (NOMINAL 10 MILES AT ALTITUDE, 3 MILES LATERALLY, 3 MILES ALTITUDE ERROR)
INITIAL VELOCITY 900 FT./SEC. MAX. (NOMINAL 450 FT./SEC.)
MANEUVER TIME 80 MINUTES.

DOCKING

INITIAL RANGE 500FT.
INITIAL VELOCITY ±1 FPS.
MANEUVER TIME 20 MINUTES

VEHICLE

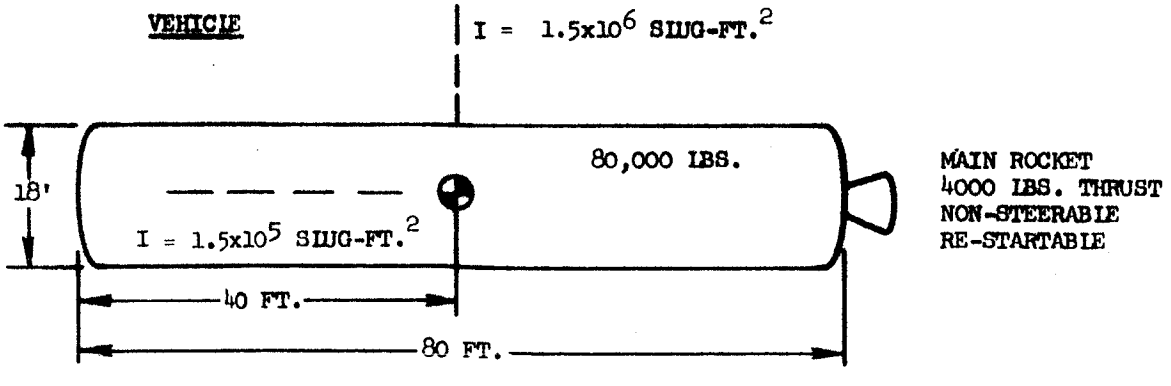


FIGURE 3-2 UNMANNED PAYLOAD REQUIREMENTS NO TUG

For example, if the torque output is M and the minimum pulse width is T_{min}, then

$MT_{min} \leq$ threshold impulse specification

$T_{min} \leq$ minimum pulse width specification

The specified threshold impulse will deliver the specified threshold rate, while the specified minimum pulse width assures a modulation frequency high enough to have a negligible effect on the dynamic response of the vehicle. Given a value of T_{min} satisfying the second inequality, the first inequality establishes an upper limit on the maximum control torque. A limited trade-off between maximum torque and duration of maximum demand is still possible. It should be noted that the maximum torque output will be required for maximum duration a number of times during the flight.

b. Unmanned Payload, with no tug

The attitude control requirements for the unmanned payload are summarized in figure 3-2 for the no-tug situation. As shown in the figure, the unmanned payload was assumed to consist of an 80,000 lb. cylinder with a non-steerable, re-startable main propulsion unit.

When the vehicle is placed in orbit it may be a considerable distance from the desired rendezvous point and transfer orbits will probably be used to reduce the initial separation. It was not possible to establish attitude control requirements for this phase of flight because neither the initial conditions nor the guidance technique have been defined.

After reaching a point within 50 miles of the permanent facility, the rendezvous flight mode is initiated. Docking maneuvers begin at a range of 500 feet. For simplicity, the astro-tug rendezvous technique was also assumed for the unmanned payload. The details of this scheme are discussed in paragraph d. below.

Most of the attitude control system requirements tabulated in figure 3-2 were obtained in a manner similar to those for the OLF. The exceptions are thrust level and angular acceleration requirements, which are the same as those specified for the astro-tug payload combination.

c. Space Vehicle, with no tug

Requirements for the space vehicle will be similar in many respects to those of the unmanned payload. Detailed discussion is reserved for paragraph e. below.

d. Astro-tug

A sketch of the astro-tug is shown in figure 3-3 along with a table of attitude control requirements for payload retrieval. The tug must rendezvous with the incoming payload, couple itself to the payload, return to the permanent facility, and dock the payload.

The rendezvous technique selected for purposes of the attitude control system investigation was the simplest possible, and is based on the results of NASA simulator studies. A collision course is established with the objective by reducing the inertial rate of rotation of the line of sight to zero. Braking thrust is then applied along the line of sight to maintain a pre-computed schedule of range rate vs range.

REQUIREMENT \ FUNCTION	PAYLOAD RENDEZVOUS		PAYLOAD COUPLING		PAYLOAD RETURN (NO HELP FROM PAYLOAD)		PAYLOAD RETURN (HELP FROM PAYLOAD)		PAYLOAD DOCKING (NO HELP FROM PAYLOAD)		PAYLOAD DOCKING (HELP FROM PAYLOAD)		
	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	
ATTITUDE CONTROL													
1. POSITIONING ACCURACY	±.5	±.5	±.5	±.5	-	-			±.5	±.5			DEG
2. MAX. ACCELERATION	2.5	4.5	.03	.03	.15	.3			.03	.03			°/SEC^2
3. THRESHOLD RATE	.05	.05	.05	.05	.05	.05	BELOW		.05	.05	BELOW		°/SEC
4. STEADY DISTURBANCES	-	-	-	-	-	-			-	-			
5. VARIABLE DISTURBANCES	-	-	-	-	-	-			-	-			
6. MAX. DEMAND, STEADY	2200	400	30	3	4000	800			800	80			FT. LBS
7. DURATION, MAX. DEMAND	6	6	-	-	24	24			-	-			SEC. (REPEATED)
8. THRESHOLD IMPULSE	45	4.5	45	4.5	1.3x10^3	1.3x10^2	NOTE		1.3x10^3	1.3x10^2	NOTE		FT-LB-SEC
9. MINIMUM PULSE-WIDTH	0.5	0.5	0.5	0.5	0.5	0.5			0.5	0.5			SEC
THRUST CONTROL													
10. MAX. FORE/AFT Δ V	900		1		900				1				FT/SEC.
11. MAX. LATERAL Δ V	-		1		-				1				FT/SEC.
12. THRESHOLD Δ V	1		.01		1		SEE		.01		SEE		FT/SEC.
13. MAX. DEMAND, STEADY	4000		60		4000				25				LBS
14. DURATION	140		10		550				100				SEC. (REPEATED)
15. THRESHOLD IMPULSE	600		6		2.5x10^3				25				LB-SEC.

PAYLOAD RENDEZVOUS

TRAVEL TO PAYLOAD; 50 MILES MAX. RANGE (NOMINAL 10 MILES AT ALTITUDE, 3 MILES LATERALLY, 3 MILES ALTITUDE ERROR)
INITIAL VELOCITY 900 FT./SEC. MAX. (450 FT./SEC. NOMINAL)
MANEUVER TIME 20 MINUTES

PAYLOAD COUPLING

INITIAL RANGE 500 FT.; INITIAL VELOCITY 1 FT./SEC.; TIME 20 MIN.

PAYLOAD RETURN

AS FOR PAYLOAD RENDEZVOUS, EXCEPT MANEUVER TIME 80 MIN.

PAYLOAD DOCKING

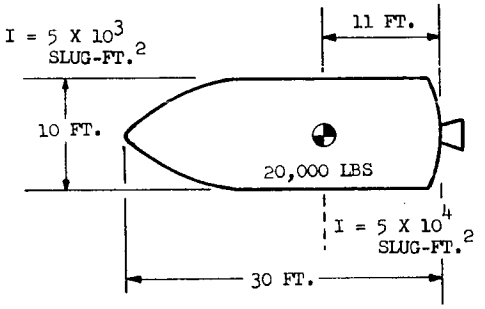
AS FOR PAYLOAD COUPLING

NOTE: MASS AND MOMENT OF INERTIA OF TUG NEGLECTED WITH RESPECT TO PAYLOAD

NOTE: PAYLOAD MAY AUGMENT TUG IN ATTITUDE AND/OR THRUST CONTROL; IF SO TUG WILL NOT PROVIDE ALL CONTROL THRUST AND/OR CONTROL TORQUES

NOTE: FOR INITIAL RENDEZVOUS WITH OLF SEE "MANNED PAYLOAD REQUIREMENTS"

NOTE: FOR OLV POSITIONING, SEE "OLV REQUIREMENTS"



MAIN ROCKET
TWO THRUST LEVELS
(4000 AND 2000 LBS.)
NON-STEERABLE
RE-STARTABLE

FIGURE 3-3 ASTROTUG REQUIREMENTS FOR PAYLOAD RETRIEVAL

The vehicle is first oriented normal to both the line of sight and the inertial angular velocity vector, after which the main propulsion unit is fired to reduce the angular velocity of the line of sight to zero. The vehicle is then re-oriented parallel to the line of sight and thrusting initiated until the range rate/range schedule is satisfied. These two operations are repeated sequentially throughout the rendezvous maneuver.

The tabulated thrust levels, angular accelerations, and rendezvous maneuver time for the tug alone were extracted from the NASA studies mentioned earlier, as were the initial ranges and relative velocities. It would appear that the initial relative velocity corresponds to an orbital plane misalignment of roughly 2°. This is quite realistic.

The maneuver time for the tug-payload combination was arbitrarily increased over that for the tug alone by an amount proportional to the payload to tug weight ratio. This was done to keep control forces at reasonable levels. Note that the total trip time for the tug is above two hours. Further time extensions might cause serious operator fatigue.

The attitude angle accuracy, velocity threshold, and angular rate threshold specified for the docking maneuver were obtained from documents supplied by Vought Astronautics. These represent the conditions required at the end of the docking maneuver. Similarly, the rendezvous phase specifications correspond to the conditions required by Vought at the initiation of the docking maneuver.

e. Unmanned Payload, with Tug

The attitude control requirements for the unmanned payload are summarized in figure 3-4 for the case when a tug is available. In this case, the payload attitude control system is required to damp the initial tumbling and, upon command from the tug, hold steady while the tug attaches itself.

Depending upon the capability of the tug, the payload may have to provide all or part of the torque and/or thrust required for the return trip and for docking. The question of whether or not to couple the payload reaction jets into the tug control system after the vehicles are joined has not been resolved.

f. Orbit-Launched Vehicle, with Tug

A possible Orbit-Launched Vehicle (OLV) configuration is shown in figure 3-5. It is constructed of a modified Saturn S-IV, two Agena B's, and an Apollo capsule. Prior to orbital launch, an attitude control system is required to stabilize the vehicle in order to avoid hitting the permanent facility, to control the vehicle while it taxis to the launch site, and to orient the vehicle for launch. As with the unmanned payload, the OLV attitude control system requirements vary according to whether or not a tug is available and if available, whether or not the OLV reaction jets are coupled into the tug control system. All three possibilities are included in figure 3-5.

Taxi to the launch site was considered to begin with an initial small velocity impulse followed by a long coasting phase. The vehicle will be re-oriented while

REQUIREMENT \ FUNCTION	INITIAL ALIGNMENT		COUPLING ATTITUDE HOLD		TUG AUGMENTATION ATTITUDE CONTROL		TUG AUGMENTATION THRUST CONTROL		
ATTITUDE CONTROL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	
1. POSITIONING ACCURACY	±5°	±5°	±.5	±.5	-	-	—	—	DEG.
2. MAX. ACCELERATION	10^{-3}	10^{-2}	-	-	.15	.3	—	—	DEG./SEC.^{2}
3. THRESHOLD RATE	.05	.05	.05	.05	.05	.05	—	—	DEG./SEC.
4. STEADY DISTURBANCES	2.5	-	2.5	-	-	-	—	—	FT.-LBS.
5. VARIABLE DISTURBANCES	.045	-	.045	-	-	-	—	—	FT.-LBS.
6. MAX. DEMAND, STEADY	26	26	-	-	4000	800	—	—	FT.-LBS.
7. DURATION MAX. DEMAND	270	540	-	-	24	24	—	—	SEC. (REPEATED)
8. THRESHOLD IMPULSE	1.3×10^3	1.3×10^2	1.3×10^3	1.3×10^2	1.3×10^3	1.3×10^2	—	—	FT.-LBS.-SEC.
9. MINIMUM PULSE-WIDTH	0.5	0.5	0.5	0.5	0.5	0.5	—	—	SEC
THRUST CONTROL									
10. MAX. FORE/AFT ΔV	—		—		—		900		FT./SEC.
11. MAX. LATERAL ΔV	—		—		—		1		FT./SEC.
12. THRESHOLD ΔV	—		—		—		.01		FT./SEC.
13. MAX. DEMAND, STEADY	—		—		—		4000		LBS.
14. DURATION MAX. DEMAND	—		—		—		550		SEC. (REPEATED)
15. THRESHOLD IMPULSE	—		—		—		25		LB.-SEC.

INITIAL ALIGNMENT

DAMPING 1 RPM INITIAL ROTATION IN ROLL
MANEUVER TIME 90 MINUTES

COUPLING ATTITUDE HOLD

HOLD SELECTED ATTITUDE FOR 20 MINUTE MANEUVER

TUG AUGMENTATION (ATTITUDE CONTROL)

ACCEPT COMMANDS FROM TUG AND PROVIDE CONTROL TORQUES

TUG AUGMENTATION (THRUST CONTROL)

ACCEPT COMMANDS FROM TUG AND PRIVIDE THRUST

NOTE:
TUG DESIGN WILL SPECIFY WHICH AUGMENTING FUNCTIONS ARE REQUIRED, IF ANY

NOTE:
SEE "UNMANNED PAYLOAD REQUIREMENTS" FOR RENDEZVOUS - DOCKING BREAKDOWN

VEHICLE:

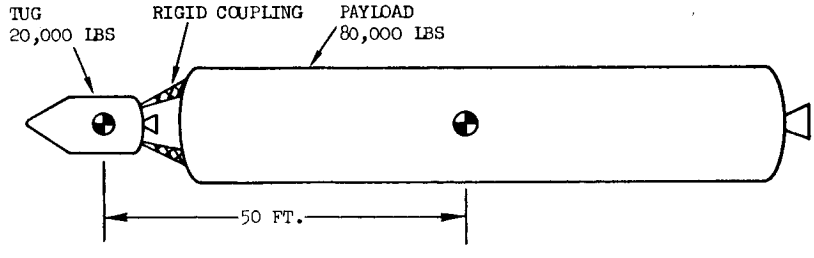


FIGURE 3-4 UNMANNED PAYLOAD REQUIREMENTS WITH TUG

REQUIREMENT \ FUNCTION	ATTITUDE HOLD NEAR OLF		TAXI TO LAUNCH SITE (NO TUG)		LAUNCH ATTITUDE AND POSITION HOLD		
ATTITUDE CONTROL	PITCH YAW	ROLL	PITCH YAW	ROLL	PITCH YAW	ROLL	
1. POSITIONING ACCURACY	±2°	±2°	±5°	±5°	±2°	±2°	DEG.
2. MAX. ACCELERATION	-	-	$2x10^{-4}$	$2x10^{-4}$	$2x10^{-4}$	$2x10^{-4}$	DEG./SEC.^{2}
3. THRESHOLD RATE	.02	.02	.05	.05	.02	.02	DEG./SEC.
4. STEADY DISTURBANCES	.33	-	-	-	.33	-	FT.-LB.
5. VARIABLE DISTURBANCES	-	-	-	-	-	-	
6. MAX. DEMAND, STEADY	-	-	20	2	20	2	FT.-LBS.
7. DURATION MAX. DEMAND	-	-	900	900	900	900	SEC. (REPEATED)
8. THRESHOLD IMPULSE	$2x10^3$	$2x10^2$	$5x10^3$	$5x10^2$	$2x10^3$	$2x10^2$	FT-LB-SEC.
9. MINIMUM PULSE-WIDTH	0.5	0.5	0.5	0.5	0.5	0.5	SEC
THRUST CONTROL							
10. MAX. FORE/AFT. ΔV	—		4.4		1.0		FT./SEC.
11. MAX. LATERAL ΔV	—		1.0		1.0		FT./SEC.
12. THRESHOLD ΔV	—		1.0		1.0		FT./SEC.
13. MAX. DEMAND, IMPULSE	—		$2x10^4$		$4.7x10^3$		LB.-SEC.
14. DURATION MAX. DEMAND	—		-		-		
15. THRESHOLD IMPULSE	—		$4.7x10^3$		$4.7x10^3$		LB.-SEC.

ATTITUDE HOLD NEAR OLF

SLAVE TO OLF ATTITUDE (LOCAL VERTICAL)

TAXI TO LAUNCH SITE

TAXI 3 MILES IN ONE HOUR; ROTATE 180° FOR RETROTHRUST

LAUNCH ATTITUDE AND POSITION HOLD

ROTATE 180° MAX. FOR LAUNCH; HOLD ATTITUDE AND POSITION AS COMMANDED FOR OLF

NOTE:

IF TUG IS USED WITHOUT TUG/OLV CONTROL SYSTEM COUPLING THE OLV HAS NO TAXI REQUIREMENTS.

IF OLV ATTITUDE CONTROL SYSTEM IS CONTROLLED FROM TUG THE ATTITUDE CONTROL FUNCTIONS LISTED ABOVE STILL APPLY BUT THE TUG WILL PROVIDE THRUST FOR TAXI.

THE TUG CANNOT ASSIST THE OLV IN LAUNCH ATTITUDE AND POSITION HOLD.

NOTE:

INJECTION, MIDCOURSE AND LUNAR ATTITUDE AND THRUST VECTOR CONTROL PROVIDED BY SATURN S-IV, AGENDA-B, AND APOLLO CONTROL SYSTEM.

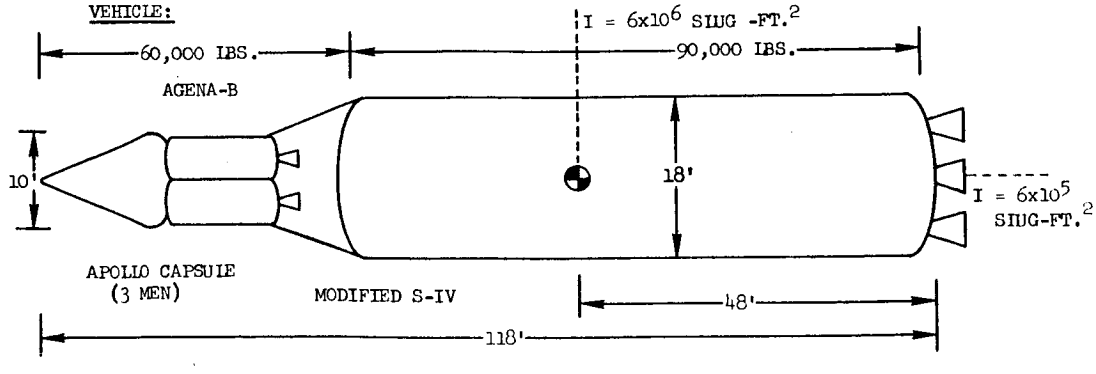


FIGURE 3-5 SPACE VEHICLE REQUIREMENTS

coasting so that a retro-thrust can be applied when the launch site is reached. The OLV will then be slewed to the proper launch direction as computed and commanded by the OLF.

It does not appear feasible to launch directly from the permanent facility because of the danger to the facility crew and because both vehicles are of roughly the same size. Any impacts may seriously perturb the OLF orbit (and structure). For similar reasons, the tug cannot assist the OLV in the final phases of pre-launch attitude hold.

The Saturn S-IV is presently expected to have a restart capability. In order to provide this capability, a coast attitude control system will be added so that adequate attitude control will be available when the main engines are off. Such a system has been sized by NASA, utilizing gaseous H_2O_2 as the propellant and swiveling reaction nozzles as torque generators. If available, this Saturn modification may well provide sufficient prelaunch attitude control and make an OLO-supplied piggyback unit unnecessary.

3-2. COMPARISON OF ALTERNATE ORBITAL LAUNCH OPERATION CONFIGURATIONS

The information assembled in the preceding section can be used to develop a comparison of the permanent-orbiting-facility/no-permanent-facility and tug/no-tug complexes mentioned earlier. The results of these comparisons, discussed further below, can be summarized very briefly:

any attitude-control equipment trade-offs seem relatively unimportant when compared to the major operational variations implied by the alternate configurations.

The major difference between the permanent-facility and no-permanent-facility configurations from the standpoint of attitude control appears to be one of equipment duplication in successive operations. If a permanent facility is available, any special equipment required for rendezvous can be reused. If no permanent facility is available, this equipment will presumably be ejected as waste before the space vehicle is injected into its cislunar trajectory. The ejected equipment will be lost and booster fuel will have to be expended in replacing it for later operations. It does appear, incidentally, that special attitude sensors will be required for docking.

Otherwise, there appears to be no major difference between the permanent-facility and no-permanent-facility configurations from the standpoint of attitude control. The former requires another vehicle and thus another attitude-control system, and some minor changes may be made in the vehicle attitude-control systems if a permanent facility is available. However, the most pressing OLO attitude-control problem, the control of the relative attitude of two vehicles in close proximity to one another, must be solved in either case.

Attitude-control requirements are also somewhat sensitive to the presence or absence of an astrotug. For example, rough calculations of the fuel required for torque and thrust, with and without a tug, lead to the tabulation of table 3-1.

TABLE 3-1

FUEL REQUIREMENTS

Phase of Payload Retrieval Operation	Thrust Requirement	Lbs fuel req'd for thrust		Attitude Control Required	Lbs Fuel req'd for torque (I_{sp} = 200)		
		Without Tug	With Tug		Without Tug	With Tug Coupled	Not coupled
1. Rendezvous with Payload	ΔV = 900FPS (I_{sp} = 300)	0	1870	3 max. demand max. duration maneuvers	0	20	20
2. Coupling to Payload	3 Max. demand, max. duration maneuvers (I_{sp} = 300)	0	15	3 max. demand, max. duration maneuvers	0	negligible	negligible
3. Payload Return	ΔV = 900FPS (I_{sp} = 300)	7,330	9200	3 max. demand, max. duration maneuvers	140	280	280
4. Payload Docking	3 max. demand, max. duration maneuvers (I_{sp} = 200)	38	45	3 max. demand, max. duration meneuvers	negligible	negligible	93
5. Totals (20,000 lb tug)	-	7368	11,130	-	140	300	393

The rather striking differences in fuel requirements are due to the following factors:

The 20,000-pound tug must be taken to the payload and returned to the facility

The tug, positioned 50 feet off the tug/payload CG, increases the vehicle mass by 25 percent and doubles the moment of inertia

If the tug cannot operate the payload reaction jets (i.e., no electrical coupling) then a lateral vernier thrust applied by the tug must be accompanied by a large compensating torque to maintain a fixed attitude during docking maneuvers.

Since the weight of the tug is not firmly established, totals have also been shown for a 10,000 pound vehicle. The use of either tug for the entire rendezvous maneuver involves a severe fuel-weight penalty. If use of the tug is restricted to the docking maneuver, the weight penalties are considerably reduced, as shown below.

Total Docking Fuel Weights

No tug:	38 pounds
10,000-pound tug:	75 pounds if coupled electrically, 230 pounds if not.
20,000-pound tug:	60 pounds if coupled electrically, 153 pounds if not.

If one assumes that the tug is a dual-purpose vehicle which would be in orbit whether used during the docking operation or not, then the use of the tug is justified if 13-27 pounds of equipment can be eliminated in the incoming payload, unless electrical interconnection of tug and payload is impractical. If the tug must operate independently, then 230-244 pounds of equipment must be eliminated to justify its use. Thus, if the tug and payload can be interconnected electrically, use of a tug appears promising. If not, the tug seems somewhat marginal.

It must be emphasized that operational advantages gained with the tug could well outweigh the weight penalties discussed above. Such factors, which are beyond the scope of an attitude-control system study, have not been discussed in this document. Finally, it should be mentioned that attitude control of all vehicles in any of the OLO configurations is well within the state-of-the-art.

3-3. VEHICLE ATTITUDE CONTROL SYSTEM SUMMARY

A summary of the detailed discussion to be given in Section IV is presented below. The major elements of the attitude-control systems recommended for the orbital launch facility, the unmanned payload, and the astrotug are presented. Tabulations of power, weight, and volume requirements are included.

The attitude-control system recommended for the orbital launch facility is quite conventional. A horizon seeker serves as the pitch-and-roll reference, slaving the vehicle to the local vertical. An integrating rate gyro is used to stabilize the vehicle in yaw. Periodic manual

realignment of the yaw axis will probably be required, although automatic alignment is feasible. A capability for manual reorientation prior to docking is provided, along with a fine attitude-hold mode for use during docking maneuvers. Torques are obtained from a cold-nitrogen reaction-jet control system. A storable bi-propellant system would offer a slight weight advantage, but is presently less reliable than the cold-nitrogen system. Table 3-2 contains summary information pertaining to the OLF attitude control system.

The attitude of the unmanned payload is controlled remotely from the Orbiting Facility. An inertial reference unit is required in the payload to derive vehicle attitude in the Facility co-ordinate system. The most significant attitude control requirement for the unmanned payload is the wide range of torques necessary to perform the requisite maneuvers. Orientation of the vehicle for thrusting during rendezvous requires large torques, while the docking procedure will require very small torques. Although a cold nitrogen system is recommended because of its higher reliability, the high torque required for rendezvous places a definite weight advantage with the storable bi-propellant system. Summary data is presented in table 3-3.

The components required for attitude control of the manual Astrotug are listed in table 3-4. The system in functionally quite similar to the unmanned payload, except that control is exercised directly rather than remotely. The recommended method of torque generation in this case is by utilization of boil-off hydrogen from the main propulsion tanks.

Attitude control requirements for the unmanned payload are listed in table 3-5 for the case when a tug is available. Comparing with table 3-3, it can be seen that the presence of a tug eliminates considerable weight from the payload itself. However, as discussed in the preceding section, the overall fuel weight requirements (tug plus payload) will be higher than for the payload alone.

TABLE 3-2

ORBITAL LAUNCH FACILITY
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

Component	Weight (lbs.)	Volume ($ft.^3$)	Power (V.A.)	Remarks
1 Integrating Rate Gyro	2	0.02	13	
1 Three Axis Rate Gyro Assembly	3	0.04	24	
1 Panel Controller	3	0.05	1	
1 Electronic Assembly				
3 Threshold Detector Amplifiers	15	0.25	50	All D.C. power required is self generated
1 Gyro Torquer Amplifier				
3 Guidance Couplers Power & Switching Components				
Cold Nitrogen Reaction Jet Control System; Nozzles, Valves, etc.	55	-	-	Alternate system using storable bipropellant; Total wt. = 69.5 lbs
Propellant Weight for 30 Days	24.5			
Totals	102.5	0.31	88	With alternate reaction control system Total wt. = 92.5 lbs
ASSOCIATED GUIDANCE EQUIPMENT				
2 Horizon Seekers				
1 Long Term Yaw Attitude Reference				
1 Telemetry Receiver				Provides capability for remote initiation of docking hold mode for initial rendezvous

TABLE 3-3

UNMANNED PAYLOAD (NO TUG)
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

Component	Weight (lbs.)	Volume ($ft.^3$)	Power (V.A.)	Remarks
3 Integrating Rate Gyros	6	.06	39	
4 Linear Accelerometers	3	.03	6	
1 Inertial Attitude Reference Unit	50	1.5	60	
1 Electronic Assembly				
3 Threshold Detector Amplifiers				
3 Gyro Torquer Amplifiers Power & Switching Components	20	.4	66	All D.C. power required is self generated
2 Electromechanical Integrators				
Cold Nitrogen Attitude and Vernier Velocity Reaction Jet Control; Nozzles, Valves, etc.	263	-	-	Alternate system using storable bi-propellant
Propellant Weight for Rendezvous & Docking	140			Total wt. = 152
Totals	479	2	171	With alternate reaction control system, Total wt. = 228
ASSOCIATED GUIDANCE EQUIPMENT				
1 Telemetry Transmitter-Receiver				
1 Radar Transponder				

TABLE 3-4
MANNED ASTROTUG (MINIMUM SYSTEM)
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

Component	Weight (lbs.)	Volume (ft.3)	Power (V.A.)	Remarks
1 3 Axis Rate Gyro Assembly	3	0.04	24	Cannot be specified until optimum division of attitude control functions between tug and payload is established. Quoted weight corresponds to electrically coupled tug-payload when tug is used for rendezvous and docking
1 Panel Controller	3	0.05	1	
1 Electronic Assembly				
3 Threshold Detector Amplifiers	10	0.20	33	
Power and Switching Components				
Hydrogen Boil-Off Attitude and Vernier Velocity Reaction Jet Controls; Nozzles, Valves, etc.	113	—	—	
Propellant Weight	160			
Totals	289	0.29	58	
ADDITIONAL EQUIPMENT REQUIRED FOR AUGMENTED SYSTEM				
1 Inertial Attitude Reference Unit	50	1.5	60	
1 Resolving Computer	10	0.20	25	
1 Electronic Assembly				
5 Electromechanical Integrators	6	0.15	25	
Additional Power and Switching Components				
4 Linear Accelerometers	3	0.03	6	
ASSOCIATED GUIDANCE EQUIPMENT				
Tracking Radar Gimbal Pickoffs				

TABLE 3-5

UNMANNED PAYLOAD (WITH TUG)
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

Component	Weight (lbs.)	Volume ($ft.^3$)	Power (V.A.)	Remarks
3 Integrating Rate Gyros	6	.06	39	
1 Electronic Assembly 3 Threshold Detector Amplifiers 3 Gyro Torquer Amplifiers	15	.25	50	
Power and Switching Components				
Attitude Reaction Jet Controls; Nozzles, Valves, etc.	263	-	-	Cannot be specified until optimum division of attitude control functions between tug and payload is established. Quoted weight corresponds to electrically coupled tug-payload when tug is used for rendezvous and docking.
Propellant Weight	140			
Total	424	.31	.89	

The realistic determination of the reliability of a complex attitude control system is an extremely difficult task. Reliability estimates were not included in the summary tables because reliability depends so heavily on factors which have not yet been specified. Examples of such factors include the packaging design details, type of circuitry employed, environment, component limitations, degree of redundancy, and maintenance considerations. However, quantitative reliability goals compatible with the present state-of-the-art have been established for the OLO attitude control equipment. These goals, the basis for their establishment, and the program required to reach them are discussed in the following paragraphs.

The first step in establishing a reliability goal is to estimate the capabilities of state-of-the-art equipment. Failure-rate data for a typical single-axis, attitude-control system is tabulated and summed in table 3-6.

The pressure regulator, valving, tank, and chamber failure rates have been extrapolated from Sperry missile experience. Applying the usual reliability formula

$$R = e^{-kt}$$

leads to the reliabilities listed in table 3-7.

TABLE 3-6

COMPONENT FAILURE RATES

Component	Failure Rate, K, failures per million hours
Threshold Detector Amplifier	127
Gyro Torquer Amplifier	36
Coupler	72
Rate Gyro	400
Integrating Rate Gyro	600
Pressure Regulator	120
Valves	50
Tanks	5
Chamber	15
Sub-total	1425
Miscellany	275
Single-Axis Attitude Control System Failure Rate	1700

TABLE 3-7

SINGLE-AXIS ATTITUDE CONTROL SYSTEM RELIABILITY

Time of mission	Probability of completing mission without a failure in one axis of attitude control system.
one day	0.96
one week	0.75
one month	0.32

Most of the vehicles will be manned so that maintenance will be possible, and the mission duration of the unmanned vehicles will be much shorter than shown above. **In any case, three important conclusions can be drawn:**

- The reliability of OLO attitude control equipment must be better than the state of the art.
- Periodic maintenance and/or replacement must be possible. This, in turn, means that modular construction must be utilized and that emergency operating modes must be provided for use during maintenance.
- Redundant circuitry must be provided to reduce the failure rate by 50 percent. This can be accomplished by providing automatic or semi-automatic replacement of some of the more critical components, such as the rate gyros.

Based on the above calculations, a failure-rate goal of less than 500 failures per million hours for each axis of the attitude-control system in a typical OLO vehicle must be obtained. This will increase reliability to the order of 0.99 for a one-day period. This goal represents a definite improvement over the state-of-the-art, yet is both realistic and attainable.

SECTION IV

RECOMMENDED SYSTEMS AND TRADE-OFF CONSIDERATIONS

4-1. ORBITAL LAUNCH FACILITY

a. Recommended System

(1) System Description

This section describes an attitude stabilization and control system for the OLF vehicle. The control system, completely contained in the OLF, starts to function at booster separation. The vehicle is aligned and controlled to the local vertical. Manual control and remote guidance input are provided to allow vehicle reorientation for rendezvous, payload docking and probe launching.

The following description is presented by control axis (pitch, roll and yaw) broken down into operating modes. The system operating modes considered are those presented in Section III.

- Initial Stabilization and Alignment
- Coarse Attitude Hold
- Docking Alignment
- Fine Attitude Hold
- Emergency Manual Control

(a) Pitch Axis (figure 3-1)

It is assumed that the OLF's pitch axis is required to be automatically aligned to the local vertical. In addition,

the system must be capable of manual or remote attitude re-orientation. Figure 4-1 is a block diagram of the basic approach to pitch-axis stabilization and control.

1. Initial Stabilization and Alignment

The OLF-Booster separation will impart torques to the OLF that produce vehicle tumbling. The tumbling rates, assumed to be in the order of one rpm, must be counteracted by the stabilization and control system to align the vehicle to the control reference. For this discussion, the control reference is assumed to be the local vertical.

The control signal for initial alignment is derived from body-mounted horizon seekers. The error signal, proportional to the error between the vehicle's pitch axis and the local vertical, is filtered and fed to the ASCS threshold detector. If the attitude error exceeds five degrees, the threshold detection fires the pitch reaction jets, torquing the vehicle towards zero attitude error. The vertical attitude error is summed with a pitch-rate gyro that provides the damping necessary to control the on-off limit cycle. The need for dual-reaction jet capability and the type of threshold detector employed (i.e. pulse-width modulator or conventional on-off) will be determined as a result of proposed trade-off studies. The alignment sequence is programmed to start after the vehicle is in orbit.

2. Coarse Attitude Hold

The system configuration for coarse attitude hold is the same as for the initial alignment. The vehicle's pitch attitude is held to within five degrees of the local vertical. The system's angular rate threshold could be set above the

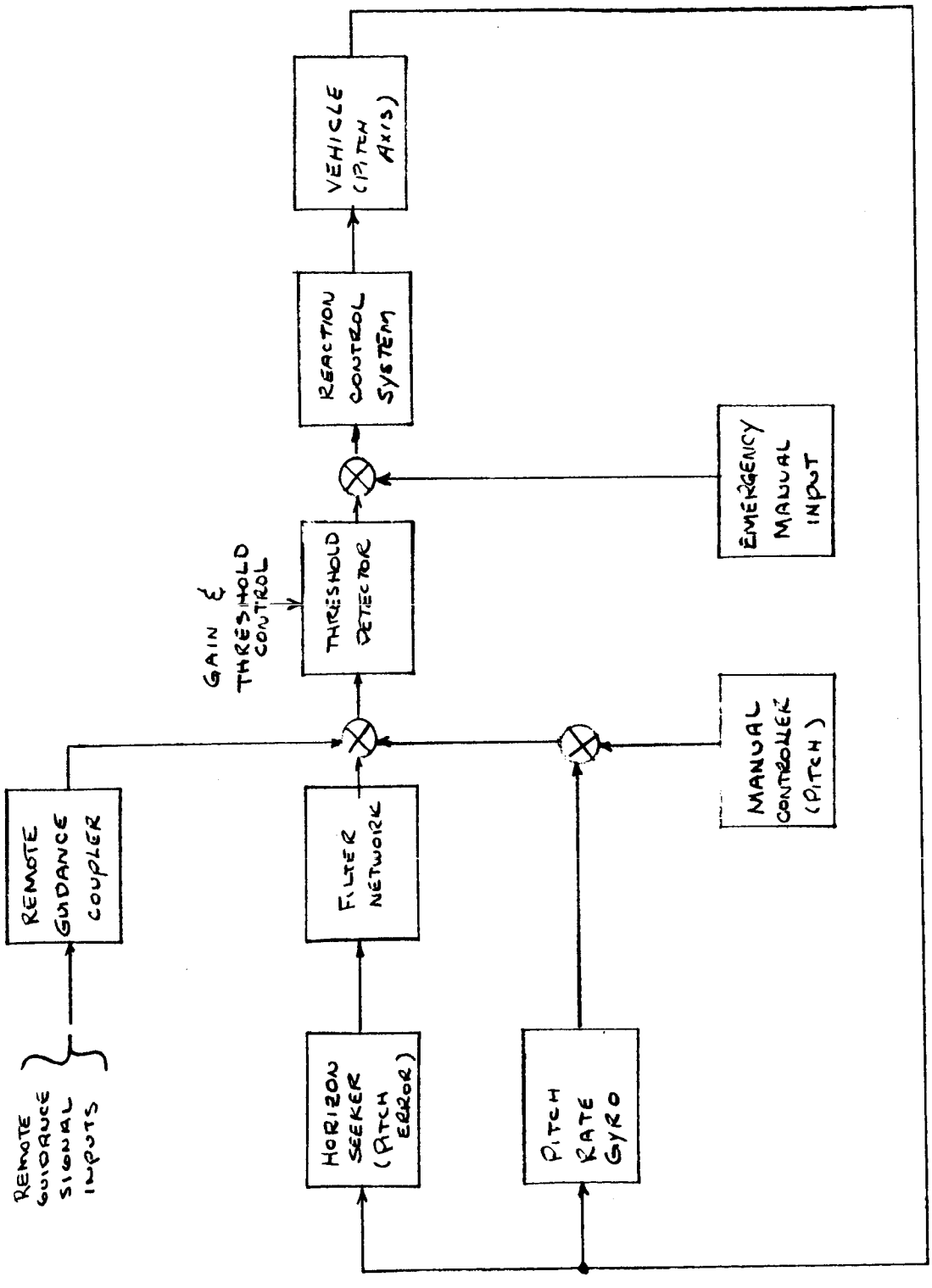


FIGURE 4-1 STABILIZATION AND CONTROL PITCH AXIS BLOCK DIAGRAM

4-degree-per-minute vehicle pitch rate required to track the local vertical, in order to prevent reaction control actuation for a satisfactorily stabilized vehicle.

During this mode of operation an integrating rate gyro could also be used for the basic attitude-stabilization reference, with the horizon seeker providing a back-up reference. This implementation would provide additional flexibility and more reliability. Utilizing an integrating rate gyro would also make the use of an electronically derived rate possible, eliminating the need for a rate gyro in this mode.

3. Docking Orientation and Fine Attitude Control

During rendezvous and docking of a payload with the OLF, the OLF's attitude will probably be required to be automatically held to one degree or less, and the vehicle must be capable of being reoriented, remotely or manually, to facilitate the mating of the OLF to the incoming vehicle.

By remote or local selection, the system's attitude and rate thresholds will be tightened to 0.5 degree and 0.05 degree per second, respectively, to provide a fine attitude control mode. The remote selection is required for rendezvous and docking of the initial incoming payload with the unmanned OLF. Remote guidance signals could conceivably be coupled to the OLF's reaction-control system to provide a capability for remotely controlling the OLF's pitch attitude for alignment during docking, but this does not appear absolutely necessary.

For a manned OLF, a manual controller is provided with a $\pm$ 30 degree attitude command capability. In addition to the pitch-attitude control, the controller provides fine attitude-control selection and gain control. A gain selector

is provided to compensate for large moment-of-inertia changes that can possibly occur when large payloads are accumulated on the OLF. Manual gain programming is proposed in lieu of a more complex adaptive control. Further study will be required to establish the need for gain control, as well as to confirm that the suggested approach be implemented.

4. Emergency Manual Control

An emergency manual control capability is provided. Provision is made for direct coupling of an emergency control to the reaction jet system to enable manual override for OLF attitude control. The override will be either through direct electrical inputs to the reaction control valve torquer or through mechanical override of the torquer output. The fail-safe characteristics of each will be considered prior to final selection.

The reaction jet controls will be authority limited for safety. Full-on (control hardover) will not disable the men on board the OLF. In the event of a system malfunction, emergency manual control is assumed to stabilize the vehicle and allow replacement of the malfunctioning equipment. Spare replacement equipment is proposed in lieu of redundancy in the interests of a less complex and therefore more reliable configuration.

(b) Roll Axis

The roll axis, shown in figure 4-2, is functionally the same as the previously described pitch axis.

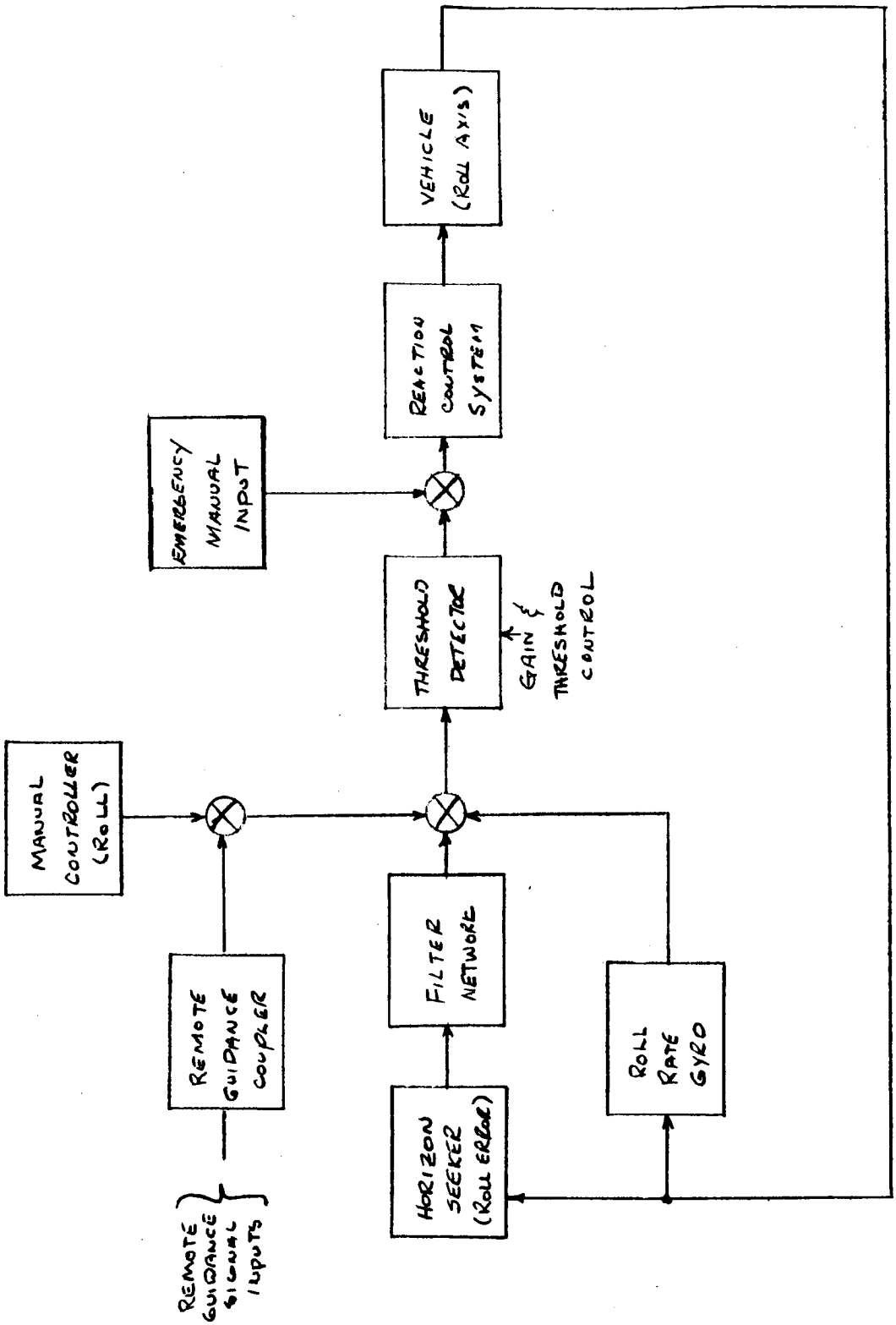


FIGURE 4-2 STABILIZATION AND CONTROL ROLL AXIS BLOCK DIAGRAM

(c) Yaw Axis

The yaw axis, shown in figure 4-3, functions in a manner similar to the roll and pitch axis except for its control reference(s) and method of manual control. The control reference for the yaw axis during initial alignment and periodic realignment phases of OLF operation is the forward velocity vector (orbit direction). Vehicle alignment to the velocity vector is desirable in order to provide proper pitch axis orientation with its attendent fuel efficiency. When the vehicle is stabilized to within the system's deadzone, the control reference is switched to a synchronized integrating rate gyro, (i.e. the gyro follows-up on itself until the switching point is reached) and the gyro then becomes the attitude reference. Deviations of the vehicle's attitude are sensed by the integrating rate gyro and when the attitude error exceeds 5 degrees (0.5 degree for fine control) or rate error exceeds the threshold, the reaction jets restore the reference attitude.

Manual or remote guidance inputs are fed through a gyro torquer amplifier to the integrating rate-gyro torque motor. This produces a gyro precession rate proportional to the input voltage. As the gyro is precessed, the vehicle attitude error grows. The yaw reaction jets are then actuated to command a yaw rate proportional to the input command. Upon removal of the command signal, (either from the manual controller or via remote guidance) the gyro stops precessing and a new reference attitude is established. Full-freedom (360 degree) yaw-attitude reorientation is available to facilitate payload rendezvous and docking. The vehicle should be manually reoriented to the velocity vector after completion of the docking maneuver.

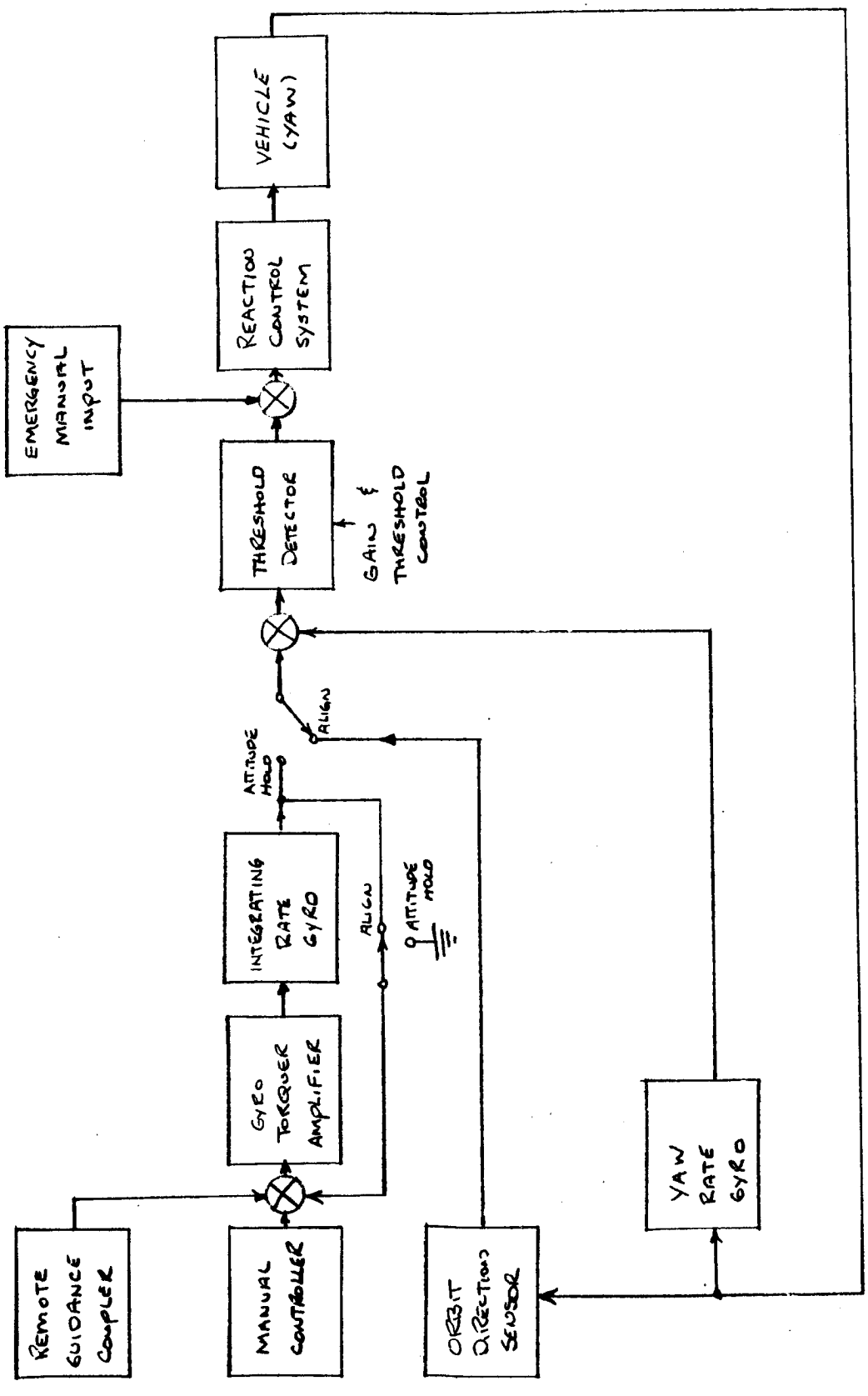


FIGURE 4-3 STABILIZATION AND CONTROL YAW AXIS BLOCK DIAGRAM

(2) Sensors

The equipments peculiar to the stabilization and control system are listed below:

Component	Volume-Ft^3	Weight-Lbs	*Power (115V,400cps)VA
Integrating Rate Gyro	0.02	2	13
Three-Axis Rate Gryo	0.04	3	24
Controller	0.05	3	1
Electronic Ass'y.	0.2	15	50
• Three threshold-detector amplifiers			
• One gyro torquer amplifier			
• Three remote-guidance couplers			
• Power and switching components			

*NOTE: All d-c power required is self generated.

The equipments required for system operation that are not considered peculiar to the stabilization and control system (i.e. equipment necessary for vehicle guidance) are as follows:

Component	Function
Two Horizon Seekers	To provide signals proportional to roll and pitch body axis deviations from the local vertical

Component	Function
Orbit Direction Sensor	To provide an output proportional to the vehicle's yaw axis orientation with respect to its velocity vector (orbit direction).
Navigation Receiver	To provide the control signals for remote control of the vehicles attitude (three axis).

The characteristics of the sensors peculiar to the stabilization and control system are as follows:

(a) Integrating Rate Gyro Assembly

This single axis assembly will mount one miniature integrating rate gyro with its required excitation circuitry. This gyro will be sensitive to the vehicles yaw attitude. The Sperry SYG 1000 is an integrating rate gyro whose performance would satisfy the requirements of the proposed OLF stabilization and control system. Performance characteristics for this unit are as follows:

Mass Unbalance (any axis)	0.2°/hr/g
Fixed torque restraint at null	1°/hr max.
Random Drift - Azimuth position	0.007°/hr
Vertical position	0.005°/hr
Anisoelastic Drift	0.01°/hr/g^2
Warm-up time	10 min. max.
weight (less excitation and mounting)	0.95 lb.

(b) Rate Gyro Assembly

This three axis rate gyro assembly will mount three miniature rate gyroscopes with their required excitation circuitry. This unit will be sensitive to vehicle yaw, pitch and roll rates. Typical performance characteristics of a unit currently used are as follows:

Threshold	0.01°/sec.
Null	0.03°/sec.
Full scale rate	$\pm$ 20°/sec.
Full scale output	5.6 volts
Motor and Pickoff Excitation	26 volts, 400 cps, single phase
Natural frequency	14 cps
Damping Ratio	0.7 ± 0.3
Weight	3 pounds max.

NOTE: Both the above units are available with self contained heaters.

The stabilization and control system electronics will employ modular construction utilizing silicon semi-conductors throughout. Signal and interlock switching will be accomplished by solid state devices (no relays will be used). Circuit redundancy will be considered where significant reliability improvement can be realized. The system will utilize 115 volt, 400 cps, single phase power only. All necessary three phase AC and DC will be self generated.

(3) Torque Generators

Three types of reaction jet systems were considered for the orbital launch facility. They are: cold nitrogen, storable bi-propellant, and H_2-O_2 bi-propellant systems. An analysis of the OLF attitude control requirements for the three systems resulted in the following weight breakdown:

Type of System	Propellant Weight per 30 day period at 0.75 lbs/day	Tare Weight	System Weight
	lbs.	lbs.	lbs.
Cold Nitrogen	24.5	55	79.5
Storable Bi-propellant	24.5	57	81.5
H_2-O_2 Bi-propellant	24.5	45	69.5

The candidate systems were rated on the basis of certain parameters which were weighted by appropriate sensitivity factors. The selection parameters and the corresponding sensitivity factors are:

Reliability	1.0
Weight	1.0
Volume	0.9
Power	0.9
Availability	0.8
State-of-Art development required	0.8
Maintainability	0.8

As a result of this trade-off procedure, the cold-nitrogen system was selected as the recommended system. This system is the first choice primarily because, based on the

present state-of-the-art, it is the most reliable. It is therefore recommended for the initial OLO vehicles. The second choice is the storable bi-propellant system. This system has a weight advantage over a stored gas system, but is not yet as reliable. It is estimated that within two years improvements in the state-of-the-art will make this the optimum system. The storable bi-propellant system is described in paragraph b. below.

Cold gas attitude control systems are in use in a number of space applications. Providing good design practice is used, they are inherently reliable, since the large background of hydraulic and pneumatic valve design and test can be applied directly to these systems without extending the state-of-the-art.

Bi-propellant systems, especially for long term storage operation, present new problems in material selection since the propellants are compatible to only a relatively few materials. This is especially true of seal materials. However, a large number of companies have been using these propellants over the last two years not only for attitude control but also as propellants for main boost engines, e.g., first state of Titan II, ICBM. This means that the backlog of information on the use of these propellants is increasing at a rapid rate and will continue over the years, especially when the Apollo program gets under way.

The cold gas nitrogen system operates in the following manner: high pressure nitrogen is stored at 5000 psi in a fibre glass container. The system is energized by a squib operated stop valve, and the nitrogen pressure is reduced to 400 psi by a pressure reducing valve. A schematic diagram of the OLF cold nitrogen system is shown in figure 4-4.

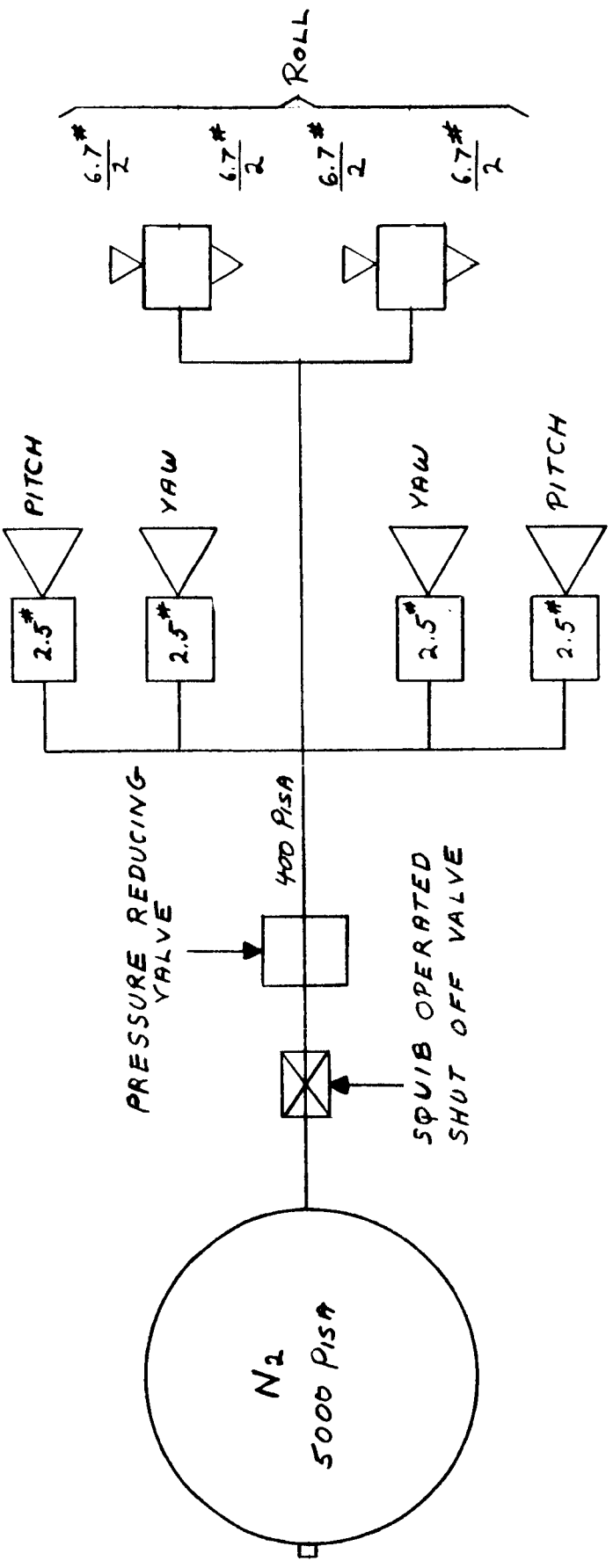
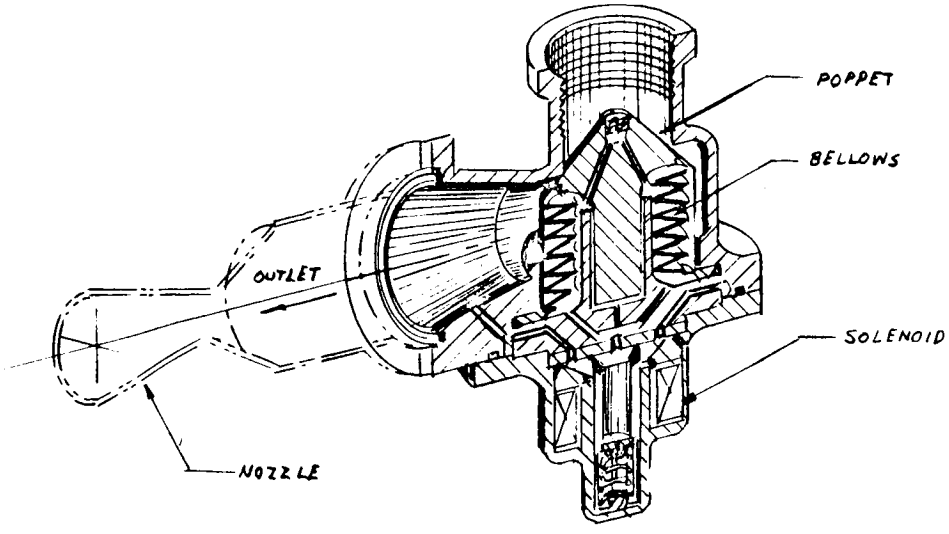


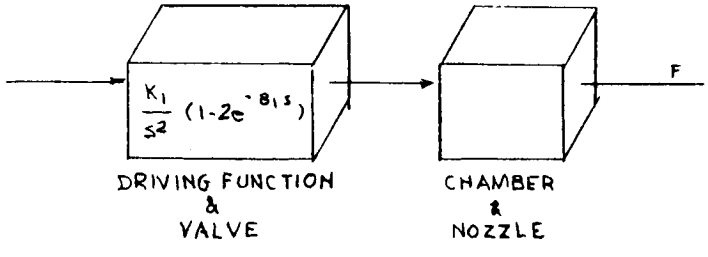
FIGURE 4-4 OLF COLD-NITROGEN SYSTEM

The control valve and nozzle package is shown in figure 4-5. The valve is energized by a pulse width modulated signal. This approach employs a solenoid-staged poppet valve that has no external leakage and develops high closing forces due to the area unbalance. The response time of this system can be adjusted by matching the valve orifice area with the solenoid orifice area. The best design configuration will minimize the gas volume behind the bellows. The position vs time curve for this valve will resemble a ramp output with a transfer function as shown in figure 4-5. Inspection of this transfer function reveals that as the pulse width input signal is decreased, the impulse applied to the vehicle will approach the impulse response of the nozzle and chamber.

b. Alternate Thrust Generator: Storable Bi-propellant System

Improvements in the state-of-the-art in the next two years are expected to increase the reliability of the storable bi-propellant system and make it the optimum choice for later OLO vehicles. A schematic diagram of the system is shown in figure 4-6. The fuel and oxidizer are stored in separate lightweight containers. A squib-operated stop valve allows nitrogen to pressurize the propellant storage containers. The nitrogen is stored at 5000 psi and reduced to 400 psi by a reducing valve. The propellant combination is hypergolic and therefore the combustion is spontaneous in the valve combustion chamber and no external ignition device is required. The valve is energized by a pulse width modulated signal. The fuel and oxidizer controls are tied together mechanically to ensure correct oxidizer lead under all conditions. The valve block diagram and a drawing of a typical unit now in operation at Vickers is shown in figure 4-7.





K_1 = SLOPE OF THE VALVE RAMP FUNCTION

a_1 = INPUT SIGNAL DURATION IN SECONDS

τ_n = NOZZLE & CHAMBER CHARACTERISTIC TIME

K_2 = THRUST TO VALVE POSITION GAIN

F = THRUST

FIGURE 4-5 SOLENOID-STAGED SYSTEM

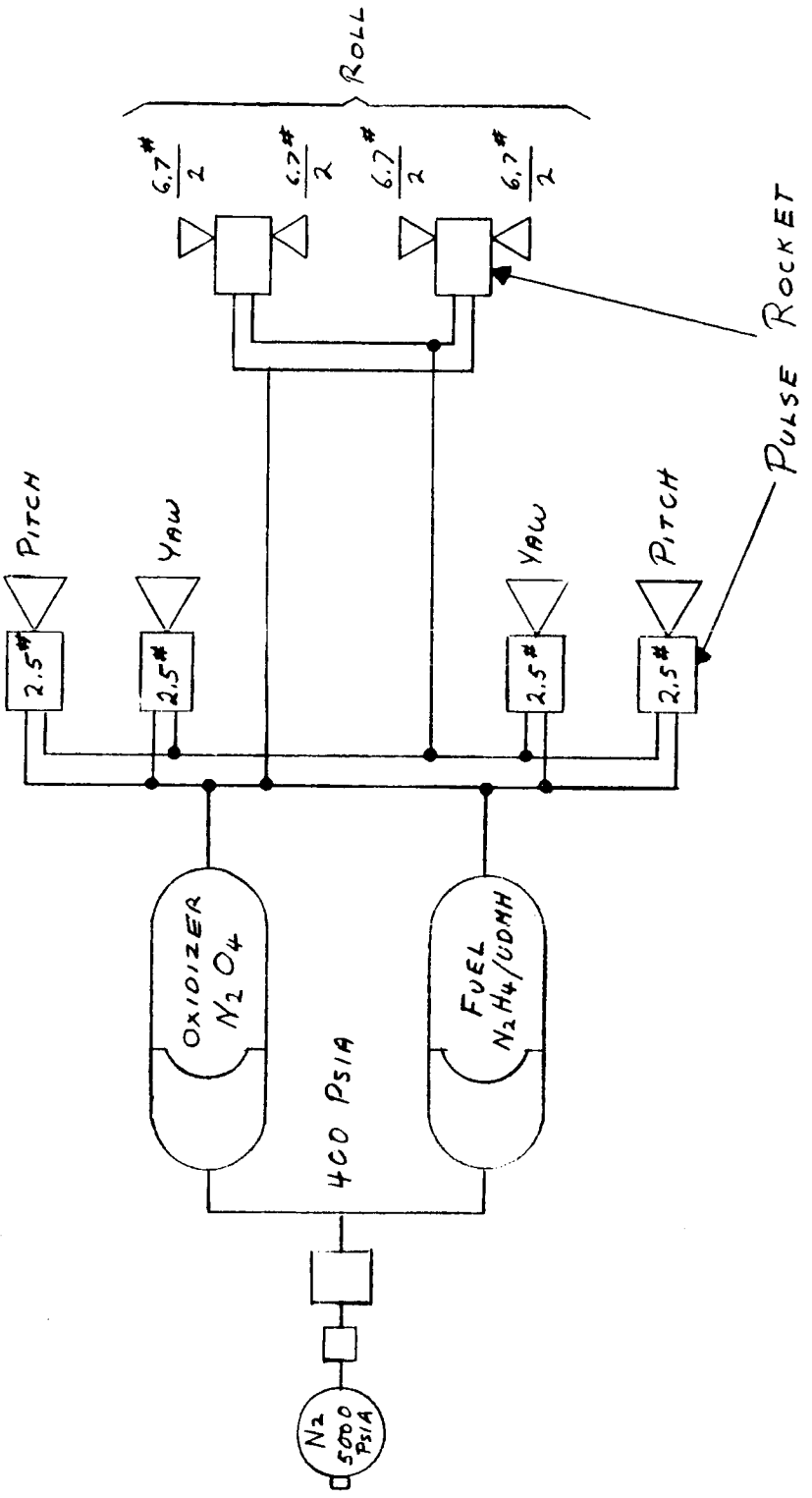
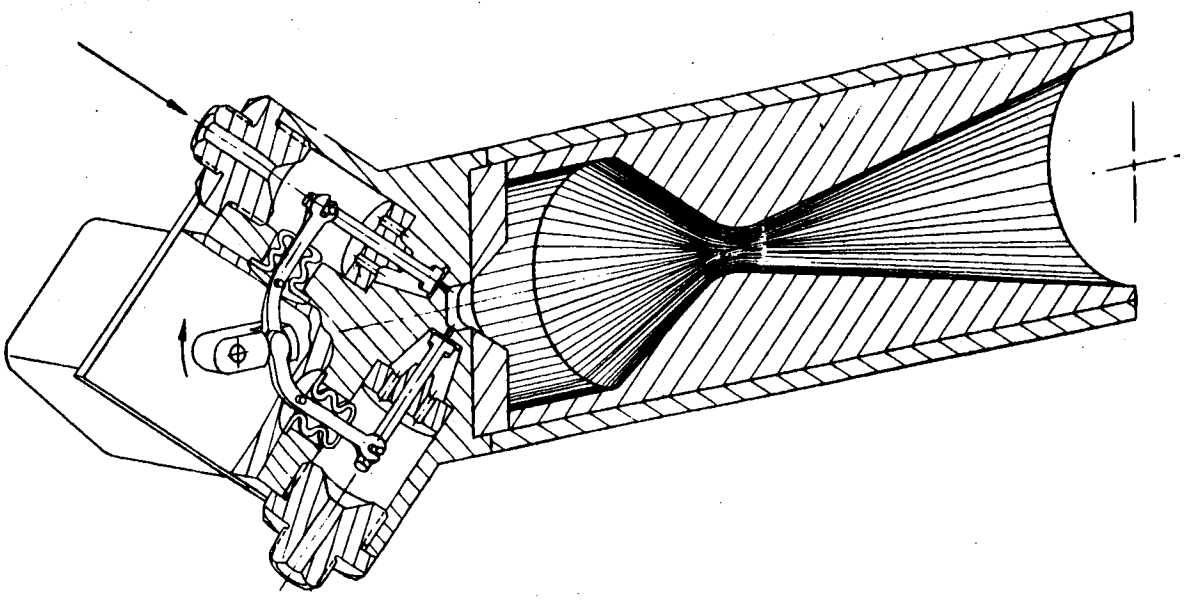
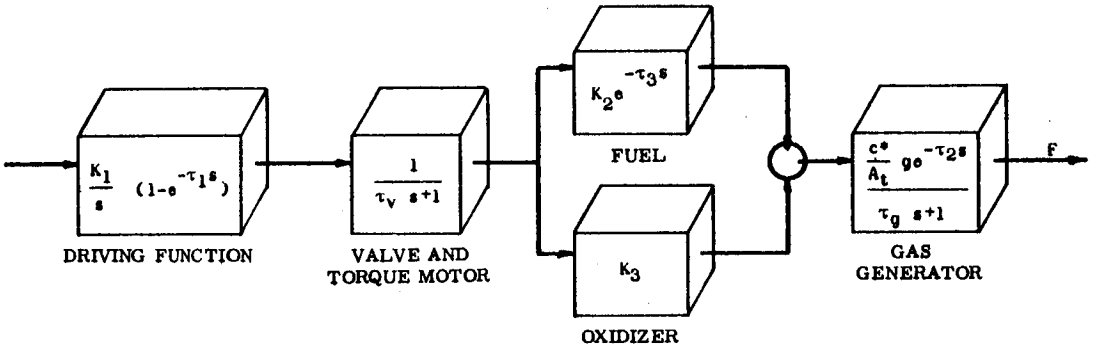


FIGURE 4-6 STORABLE BI-PROPELLANT SYSTEM SCHEMATIC



A-



.008 SEC. – τ_v = VALVE AND TORQUE MOTOR CHARACTERISTIC TIME

τ_1 = PULSE WIDTH - SECONDS

.002 SEC. τ_2 = TRANSPORT AND COMBUSTION DELAY

.002 SEC. τ_3 = OXIDIZER LEAD AND OVER-RIDE DESIGNED INTO VALVE

.001 SEC. τ_g = GENERATOR CHARACTERISTIC TIME

K_1 = INPUT SIGNAL STRENGTH - AMPS

$\frac{K_2}{K_3}$ = FUEL TO OXIDIZER RATIO = $\frac{1}{1.62}$

c^* = GENERATOR CHARACTERISTIC VELOCITY

A_t = THROAT AREA

g = GRAVITATIONAL CONSTANT

B- BLOCK DIAGRAM WITH PULSE-WIDTH MODULATED SIGNALS

FIGURE 4-7 STORABLE BI-PROPELLANT REACTION-CONTROL SYSTEM

The propellants chosen for the bi-propellant attitude control system are $N_2 O_4$ - Nitrogen Tetroxide, as the oxidizer, and Aerozine 50 as the fuel. The latter is a 50-50 combination of UDMH (unsymmetrical Dimethyl Hydrozine) and Hydrozine, $N_2 H_4$.

Since the vapor of Aerozine 50 is 90 percent UDMH, the toxicity of Aerozine 50 is the same as UDMH. UDMH is fairly toxic, a maximum allowable concentration of 0.5 ppm being recommended. It is not corrosive. A respirator with an ammonia cannister, splashproof goggles and vinyl-coated gloves should be worn. Detection levels by odor are much lower than the level of hazardous concentrations.

Nitrogen tetroxide is classified as a poisonous gas and artificial breathing apparatus is required to handle it. In an OLO application, the oxidizer and control equipment would not be stored in a crew compartment, personnel handling the material outside the vehicle will require breathing apparatus for normal condition.

The fire hazards of Aerozine 50 may be either of the fuel type or the fuel and oxidizer type. The fuel type fire presents the same problems as a gasoline fire, except that firefighting is easier in the case of Aerozine 50 because it is soluble in water. The fuel and oxidizer type fire is much more vigorous but it is controllable by the use of large volumes of water, and is free of the detonation hazards of the non-hypergolic, immiscible types of propellants.

Nitrogen tetroxide is a stable, non-flammable, shock insensitive material. In the absence of combustible materials, and when it is free of dissolved organic contaminants, there are no fire or explosion hazards associated with its use.

The combination of Aerozine 50 and Nitrogen Tetroxide has been in use at Vickers, Inc. for over one year and no accidents have occurred. A significant point to be noted is that this combination of propellant has been selected for the Apollo capsule attitude control system.

4-2. UNMANNED PAYLOAD WITHOUT TUG

a. Recommended System

(1) System Configuration

The unmanned payload will rendezvous with the OLF and dock by means of commands generated in the OLF. During rendezvous, the attitude control system must position the payload in attitude for firing the main propulsion rocket. For docking, the attitude control system must be capable of translating the vehicle laterally without rotation and be capable of positioning the vehicle accurately with respect to the OLF. The control system must, therefore, accept attitude, lateral velocity, and thrust commands, and produce torques and lateral accelerations. In addition, inertial attitude angles and velocities must be measured in the vehicle and transmitted to the OLF so that the proper attitude and lateral velocity commands may be computed.

One axis of the recommended attitude control system configuration for the unmanned payload is shown in figure 4-8. Both attitude and lateral vernier velocity control is provided, but the velocity control is used only for docking. The attitude control loop will produce an angular rate proportional to the input signal telemetered from the OLF, and will hold the existing attitude in the absence of an input. The lateral velocity control loop will produce a lateral velocity

proportional to the input from the OLF and will hold essentially zero velocity in the absence of an input. An inertial reference unit is used to measure the inertial attitude and velocity of the vehicle for transmission to the OLF. A transponder, which is used to aid radar tracking of the payload from the OLF, is also shown in figure 4-8.

The equipment required in the OLF to derive commands for rendezvous and docking of the payload is shown in figure 4-9. Although the mode of operation shown is automatic with manual override, it may also be completely manual or manual with automatic assists. The selection of a specific control method will be based on how well the human operator can perform the rendezvous and docking maneuvers in terms of time, extra fuel, and the amount of automatic equipment replaced by him in the OLF.

If manual control is used, the operator in the OLF will have a display of the payload's position, orientation, and angular rates with respect to both the OLF and inertial space. From these quantities he will determine the proper attitude rate and thrust on-off commands to accomplish a rendezvous. He will command a sequence of maneuvers in which the vehicle is alternately oriented to direct the main thrust so as to stop the rotation of the OLF/payload line of sight in inertial space, and then reoriented to direct the thrust so as to follow a certain range-range rate program. For docking, accurate payload position and orientation with respect to the OLF docking centerline will be displayed and the operator will determine the proper payload angular rates and lateral velocity commands.

If automatic control is used, the human operator will be replaced by a computer operating on essentially the

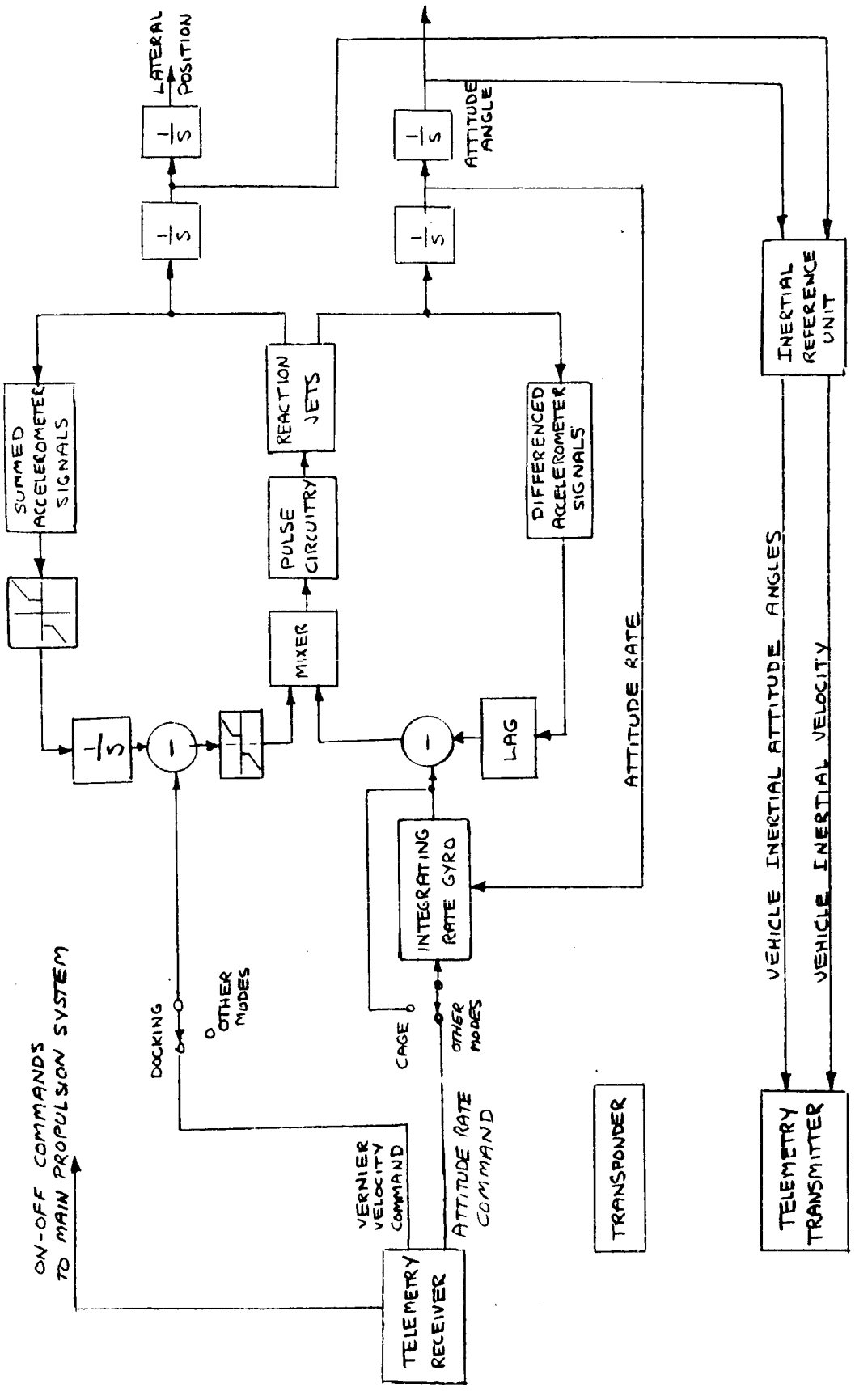


FIGURE 4-8 UNMANNED PAYLOAD ATTITUDE AND VERNIER-VELOCITY CONTROL SYSTEM (NO TUG)

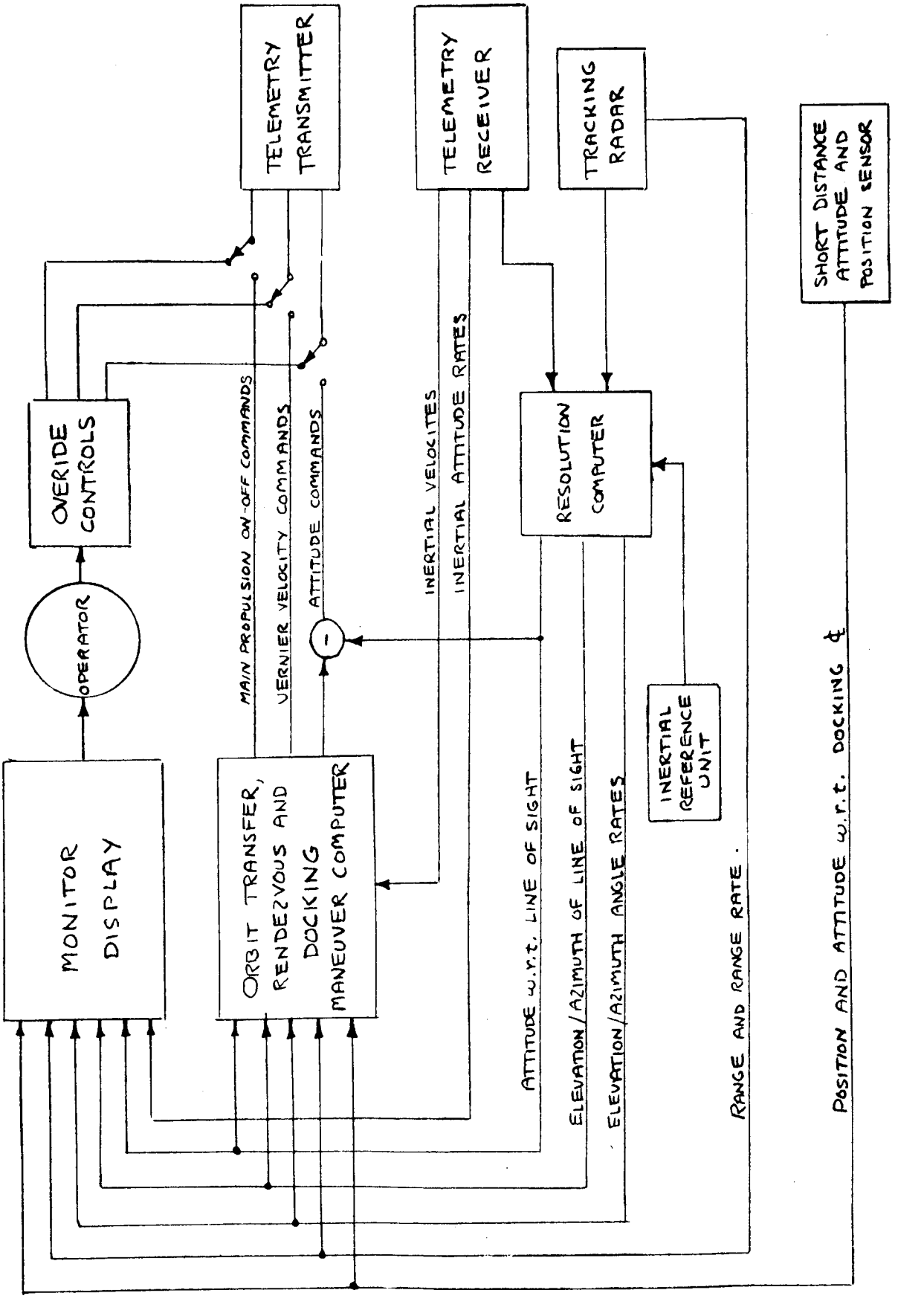


FIGURE 4-9 OLF EQUIPMENT FOR AUTOMATIC CONTROL OF UNMANNED PAYLOADS

same input information. A significant difference will be that the computer will allow more efficient (in terms of time and fuel) rendezvous and docking maneuvers.

(2) System Operation

The torque generators for both the attitude and lateral velocity control loops will be pairs of fixed-thrust reaction jets of the type described in paragraph 4-1. Torques about the vehicle axes are produced by two pairs each of pitch, roll, and yaw reaction jets. Accelerations along the vehicle axes will be produced by a pair of the pitch or yaw reaction jets firing in the same direction and by an additional pair of fore-aft reaction jets. The signal mixer and pulse circuitry shown in figure 4-8 determine the selection of reaction jets and the impulse duration to produce the desired angular on lateral acceleration.

The body mounted integrating rate gyros provide a "floating" attitude displacement reference. The operation of the IRG is illustrated in figure 4-10. The IRG is sensitive to vehicle rates about its input axes and produces a displacement of its output axis proportional to the angular rate. The gyro can also be torqued about its output axis. This results in an output rate proportional to the input (torquing) signal, which represents an angular rate command. The gyro output is proportional to the integral of the difference between the commanded and actual angular rate and is therefore a displacement error.

An angular rate damping signal is derived by passing angular acceleration signals, obtained from accelerometers, through a lag network. The manner in which the outputs of two linear accelerometers can be combined to measure angular

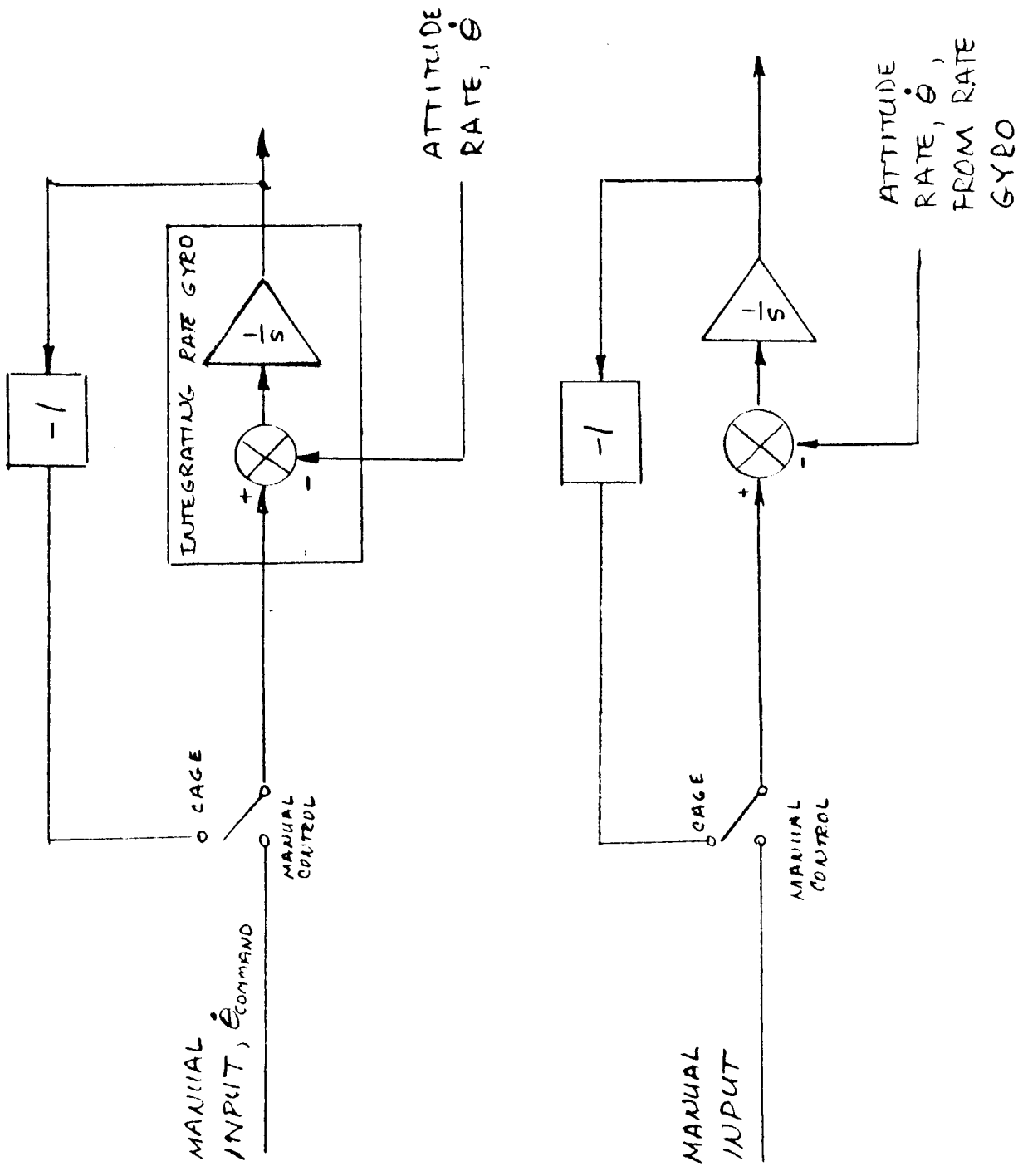


FIGURE 4-10 DRIVEN DISPLACEMENT CONTROL SYSTEM WITH INTEGRATING RATE GYRO VS ELECTROMECHANICAL INTEGRATOR

acceleration is shown in figure 4-11. Linear accelerometers have been suggested as angular acceleration sensors because they are already in the vehicle being required for lateral path-damping signals, and because they are more reliable than rate gyros.

Initial orientation of the payload before it is acquired by the OLF is accomplished by having the IRG follow-up on itself (electrical caging mode) until just prior to booster separation. The payload will then hold the attitude existing at separation.

In the lateral velocity control loop, velocity is measured by integrating the sum of the two linear accelerometer signals mentioned above. Figure 4-11 also shows how the accelerometer signals can be combined to measure the linear velocity of a point in the vehicle. The error between the commanded and measured velocity results in a proportional lateral acceleration.

It should be noted that during rendezvous, the velocity control loop is open and the payload will rotate about its center of gravity in response to attitude commands. When docking, however, both loops are operating and the summed accelerometer signal in the velocity loop will sense the velocity of a point whose location is determined by the manner in which the signals are summed (see figure 4-11). The velocity loop will prevent this point from translating by redistributing the torque commands called for by the attitude loop. This will cause the vehicle to rotate about the selected point rather than the center of gravity. The ability to force the payload to rotate about an arbitrary point which is obtained in this manner may be useful in the docking maneuver.

To accomplish a rendezvous by manual control from the OLF, the operator will generate attitude rate commands

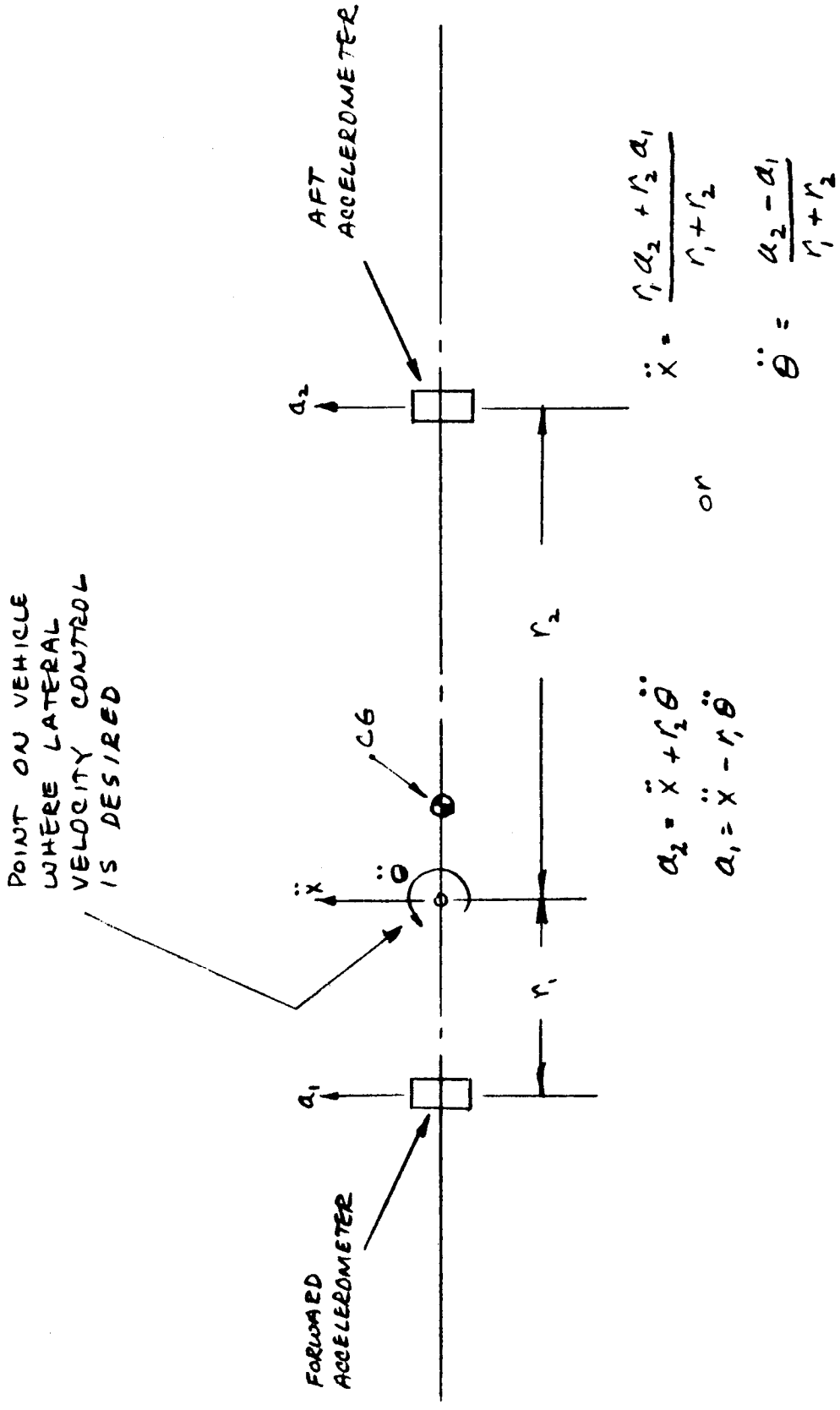


FIGURE 4-11. SCHEMATIC ARRANGEMENT OF LATERAL ACCELEROMETERS FOR MEASURING ANGULAR AND LINEAR ACCELERATIONS

and rotate the payload until it is at the proper attitude for firing the main rocket as indicated by his display of payload attitude with respect to the line of sight, measured in payload body axes. These angles are obtained by combining the payload and OLF inertial orientation angles with the tracking radar orientation with respect to the OLF. The proper attitude for firing, and the required thrust duration, are determined by the other displayed quantities and the rendezvous technique being used. Some of the displayed quantities shown in figure 4-9 will probably be pre-digested automatically to simplify the operator's task. The payload may be docked using a display of payload position and orientation with respect to the OLF docking centerline, or by direct visual observation. The operator will generate attitude rate and lateral velocity commands.

With automatic control of rendezvous from the OLF, the proper payload attitude for thrusting will be determined by a computer. Payload attitude will be controlled in a closed loop manner by comparing the payload orientation with respect to the line of sight to the computed orientation. The attitude error will be transmitted to the payload as an attitude rate command.

For automatic docking, closed loop control of payload position with respect to the OLF docking centerline will be provided. Position errors measured in payload body axis will be transmitted to the payload as lateral velocity commands.

(3) Sensors

(a) Integrating Rate Gyro

The performance specifications of a typical IRG are listed in paragraph 4-1. This gyro will satisfy the requirements

of the unmanned payload control system shown in figure 3-2. Since the drift rate is less than 0.01 deg/hr., the IRG is also a satisfactory long period displacement reference for the attitude control loop prior to OLF acquisition.

(b) Linear Accelerometers

Preliminary calculations indicate that the linear accelerometers used to measure angular and linear accelerations can be of the E-pickoff type. It is true that a null output error will result in an attitude angle error and a residual lateral acceleration, and that any mismatch of accelerometer gradients will result in cross-coupling effects. For example, lateral velocity error will arise from a constant angular rate and an attitude angle error will be caused by a constant lateral acceleration. However, the magnitudes of these effects have been calculated and appear to be negligible.

The accelerometer null output will require an IRG output and therefore a positioning error to cancel it. A null output of 1% of full scale will cause an attitude hang-off of approximately .1 deg., which is within the positioning accuracy tolerances. The accelerometer null output also enters the velocity control loop and results in a steady state acceleration. Thus the null error must be removed by deadzones on the accelerometer output and on the velocity error, as shown in figure 4-8. This, in turn, will impose a velocity threshold on the system. The minimum value of this threshold is a function of the dynamic response of the accelerometer-integrator combination (not shown), and the characteristics of the pulse modulation circuitry and reaction jets. The velocity threshold of 0.01 fps called for in the table of figure 3-2 for the docking mode appears to be

unrealistic and some difficulty is anticipated in achieving it. If the threshold specification cannot be met using accelerometers, an alternate solution to the lateral control problem is required. One possibility is to provide closed loop control of payload lateral position. The short range position sensor would serve as a position reference and derived lateral velocity would be used for damping.

The cross-coupling errors which are caused by mismatched accelerometer gradients have also been estimated. With a constant angular rate of 12 degrees per second, a 1 percent mismatch in gradients will cause a 0.02 fps velocity standoff. With a constant linear acceleration of 10 fps^2, a 1 percent mismatch will cause a 3-degree attitude error. Cross-coupling errors are not expected to be a problem in the docking maneuver, since the angular rates and lateral accelerations are low and are of short duration.

(4) Torque Generators

Four types of reaction-jet systems were considered for the unmanned payload. An analysis of the unmanned payload requirements resulted in the following weight breakdown.

Type of System	Propellant Weight	Tare Weight	System Weight
Cold Gas-Nitrogen	139.35 lbs	263.65 lbs	403.0 lbs
Storable Bi-propellant	32.8 lbs	119.2 lbs	152.0 lbs

The systems were evaluated using the same selection parameters and weighting factors as for the OLF. (See paragraph 4-1, a.) The results of the trade-offs are as follows:

First Choice - Cold Nitrogen
Second Choice - Storable Bi-propellant

Here again the cold Nitrogen system is the first choice, primarily because of its present reliability advantage. Within two years, however, improvements in the state-of-the-art are expected to make the storable bi-propellant system the optimum choice.

b. Alternate Systems and Trade-offs

(1) Alternate Sensors

Alternate sensors that have been considered for use in the payload attitude control loop are rate gyros in place of the accelerometers and a stable platform attitude reference in place of the IRG.

A rate gyro can be used in place of the accelerometers and lag network to provide an attitude rate damping signal. The two techniques are dynamically equivalent. The factors involved in the choice of sensor are as follows:

- Accelerometers are required for lateral path damping and are already in the system.
- Calculations show that simple E-pickoff type accelerometers will probably be satisfactory. These are more reliable than rate gyros.
- The problems that arise in measuring aircraft angular acceleration with linear accelerometers, due primarily to gradient mismatch and null imbalance, do not appear to be significant in space. The difference is due to the absence of vibration and the low range required.

- The use of rate gyros will eliminate the cross-coupling errors between the lateral velocity and attitude control loops. At the moment these do not appear to be a problem.

The inertial attitude angle reference (platform) already in the payload can conceivably be used in place of the IRG. However, resolvers will be required to derive body axis angles from the inertial angles, and a synchronizing integrator will also be required in each axis to synchronize the control system with the resolved platform outputs prior to initiation of remote control. Manual control of the payload from the OLF would be the same, with the operator generating angular rate commands. These would be fed through the synchronizing integrators to give a "driven displacement" system similar to that obtained with the IRG's. The factors involved in the choice of IRG sensors are as follows:

- The use of the IRG eliminates the need for resolvers and synchronizers in the payload and places them in the OLF where they can be used for each payload.
- The IRG provides initial orientation of the payload when placed in follow-up prior to booster separation.

4-3. SPACE VEHICLE (S/V) NO TUG

In the absence of tug assistance, the S/V must be taxied to a launch site some distance from the OLF and oriented for launch. This involves orientation of the vehicle to thrust away from the OLF, coast to the launch site, reorientation for braking at the launch site, and orientation for launch. The latter maneuver may be controlled from the OLF in the same manner as the unmanned payload in order to take advantage of its centralized sensing and computing facilities. The control system requirements will be similar to those of the unmanned payload and will not be discussed in detail here. (See paragraph 4-2.)

The torque generators for the Space Vehicle will operate on hydrogen from the main propulsion tanks. This is the same system as is recommended for the astrotug. The system is described in paragraph 4-4, a, (4), and shown in figure 4-12.

4-4. MANNED ASTROTUG

a. Recommended System

(1) System Configuration

The astrotug's functions include rendezvous and docking with the incoming payload, followed by rendezvous and docking at the OLF with the payload in tow. The rendezvous technique assumed is a simple one based on the results of NASA simulator studies as described in paragraph 3-1,d. Control of tug attitude will be required for rendezvous, while control of both attitude and lateral velocity will be

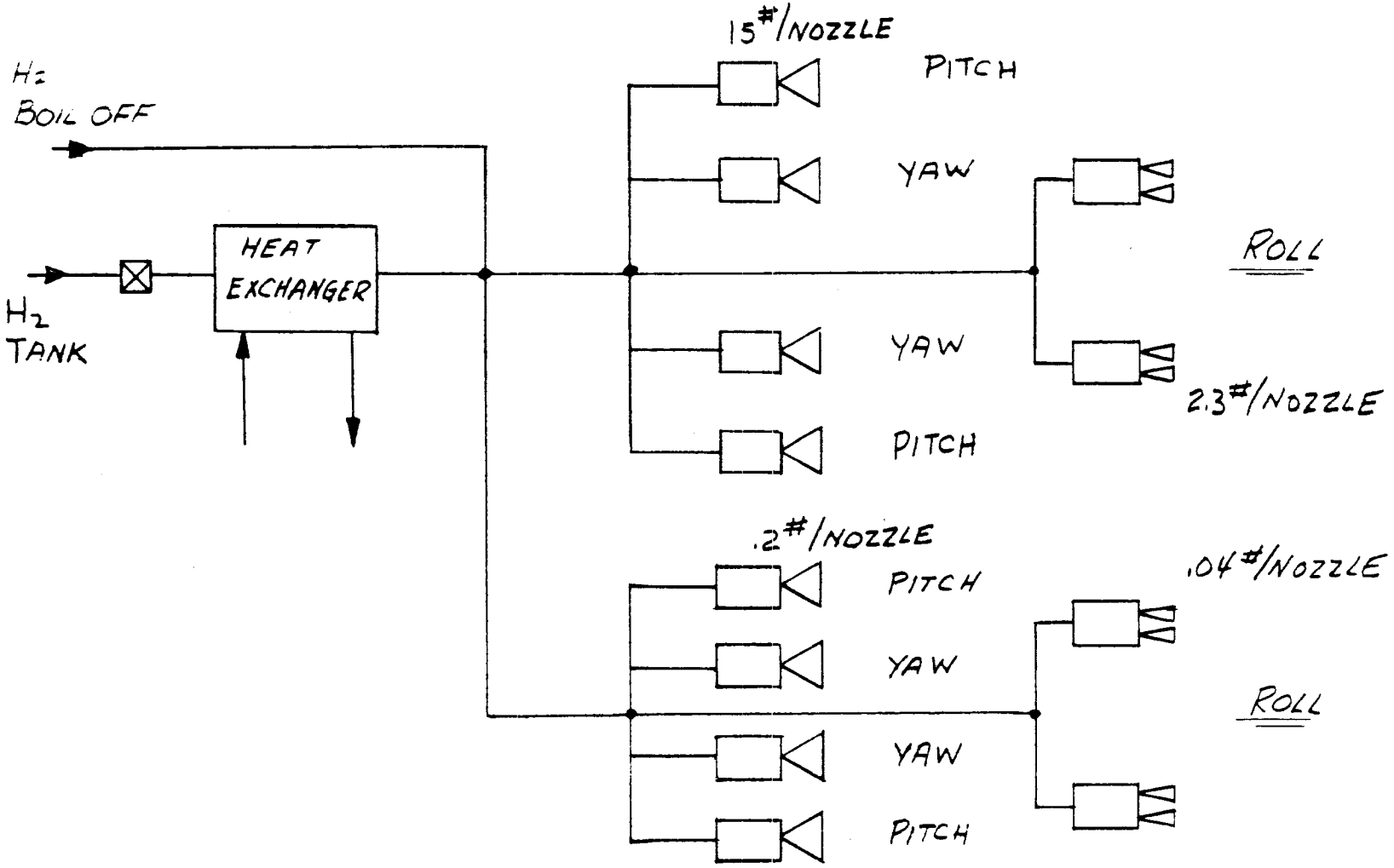


FIGURE 4-12 S/V SCHEMATIC

WEIGHT ESTIMATE (SINGLE SYSTEM : NO REDUNDANCY) = 42 LBS.

required for docking. During payload return, the payload attitude control system may or may not be coupled to the tug control system. The system configurations discussed below cover both possibilities.

One axis of the recommended system configuration for the manned astrotug is shown in figure 4-13. The system provides control of attitude rate and lateral acceleration. The attitude control loop will produce a vehicle rate proportional to the pilots input command and will provide short period attitude hold in the absence of an input. This basic, or minimum, system places the maximum workload on the pilot. The pilot must also be provided with a display (not shown) of tug position, orientation, and angular rates with respect to the payload or OLF and to inertial space. An inertial reference unit and a tracking radar will be required to provide information for the display. With these quantities the pilot can determine the proper attitude rate and thrust on-off commands to accomplish a rendezvous. Optical references will be used in docking on the payload and OLF.

The ability of the pilot to accomplish efficient rendezvous and docking maneuvers with the minimum system is questionable. An augmented tug control system is shown in figure 4-14 which will provide pilot relief as well as more precise control by using additional equipment. Both attitude angle and lateral velocity control loops are provided. The attitude control loop will accept attitude rate commands from the pilot and hold the existing attitude angle accurately in the absence of a command. The lateral velocity control loop will accept lateral velocity commands and hold essentially zero velocity in the absence of a command.

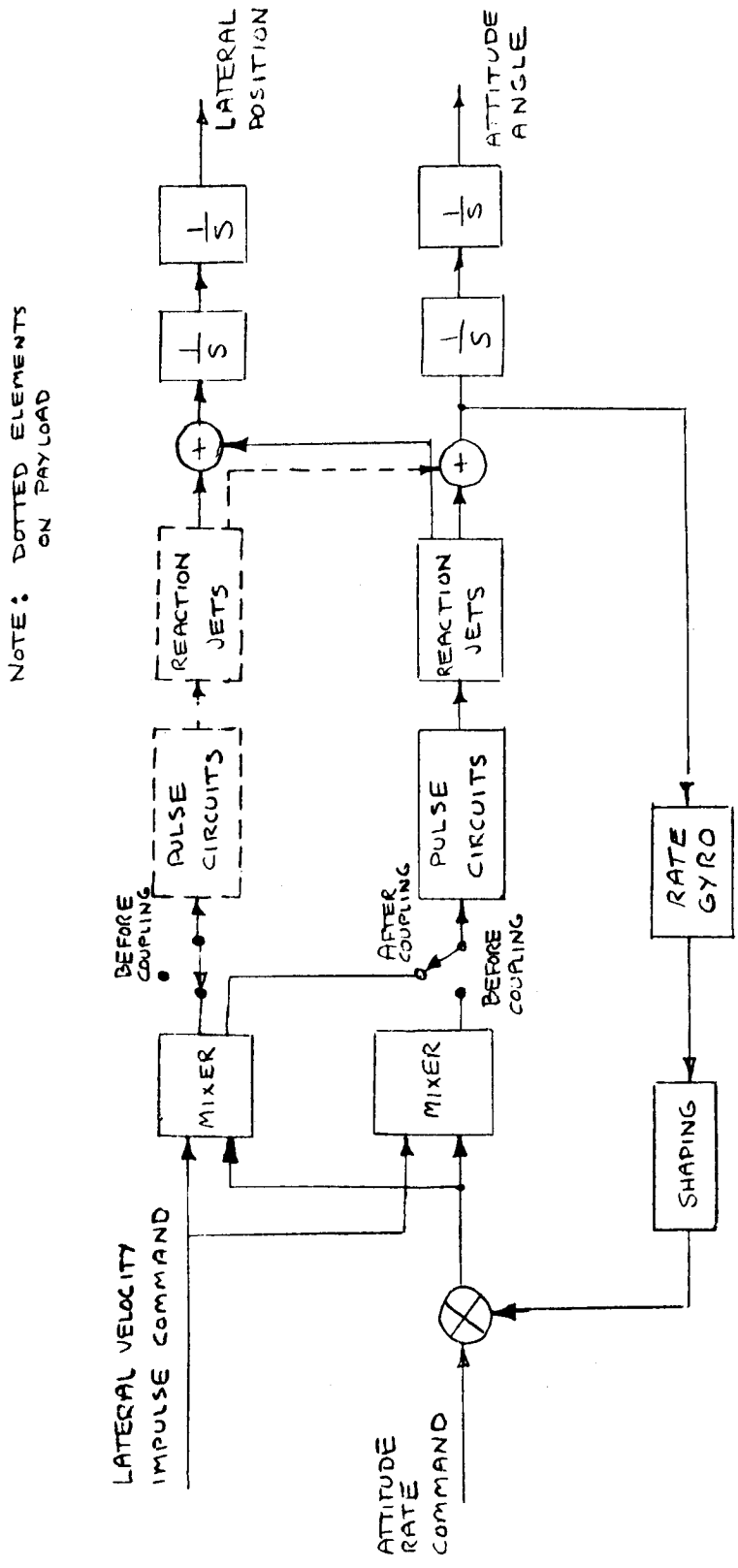


FIGURE 4-13 MINIMUM ATTITUDE-CONTROL SYSTEM FOR MANNED ASTROTUG

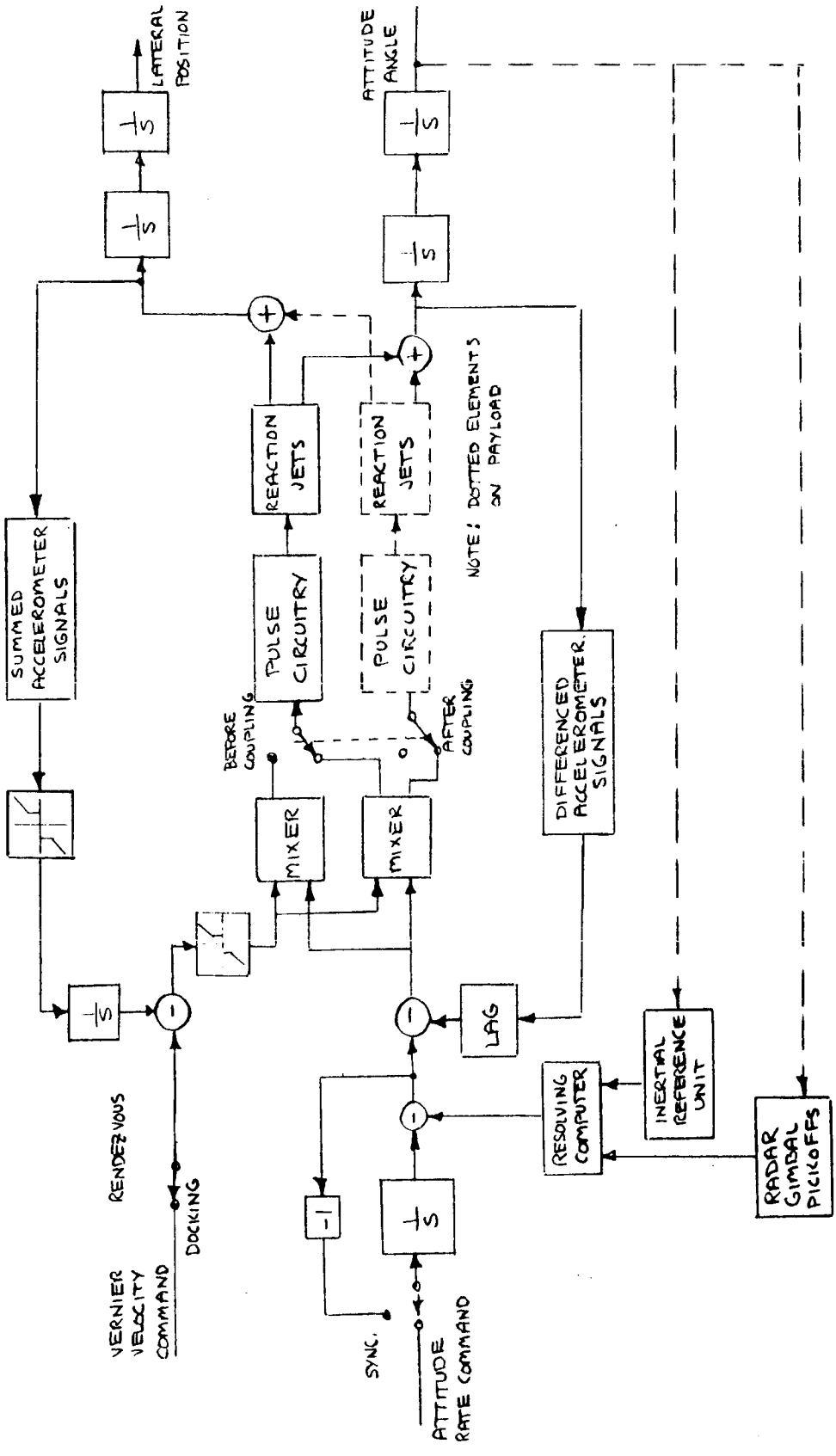


FIGURE 4-14 AUGMENTED ATTITUDE AND VERNIER-VELOCITY CONTROL SYSTEM FOR MANNED ASTROTUG

(2) System Operation

The torque generators for the tug control system will be fixed thrust reaction jets of the type described in paragraph 4-1. The signal mixers and pulse circuitry shown in figures 4-13 and 4-14 determine the selection of reaction jets and the impulse duration to produce the desired angular and lateral acceleration.

In the minimum system rate gyros are used to sense angular rates in body axes. The difference between the measured angular rate and the input signal commands a proportional torque. To obtain a lateral velocity, the pilot applies a lateral velocity impulse, that is, he determines the duration of the lateral thrust directly and closes the loop through his display.

The augmented tug control system is similar to unmanned payload system described in paragraph 4-2. Both systems make use of summed and differenced linear accelerometer signals to measure lateral and angular accelerations. The two systems differ only in the method of deriving an angular displacement error and of inserting commands into the attitude loop. The tug attitude control system derives the attitude of the vehicle by resolving signals from an inertial reference unit and radar gimbal pickoffs. Synchronizing integrators are required to assure a zero attitude angle error signal prior to engaging the attitude control system. In normal operation, attitude rate commands are inserted through the synchronizing integrators. This results in a "driven displacement" type of control.

The operation of the velocity control loop and the interaction of the velocity and attitude control loops during

docking is identical to that of the unmanned payload system and is described in paragraph 4-2,a,(6).

(3) Sensors

(a) Rate Gyro

The performance specifications of a typical 3 axis rate gyro assembly are listed in paragraph 4-1. In the minimum system configuration, this gyro would satisfy the requirements of the tug control system shown in figure 3-3. With a gyro null of 0.03 deg/sec. it will take 16 seconds to exceed the 0.5 deg. positioning accuracy specification. This appears satisfactory for short period attitude hold. The threshold rate of 0.01 deg/sec. is within the specification.

(b) Linear Accelerometers

Accelerometers are discussed in paragraph 4-2,a,(3).

(c) Inertial Reference Unit

The inertial reference unit is required in the tug to supply information for the pilot's display for navigation during the rendezvous maneuver. Its direct use in the augmented attitude control system is discussed in the next section. No difficulty is anticipated in meeting the attitude control specifications with this sensor.

(4) Torque Generators

A schematic diagram of the reaction jet system recommended for the manned astrotug is similar to that shown for the S/V. The system operates on hydrogen from the main propulsion tanks, assuming that this is one of the propellants

used for the tug thrust engine. For the majority of operations, there will be an adequate supply of boiloff from the cyrogenic hydrogen storage. However, for a condition where additional hydrogen is required, a supply of liquid hydrogen will be taken from the main tank and flashed in a heat exchanger operating on waste heat from the crew or electronic compartments. A gas pressure of 10 psi and a gas temperature of 460°R are assumed for this system.

There are two sets of nozzles for control about each axis. One set will produce high torque levels for payload rendezvous and return, the other set will produce low torques for the payload coupling. The maximum thrusts required per nozzle for the four tug functions are tabulated below.

	Payload Rendezvous	Payload Coupling	Payload Return (No Help	Payload Docking From Payload)
Max. Thrust per nozzle	Pitch 73.4	1.0	133.0	26.7
	Yaw 73.4	1.0	133.0	26.7
	Roll 40.0	0.3	80.0	8.0

Due to the variation in thrust levels required for the four tug functions, a variable throat area nozzle is proposed to provide the three highest values shown above. A small unit will be used for payload coupling.

The weight breakdown for the system is as follows:

System tare weight	113 lbs
Propellant weight chargeable to attitude control	77 lbs
Total System Weight	190 lbs

b. Alternate Systems and Tradeoffs

(1) Alternate Configurations

The two configurations considered for the astrotug, that is, the minimum and augmented systems, have already been described. The minimum system may be all that is needed for rendezvous and docking and has the advantage of being simpler and probably more reliable. The advantages of the augmented system are as follows:

- Control of attitude displacement through a long period displacement reference is more accurate and provides more pilot relief than short period control through a rate reference, particularly in the docking maneuver.
- Closed loop control of lateral velocity is more accurate and provides more pilot relief than open loop control.
- The "driven displacement command" type of control combines a desirable maneuver characteristic (rate inputs from the pilot) with accurate attitude displacement control.

(2) Alternate Sensors

Rate gyros can be used in place of accelerometers in the attitude loop. This alternate was discussed in paragraph 4-2,b,(1).

Integrating rate gyros can be used in place of the inertial reference, radar pickoffs, resolving computer, and synchronizers to provide a displacement reference for the attitude loop. This tradeoff was evaluated for the unmanned

payload control system with the result being the choice of the IRG. The factors involved in the choice of the inertial reference unit, et. al., for the astrotug are as follows:

- The inertial reference unit radar gimbal pickoffs, and resolving computer are already required on the astrotug for driving the pilot's display. (This was not the case in the unmanned payload tradeoff).
- With "driven displacement" control, the output of the synchronizing integrator, which is the actual displacement command, can be displayed permitting a more precise control of attitude. The displacement command can be positioned by the pilot and the system will follow-up on it with no error.

4-5. UNMANNED PAYLOAD (WITH TUG)

a. Recommended System

(1) System Configuration

When tug assistance is available, the unmanned payload control system is required to damp the initial tumbling and upon command from the tug, hold its attitude while the tug attaches itself. The control system may also be required to accept thrust commands from the tug in the event that the payload has to provide all or part of the attitude and/or propulsion forces required for the return trip and docking.

The recommended control system configuration for the payload is shown in figure 4-15. The torque generators and the associated control circuitry is similar to that described for the other vehicles. Integrating rate gyros are used as the attitude reference.

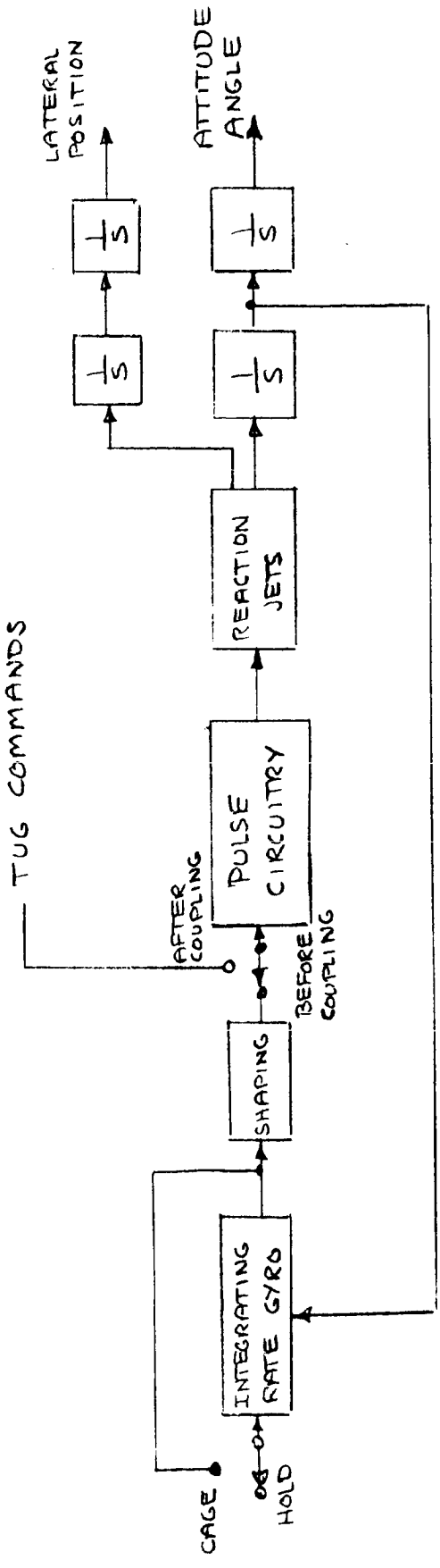


FIGURE 4-15 UNMANNED PAYLOAD ATTITUDE-CONTROL SYSTEM (WITH ASTROTUG)

(2) System Operation

The integrating rate gyro is sensitive to body axis rates and produces an error signal proportional to the integral of the rate, that is, a displacement error. The IRG is caged during the injection phase and released just prior to booster decoupling. This establishes a reference attitude which the control system will hold while waiting for the tug to arrive. The shaping network is used to provide damping in the loop.

Two modes of operation are provided in the control system. One is an equivalent "low gain" mode for the period before tug arrival when it is only necessary to hold attitude to ± 5 degrees (see figure 3-4). The second is a high gain mode for the tug coupling maneuver when it is necessary to hold attitude to ± 0.5 degree. The two modes may be implemented by changing the error threshold in the pulse circuitry on command from the tug. The use of an equivalent low gain mode will reduce the weight of propellant required for attitude hold.

(3) Sensors

(a) Integrating Rate Gyro

The performance specifications of a typical IRG are listed in paragraph 4-1. This gyro satisfies the requirements of the unmanned payload control system shown in figure 3-4. With a drift rate of 0.01 degree per hour it will require 500 hours to exceed the 5-degree positioning-accuracy requirement of the initial alignment mode.

b. Alternate Systems and Trade-Offs

(1) Alternate Configurations

During payload return it may be advantageous to couple the payload reaction jets into the tug control system to provide all or part of the attitude control and/or propulsion forces required. The factors involved in evaluating these alternatives are the torque and thrust requirements and the fuel requirements for the payload and tug. The tug and payload requirements are shown in figures 3-3 and 3-4 respectively. The pertinent fuel trade-off comparisons are as follows:

- The maximum torque output required from the tug control system is almost doubled (4000 to 2200 ft-lbs) if there is no coupling. (See figure 3-3).
- If coupling exists the payload system must provide all or part of 4000 ft-lbs maximum compared to 26 ft-lbs with no coupling (figure 3-4).
- The maximum tug thrust required (4000 lbs) is the same with or without thrust augmentation from the payload.
- If payload thrust augmentation is used the payload must provide all or part of 4000 lbs thrust, compared to no requirements with no thrust augmentation.
- With no payload assistance the tug uses 400 lbs more fuel than with assistance.
- With payload assistance the payload uses 300 lbs more fuel than with no assistance.

From the above comparisons it can be seen that the tug payload coupling question involves a number of other factors including the design of the torque and thrust generators of both vehicles and cannot be answered at this time.

(2) Alternate Sensors

An inertial reference unit could be used in place of the three integrating rate gyros in the payload control system. However, this alternate has nothing to recommend it. The IRG's are satisfactory and also enjoy a considerable size and weight advantage.

A rate gyro and an electromechanical integrator could also be used in place of an IRG. This alternate has a size, weight, and reliability disadvantage. The combined effect of the rate gyro and integrator null outputs in cascade gives it an accuracy disadvantage as well.

4-6. SPACE VEHICLE (WITH TUG)

a. Recommended System

The recommended system for the OLV when a tug is available is the same as that used for the unmanned payload. The discussion in paragraph 4-5 also applies to the OLV.

SECTION V

COST AND SCHEDULE

5-1. GENERAL

A summary of the estimated cost of developing and procuring attitude-control equipment hardware for OLO vehicles through 1966 is given in this section.

The program and expenditures recommended for calendar year 1962 are discussed in detail in Section VI.

5-2. BASIS FOR ESTIMATE

Development cost estimates are based on a generalized four year program plan for all vehicles. Major milestones in this program include the following:

First Year - Systems study and design reliability evaluation

Establishment of system specifications.

Second Year - Preparation of engineering models

Performance Evaluation of system models

Reliability evaluation test

Environmental evaluation tests

Component Design

Product Follow-up

Design of Support Equipment

Third Year - Product Follow-up

Preparation of operating and service instructions

Formal Environmental Tests

Formal Reliability Evaluation Tests

Field Test Follow-up

Fourth Year - Product modifications

Product Follow-up

Reliability Evaluation and Follow-up

Field Tests Follow-up

The production schedule consists of delivery of the first engineering prototype early in the third year, with further prototypes delivered throughout the third year. The first production model would be delivered for field testing early in the fourth year.

Hardware procurement cost estimates were obtained by comparison with production hardware of similar complexity. It should be noted that both development and procurement costs reflect the heavy emphasis placed on reliability. The proposed reliability program is discussed in Section VI.

Development and procurement expenditures are scheduled in accordance with the "typical program" outlined by Vought Astronautics. The major milestones in this program are

Circumlunar Mission	March 1966
Lunar Orbit Mission	July 1966
Lunar Landing Mission	April 1967

Requirements for operational availability of the individual vehicles in the OLO complex have not been established by Vought Astronautics and are beyond the scope of an attitude control system study. The following assumptions were made for purposes of estimating nominal delivery rates:

- Unmanned orbital flight tests of the initial orbiting facility, the orbital launch vehicle, and the initial unmanned payload will be performed in 1964/65 period. These vehicles must attain operational status by the end of 1965 in order that manned tests can be conducted during 1966. At least ten sequentially successful unmanned test flights will be required before a vehicle achieves a man-rated status.
- The first application of the initial orbiting facility, the initial unmanned payload and the initial orbital launch vehicle will be in the lunar orbit missions of mid-1966. The circumlunar mission will be a direct shot with no orbital operations.
- Unmanned orbital flight tests of the intermediate orbiting facility, the intermediate unmanned payload, and the astrotug will be performed during the 1965/66 period. Manned tests will follow in late 1966, with the first application of these vehicles occurring in the lunar landing mission of early 1967.

5-3. ESTIMATED COSTS AND SCHEDULE

A summary of attitude control equipment development costs is given in Table 5-1. Several assumptions are implicit in the figure in addition to the basic estimating procedures outlined in the preceding section.

- The cost estimates are based on concurrent development of all systems by Sperry Rand.

- The cost of the orbital launch facility includes development of sensors and attitude controls for use during retrieval and docking. This represents a major OLO problem area.
- The initial orbiting facility and unmanned payload utilize cold nitrogen reaction controls while the intermediate models have storable bi-propellant systems. The quoted price includes the funds necessary to complete development of the storable bi-propellant systems.

A summary of attitude control hardware procurement costs is also shown in table 5-1. The costs are, of course, contingent on the reference OLO schedule given in the table.

Schedules other than the one shown may be established by the operations analysis now underway at Vought Astronautics. If a particular vehicle is not required until a later date than shown, the initiation of development of an attitude control system for this vehicle may be delayed.

Assuming the program length is as shown in table 5-1, the price will increase roughly 4% per year of delay because of price index increases and growth in system complexity. Program stretchouts will raise the total cost, since a basic organization must be supported for a longer time. Development of those vehicle attitude control systems which must be operational in 1965 cannot be delayed. The extensive qualification and flight test program required for manned orbital vehicles makes it imperative that system specifications be established by the end of 1962.

TABLE 5-1

SUMMARY OF ATTITUDE CONTROL EQUIPMENT
COSTS FOR ALL OLO VEHICLES IN THOUSANDS OF DOLLARS

Calendar Year / Vehicle	1962	1963	1964	1965	1966	Five Year Total
INDIRECT OPERATING EXPENSES						
Attitude Control Hardware Development Costs						
• Orbital Launch Facility						
Initial	130	500	600	350		1580
Intermediate		110	430	510	300	1350
• Space Vehicle						
Chemical	90	350	420	250		1110
• Miscellaneous						
Unmanned Payload						
Initial	60	220	270	150		700
Intermediate		70	270	330	180	850
• Orbital Support Equipment						
Astrotug		130	340	460	290	1310
TOTAL INDIRECT OPERATING	280	1380	2380	2050	770	690[illegible]

TABLE 5-1 (Cont)

SUMMARY OF ATTITUDE CONTROL EQUIPMENT
COSTS FOR ALL OLO VEHICLES IN THOUSANDS OF DOLLARS

Vehicle	Unit Cost, Thousands of Dollars	Rate of Procurement					Five Year Total Cost
DIRECT OPERATING EXPENSES		1962	1963	1964	1965	1966	Thousands of Dollars
Attitude-Control Hardware Procurement Costs							
• Orbital Launch Facility							
Initial	110			AABBBB	CCCCCCCCCC	DE	2310
Spare Parts				F	F F		
Intermediate	120				A A BBBB	C C	1080
Spare Parts					F		
• Space Vehicle							
Chemical	75			AABBBB	C C [1]	DE	900
Spare Parts				F		F	

Nomenclature:
A = Ground Test
B = System Ground Test
C = Flight Test
D = Manned Flight Test
E = First Operational Use
F = Spare

Notes:
1 Ten tests not required for man-rating because vehicle is modification of existing vehicle
2 No man-rating required

20-40% spare parts costs during initial evaluation tests is an average price

TABLE 5-1 (Cont)

SUMMARY OF ATTITUDE CONTROL EQUIPMENT
COSTS FOR ALL OLO VEHICLES IN THOUSANDS OF DOLLARS

Vehicle	Unit Cost, Thousands of Dollars	Rate of Procurement					Five Year Total Cost
DIRECT OPERATING EXPENSES (Cont)		1962	1963	1964	1965	1966	Thousands of Dollars
• Miscellaneous							
Unmanned Payload	80				A A BBBB F	C C [2]	720
Initial Spare Parts							
Intermediate Spare Parts	100					AABBBB F	700
• Orbital Support Equipment							
Astrotug	100				A A BBBB F	C C [2]	900
Spare Parts							
TOTAL DIRECT OPERATING				1,295	3,570	1,745	6,610

Nomenclature:
A = Ground Test
B = System Ground Test
C = Flight Test
D = Manned Flight Test
E = First Operational Use
F = Spare

Notes:
1 Ten tests not required for man-rating because vehicle is modification of existing vehicle
2 No man-rating required

20-40% spare parts costs during initial evaluation tests is an average price

SECTION VI

PROBLEM AREAS AND RECOMMENDED PROGRAMS

6-1. PROBLEM AREAS

There are two pressing attitude-control problems peculiar to the orbital launch concept. These are

- Measuring and controlling the relative attitude of two vehicles in close proximity
- Attaching the astrotug to a second vehicle in such a manner that stable, efficient operation of the coupled pair is possible. The question of where and who applies what control torque is of particular importance from the standpoint of attitude control.

In addition, many problems not specifically arising from the orbital launch concept must be solved. The rate of the human, control accuracies, and dynamic response are typical example. A broad study of a few of the problems associated with development of attitude control systems for the OLO vehicles was attempted in the preceding sections, but much work remains to be done.

a. The Relative Attitude Control Problem

There are three aspects of the relative attitude control problem: relative attitude of two vehicles must be sensed for control purposes, it must be maintained while

relative velocity impulses are applied, and it must be controlled to the requisite accuracy.

Many methods of sensing relative attitude can be envisioned, particularly when it is recognized that the important control parameter is not vehicle attitude, but rather the displacement of the nose and tail from the desired line of approach. Sensing techniques leading to more or less independent control of nose and tail position seem particularly attractive from the viewpoint of automation. Examples would include measurement of nose or tail displacement through hyperbolic phase comparison techniques, use of semi-independent beam-rider control of both the nose and the tail, or tugging on a rope attached to the vehicle while applying compensating thrusts to the tugged vehicle. Other sensors, such as radar, optical and infra-red devices, and direct visual observation should also be considered.

Maintaining the relative attitude of the two vehicles while correcting their relative velocity (or position) requires that the point on the vehicle whose velocity is to be controlled be defined. Restating the problem, it is necessary to select a pivot point when commanding a change in attitude with no change in position. The desired location of this pivot point will probably vary as the docking maneuver proceeds. It will not generally be at the C.G. In fact, the CG location will change with time and will vary from vehicle to vehicle. Again, some measure of independent control of the nose and tail seems desirable.

The method used to control relative attitude must also be established. Such factors as the degree of automation and the division of command authority between the two vehicles have an important bearing on operational success. As an example, maximum use of the crew is normally desired

for reliability and conservation of equipment weight, space and power. However, relative attitude control of two vehicles in close proximity is a function which seems particularly amenable to automation. Every docking operation will be the same. A high degree of accuracy but little judgment is required to control relative attitude. Finally, manual control would require extremely close attention for extended periods of time.

b. The Coupled Tug-Payload Attitude-Control Problem

Since the tug will probably not be employed in early orbital launch operations, this problem is not considered as urgent as the preceding one and initiation of work in this area may, if necessary, be delayed. Summarizing briefly, aspects of this problem include:

- Optimizing tug location with respect to the second vehicle
- Deciding between rigid and flexible couplers.
- Dividing the thrusting and attitude control torquing functions between the vehicles.
- Investigating the degree to which electrical interconnection of the two vehicles is desirable.
- Establishing the maneuverability, dynamic response, stability, and fuel efficiency of the coupled pair.

c. Attitude Control System Development Problems

Many stability and control areas require study and analysis in order to determine the optimum configurations consistent with maximum reliability and safety, and minimum cost and weight. Design studies must be initiated immediately

if the OLO vehicle attitude control systems are to be operational by 1966. Areas which should be studied as part of the system design study include:

- Establishment of stability and maneuverability requirements for the vehicle
- Investigation of the role of the crew. Primary, back-up, and emergency functions of the crew must be defined and the remaining functions automated
- Selection of firm system configuration
- Selection of sensors, control technique (pulse-width modulated vs on/off), reaction-torque generators, displays, and manual control equipments
- Investigation of vehicle stability and dynamic response to establish gain switching and shaping network requirements
- Allotment of error and unreliability margins to each subsystem and/or component
- Establishment of integration factors such as electrical, hydraulic, and pneumatic power; equipment weight and space; and the form, level, and impedance matching required for all signals flowing between subsystems. It is also necessary to establish which sensors may be common-use items with other systems and to define the interface equipment required for successful integration.

One particular problem worthy of special attention is that of developing bi-propellant attitude-control systems for the intermediate OLO vehicles. Extended performance and environmental testing of a representative bi-propellant

system would lead to increased confidence in system reliability. The state of the bi-propellant attitude control system art should then be comparable with that of cold gas systems, ensuring a trouble free qualification test program on the OLO systems.

6-2. PROGRAM RECOMMENDATIONS

Initiation of the following programs early in CY 1962 appears imperative if the OLO schedule is to be maintained:

a. Relative Attitude Control Study

Tasks comprising this study are

- Comparison and evaluation of alternate methods of sensing relative attitude
- Synthesis of several tentative attitude/vernier velocity control systems and preliminary analytical comparison of them
- Analogue computer simulation of the several most promising systems followed by detailed computer evaluation. Manual control effectiveness would be investigated as part of this phase
- Preparation of system specification, including sensor selection, definition of a control technique, and component requirements.

b. Attitude Control System Development

This task would include the following items for each vehicle control system considered:

- Investigation of the role of the man
- Selection of system configurations, sensors, attitude control torquers and control techniques.
- Analogue computer investigation of vehicle/control system dynamics
- Establishment of component specifications and interface requirements.

It is recommended that such a program be undertaken for development of control systems for the initial OLF, the Orbital Launch Vehicle, and the Unmanned Payload, in the near future. If at all possible, work should also be initiated on the storable bi-propellant development program and the coupled tug-payload attitude control study. Development of attitude controls for other vehicles can be delayed if absolutely necessary.

Finally, a discussion of a proposed reliability program, essential to the success of the recommended studies, has been included in the following section.

6-3. RELIABILITY PROGRAM

It is recognized that reliability is an indispensable requirement for successful Orbital Launch Operations. The only way to achieve high reliability is through an intensive, reliability-oriented design effort. This requires a developmental program in which

- Reliability is considered a primary design objective

  All personnel involved with the development program must be made aware that management has

established reliability as a major goal and is willing to pay the price required to reach this goal.

- Reliability must be treated as a quantitative design parameter
  Every set of alternate configurations or components should be compared on the basis of reliability. Every proposed system modification aimed at improved performance should be evaluated in terms of its effect on system reliability. Quantitative trade-offs are to be the basis for design decisions whenever possible.

- Reliability improving design techniques should be utilized whenever possible
  Several of the most important techniques to be used include: the use of standard, field-proven components, component derating, careful attention to packaging design in order to assure an environment suitable for the selected components, and a conscious striving for design simplicity.

- Reliability Control must be exerted positively through a Reliability-oriented design review.
  Every drawing is to be reviewed and given:

  A functional check - will it work?

  A component application check - are the components suitable for the application? Derating, environmental limits, and tolerance acceptability as to be emphasized.

  A packaging check - are the components mounted and located in such a manner that they will withstand the expected environment? Vibration

resistance and ability to dissipate heat must be given special attention.

A failure loading check - does failure of any one component impose undue loads on the rest of the system? Can the failure of a component be made non-critical so that continued degraded operation is possible?

This engineering reliability effort is to be followed by a thorough quality assurance program. A particular effort should be made to maintain very close liaison between Quality Control and Engineering in the early phases of the development cycle in order that Quality Control can inform Engineering of reliability-critical areas, and in order that Quality Control may become aware of potential difficulties.

VOLUME V

ORBITAL LAUNCH OPERATIONS

PART III

SECONDARY POWER

TABLE OF CONTENTS

	Page
1.0 Summary	3-1
2.0 Introduction	3-5
3.0 Initial Considerations	3-7
4.0 Description of Recommended System (Chemical-Dynamic)	3-9
4.1 Candidate System	3-9
4.2 Assumptions	3-16
4.3 Parametric Studies	3-17
4.4 System Parameter Sensitivity	3-18
4.5 OLV Auxiliary Power Unit	3-19
4.6 Alternate System for OLV	3-24
4.7 Astrotug APS	3-27
4.8 Incoming Payload APS	3-28
4.9 OLF Secondary Power	3-28B
5.0 Cost and Schedule	3-30
6.0 Problem Areas	3-33
Appendix A Supplementary Systems	
Appendix B Definitions	

VOLUME III - SECONDARY (AUXILIARY) POWER

1.0 - SUMMARY

The Sperry Rand Corporation is supporting Marshall's Orbital Launch Operations Study (Vought Astronautics Contract NAS 8-853) by investigating the major Astrionic requirements for the program. The subject of this volume is Secondary (Auxiliary) Power Systems (APS), which covers the work accomplished by Vickers of Sperry Rand.

The results of studies conducted by Sperry Rand verify selection of the Snap 8 nuclear power plant for the OLF concurring with Vought and also establishes the advantages of using hydrogen-oxygen engines as an auxiliary power system for the OLF standby system as well as being the primary source of electrical energy for the OLV and the Astrotug. This cryogenic expansion engine has proved reliability through nearly 200 hours of testing. It has been designed and qualified expressly for space vehicle application where power is required ranging from two to thirty kilowatts with continuous operational time up to one week, and intermittent duty cycles up to one month (see Figure 3-1).

The H_2-0_2 chemical system recommended for the OLF, incoming payloads, Astrotug, and as a standby system for the nuclear powered OLF was selected on the basis of power-to-time requirements and its essentially 100 percent reliability, light weight, and low cost. Trade-off studies on the cryogenic auxiliary power versus fuel cells and solar cells have shown considerable substantiation of this choice.

Early models of the test engine achieved adiabatic efficiencies as high as 75 percent at operating pressure ratios ranging between 20 to 1 and 50 to 1. The brake specific propellant consumption (BSPC) achieved during these early tests was 1.3 pounds per horsepower-hour. From

further studies to be conducted in hydrogen-oxygen engines, efficiencies will approach 85 to 90 percent by 1964. Vickers has established this hydrogen-oxygen engine as an optimum auxiliary power unit for several applications in the orbital launch operation program.

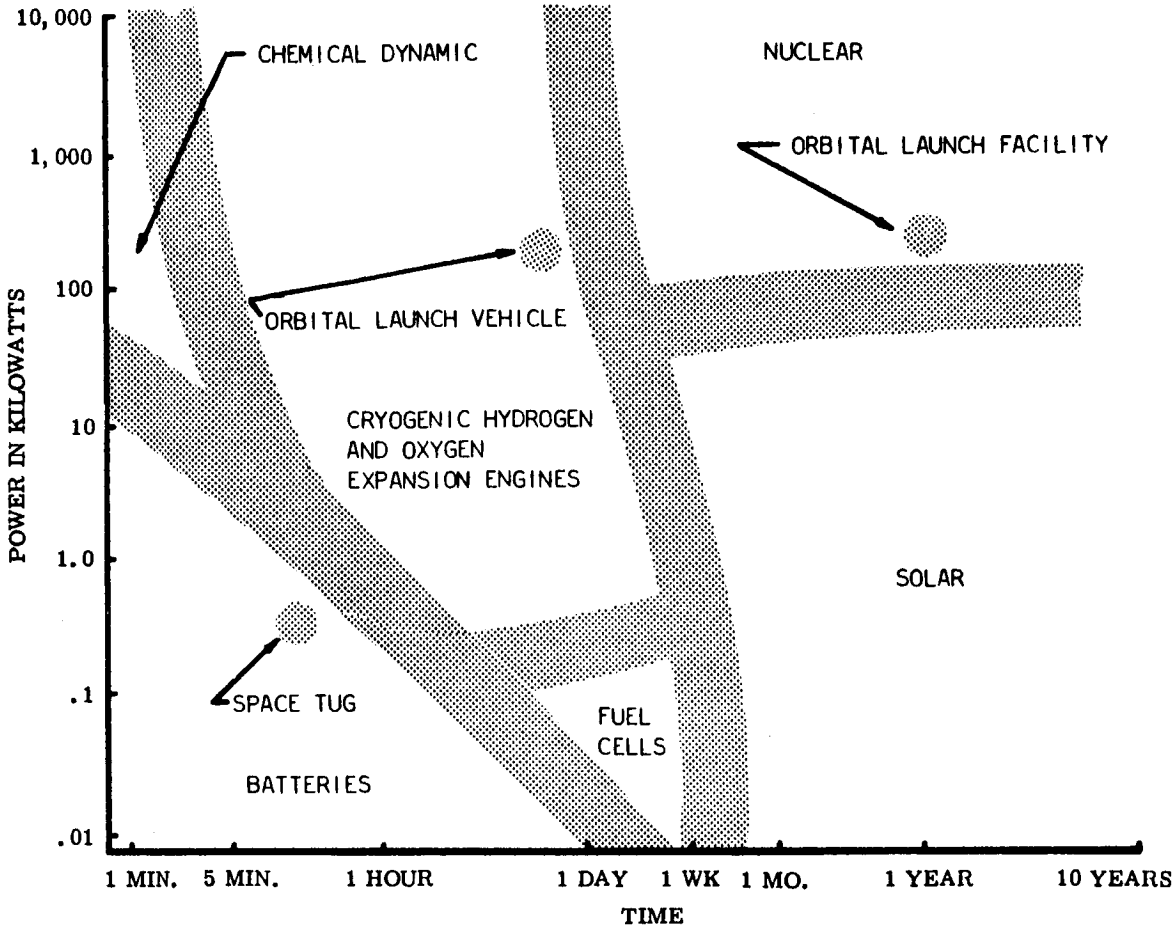
The principal conclusions of this work are:

1. The auxiliary power aspects of the OLO program are within the time scale for state of the art development as conceived by Sperry Rand. No major technological breakthroughs are necessary to successfully establish OLO operations.
2. Certain follow-on study work is absolutely necessary to firmly establish plans for successful OLO operations. Since this work is within Vickers' capabilities, it is requested that assignment of it to Sperry Rand for the next funding period be considered. In the area of auxiliary power, this work includes:
   a) Establish the design parameters necessary for OLO operations so that system design can be initiated.
   b) Specifications for APS and life-science support must be prepared outlining complete requirements for the overall OLO program.
   c) A more penetrating study of the OLF, OLV, Astrotug, and incoming payload requirements must be made so that maximum performance and reliability can be achieved.

3. Certain follow-on work related to the OLF also appears necessary. This work is described as follows:

a) The shielding requirements for the Snap 8 power system appear to have a considerable influence on the overall configuration of the OLF. A more penetrating study of the shielding requirement and its effect on the arrangement and operation of the OLF is recommended.

b) The development of the runner-up OLF power plant utilizing solar energy is progressing at a fairly rapid pace. This progress should be continuously monitored to take advantage of any major new developments.


10,000
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POWER IN KILOWATTS
CHEMICAL DYNAMIC
NUCLEAR
ORBITAL LAUNCH FACILITY
ORBITAL LAUNCH VEHICLE
CRYOGENIC HYDROGEN
AND OXYGEN
EXPANSION ENGINES
SOLAR
SPACE TUG
FUEL
CELLS
BATTERIES
1 MIN.
5 MIN.
1 HOUR
1 DAY
1 WK
1 MO.
1 YEAR
10 YEARS
TIME

Fig. 3-1

2.0 - INTRODUCTION

The purpose of this report is to present the results of investigations conducted in the fields of auxiliary power systems. The Sperry Rand Corporation is conducting an across-the-board analysis of the various phases of Orbital Launch Astrionics to aid the Marshall Space Flight Center in developing a firm and well-documented plan for establishing the highly advantageous technique of Orbital Launch. The work is being conducted in connection with Vought Contract NAS 8-853, and is being guided by Vought work statements and coordinating documents. Every advantage is being taken of the results obtained by Vought in the earlier phases of the contract.

This phase of the Astrionic work is being funded by Sperry Rand as a contribution to establish the key engineering programs necessary to achieve success in this important endeavor.

Rapid technological advances in aircraft, missile, and orbital space vehicle programs have demanded concurrent progress in auxiliary power system (APS) concepts. Vickers Incorporated has assumed the responsibility of advancing this art by developing "packaged power" systems.

Space vehicle applications for auxiliary power systems dictate complete and independent power generating systems to replace the shaft power takeoff concept normally used on vehicles with reciprocating or turbine engines.

Since the forms of power required by space vehicles are similar

to those used by conventional vehicles (hydraulic, electrical, etc.), the auxiliary power systems evolved by Vickers are natural extensions of past product lines. Although many of these APS concepts are an integrated part of the prime mover power system, Vickers has also developed systems that are self-sufficient and extract primary power from an independent source.

Study the relative power requirements of various systems for the OLO program, the power required for each application and the possible need for multiple systems were primary factors considered. The studies investigated auxiliary power requirements aboard the Orbital Launch Facility (OLF), the Spacecraft (S/C), incoming payloads and Astrotug.

It became apparent early in the investigation of power plant requirements for the OLF, that the results of the Sperry analysis were closely following those obtained by Vought. The selection of the Snap 8 power plant system was confirmed on the basis of mission requirements, reliability, availability, size and weight. This is discussed further in Section 4.9 of this report. For standby power requirements for this facility and for power supplies for the other orbital vehicles, however, considerable further detailed analysis appears to be required.

3.0 - INITIAL CONSIDERATIONS

Orbital and Space-Flight Vehicles - The requirements for the on-board OLO power systems were taken from Vought Astro Design Data, Vol. II. (Power parameters were given for the four vehicles and are shown in table 3-1.)

The auxiliary power system is used primarily for attitude controls and electrical power, but will also offer a general power source. In applications where long-term nuclear power is desirable, an APS can be utilized as a standby power system.

Table 3-1 is a breakout of the vehicle power requirements for the guidance and control and overall electrical power requirements. Later in the report the discussion will be illustrated to show how the selected auxiliary power system will apply directly to these three vehicles.

TABLE 3-1 VEHICLES AND POWER SCHEDULE

VEHICLE	POWER SYSTEM SCHEDULE		
	YEAR	POWER REQUIRED	
O. L. F	1968	30	KW
	1970-1975	60	KW
SPACECRAFT	1964-1966	1.5	KW
	1966-1969	2.5	KW
	1969-1975	3.5	KW
	3 MAN SPACECRAFT		
	Guidance	2.3	KW/20min
	Misc. Power	1.7	KW/120 Hr.
	Lunar Rest	1.4	KW/x days
	Final Phase	1.6	KW/1 Hr.
	6 MAN SPACECRAFT		
	Launch	3.2	KW
	Mid Course	2.8	KW
	Lunar Land	3.2	KW
	Lunar Rest	2.7	KW
	Lunar Launch	3.2	KW
	Mid Course	2.8	KW
	Earth-Guidance	2.5	KW
ASTRO TUG	1968-1975	1.80	KW
	G. & C.	.45	KW
	Lighting	.10	KW
	Envr. & Inst.	.60	KW
	App. Total Power (2 hour work period)	1.2	KW
INCOMING PAYLOAD	1968 (Estimated)	1.10	KW
	G. & C. - Inst.	.2	KW
	Attitude Control	.9	KW
INTERPLANETARY TRAVEL	1971 - 19xx	300	KW

4.0 DESCRIPTION OF RECOMMENDED SYSTEM

Studies conducted by Vickers revealed that a dynamic hydrogen-oxygen auxiliary power system is capable of achieving performance equal to or better than that achieved with fuel cells. In addition, the chemical dynamic system can provide two additional desirable functions:

(1) operate on low pressure boil-off gases from the main propulsion system tanks,

(2) provide drinking water for the crew.

4.1 - Candidate System

A typical H_2-0_2 APS concept using the Vickers cryogenic engine is shown schematically in figure 3-1. In this system, an electrical alternator and a hydraulic pump are mounted on the H_2-0_2 engine, giving both mechanical and electrical power. Use of a through shaft in the alternator eliminates any need for gearboxes.

The hydraulic pump not only powers the vehicle servo system and positioning of solar devices but also operates in a bootstrap manner to power the hydrogen and oxygen propellant pumps.

The hydraulic fluid is cooled by circulating through a heat exchanger in the hydrogen line that leads to the engine combustor. The alternator is cooled by circulation around the alternator jacket and through a similar in-line heat exchanger. The hydrogen fuel also cools the electrical equipment compartment by absorbing heat from a nitrogen environmental control system.

The combustion supply pressures of hydrogen and oxygen are independently controlled by pressure regulating valves fed by reservoir pressures.

Since the hydrogen and oxygen propellants offer inherent engine system cooling, it is unnecessary to equip the engine with water jackets. Moreover, the engine exhaust is pure steam that can be condensed for crew drinking water. Water-payload penalties are thereby eliminated using the H_2-0_2 propellant concept.

Recognizing the importance of low propellant consumption and high reliability in power generating systems for manned orbital space vehicles, Vickers has instituted a research and development program demonstrating the feasibility of a positive displacement power plant for operation on propellant boil-off gases from the main propulsion system. A prototype illustration of this engine is shown in figure 3-2.

Advantages of such a power plant over fuel cells include:

1. Lower installed weight per kilowatt-hour.
2. Growth potential: lower installed weight per kilowatt-hour, lower brake specific propellant consumption (BSPC), and up to 1000-hour mission duration.
3. Fewer problems with zero-gravity conditions.
4. Capability of achieving higher thermal efficiency.
5. Early availability as a result of using proven concepts and hardware.
6. Low development cost due to parallel R & D programs now in progress.

The advantages of the H_2-0_2 system over conventional solar cells are:

1. The solar cell power system is presently much heavier than the combustion APU and it appears that by 1965 there will still

be a considerable weight advantage to the combustion unit.

2. Operational reliability of a solar array requires continuous orientation to the sun during the flight.
3. Protection from shock during the lunar landing and take-off operations is required. In particular, there is a real risk that the solar array could be damaged by a hard landing on the Moon's unknown surface.
4. Operational flexibility of the combustion APU is much greater because it is capable of operating in the absence of sunlight. For scientific purposes it permits investigation of lunar conditions during the lunar nighttime as well as daytime. Sunless operational capability provides an extra safety factor in case the OLV lands in deep dust, or the shadow of a tall mountain, or some other malfunction strands a crew on the return launch from the moon. Since the latter occurrence would probably take place toward the end of a daytime period the crew would face a problem in generating enough auxiliary power to maintain normal functioning of the vehicle internal environment and to communicate with a rescue vehicle. If the return launch rocket were also an H_2 0_2 system the crew could use the OLV fuel supply for survival. A rescue vehicle for this operation would probably operate during lunar nightime if the orginal exploration were a daytime expedition. Therefore

a combustion APU is mandatory! There is a measureable risk of failure to take-off from the moon and international prestige and political considerations make a successful rescue attempt very desireable as an illustration of comprehensive capability in the space field.

5. The solar cell system contains other difficulties which are probably solvable but still present considerable difficulties. Some of these problems are erection of large solar arrays, destructive effects of meteorites, and deteriorating effects of solar flares on solar cells.

Studies have been made on a regenetative power system concept, indicating the capability of achieving a BSPC of 0.85 lbs/hp-hr. Rapid availability of such a system is assured by utilizing proven thermodynamic concepts and power cycles differing from conventional power plants only

in the choice of fuel and oxidizer. Operation at conservative conditions of temperature, pressure, and speed further assure rapid availability of reliable components and systems.

A program is proposed which includes analysis and optimization studies, feasibility and performance testing of critical components, and similar testing of a complete "breadboard" auxiliary power system for an Orbital Launch Program.

Vickers is currently conducting a research and development program for Aeronautical Systems Division, Wright-Patterson Air Force Base, on a hydrogen-oxygen internal-combustion positive-displacement engine. An engine of the same or slightly modified design can be used in the OLO program. This current Air Force-sponsored program will also augment the OLO program in other major areas, including cylinder heat rejection, piston friction minimization, practical oxygen injection, and combustion properties of hydrogen and oxygen in internal combustion engines.

Plans are now in progress to develop a complete secondary power system that will provide environmental requirements for the crew on the OLO vehicles. By utilizing the steam exhaust and available nitrogen that would be required for cooling systems for electronic equipment, it would be possible to develop a temperature, pressure, and humidity control, and a breathing air source for the crew members. An integrated H_2-O_2-N_2 system could support the life sciences by providing both water and a source of ecological needs of the men. This type system would save many

pounds of overall weight if it were designed into a vehicle requiring approximately one month operational time.

By combining with the nitrogen the three natural atmospheric gases required to sustain life can be obtained. This system will not only provide the auxiliary power for any electrical requirements, but will provide the drinking water and a source of moisture for cabin humidity control. The superheated exhaust can provide a temperature control through a heat exchange process. The auxiliary power system can be utilized in a third manner; it could also provide an environmental pressure control because it can drive an air pressure pump for the cabins and crew living quarters. Basically, the H_2-O_2-N_2 system has the simplicity and coherent design that would be required for four other comparable systems. For these reasons, Vickers believes it worthwhile to recommend the system for the Vought OLO program.

1. Electrical power output - 2 kw
2. Generator efficiency - 90 percent
3. Hydrogen and oxygen both stored at 30 psia
4. Both hydrogen and oxygen available in required quantities as boil-off gases.

A regenerative-type power system has been analyzed for basic feasibility and is presented in the following paragraphs. It has been analyzed as though no heat sink would be available in space other than the incoming propellants. To make a fair comparison between fuel cells and

expansion engines, it is necessary to study a complete spectrum of systems operating with oxygen/hydrogen ratios between 2.0 and 8.0 (the latter ratio representing the oxygen/hydrogen ratio of a fuel cell). It may well be that there is an optimum combination of heat sinks and oxygen/hydrogen ratios which result in the most suitable power plant system. It is important to note that reciprocating power plant systems can be designed to operate at oxygen/hydrogen ratios of 8:1 with efficiencies of 60 percent or better; the resulting fuel consumption should be directly competitive with practical fuel cells and progressively competitive with solar cells and silver zinc batteries.

Even though an objective comparison between solar cells for 28-day operation and dynamic power plants is difficult without a detailed study, it is revealing to compare typical examples of these two types of systems. In general, weight and bulk of solar cells are very high. They are susceptible to meteoritic bombardment, cosmic radiation and are difficult to handle.

On the other hand, most chemical dynamic systems must employ methods of dissipating the exhaust. Condensors must separate the liquid from the gas under zero gravity conditions.

This problem is inherent in the tankage system. However, by using a bladder separating the working fluid from the inert gas it is possible to separate the cryogenic liquids from the pressurized gas. In general, the weight requirements for a hydrogen, oxygen system during a 28-day mission would be considerably less than solar cells. It must be kept in mind that

this system is proposed with the overall viewpoint of supporting life sciences. Table 3-2 shows the candidate systems for the four vehicles.

TABLE 3-2 AUXILIARY POWER SYSTEMS FOR THE OLO VEHICLES

VEHICLE	RECOMMENDED AUXILIARY POWER SYSTEM
OLF	SNAP VIII STANDBY H_2-O_2 SYSTEM
SPACECRAFT	H_2-O_2 APS MANNED VEHICLE
ASTRO TUG	H_2-O_2 APS MANNED VEHICLE
INCOMING PAYLOAD	1. HOT GAS APS 1 kw for 1 hour 2. H_2-O_2 APS

4.2 Assumptions

Although only a limited amount of information is available to formulate the bases for preliminary studies of possible auxiliary power systems, the following assumptions have been made to clarify the problem statement:

1. The auxiliary power system is designed as a self-contained unit.
2. Reliability is achieved by the use of redundant systems which weight and efficiency calculations take into consideration.
3. A high degree of mechanical design sophistication is considered permissible.

4. A progression of the state-of-the-art will result from any further study.

5. The influence of solar heating (which has been omitted from this preliminary study) can be made available.

6. Propellant tanks and motor lubrication systems will be designed to operate in a zero-g environment.

7. APU operation may be monitored by personnel and/or remote servo systems, and the power units may be completely regulated during flight.

8. No special "overload" components are required and power over the full range is supplied by the single system.

4.3 - Parametric Studies

During the research and development phase of the internal combustion engine program, a continuing study is being made of the important parameters in an effort to minimize development time.

1. Weight and Volume - Materials, packing techniques, and hardware design will be investigated for minimum weight and space requirements.

2. Power Requirements - Sufficient growth potential will be incorporated into all candidate systems to allow for future growth. The most effective means of insuring the desired growth potential will be studied.

3. Availability and Development - For an internal combustion positive-displacement engine to be competitive, it must be able to meet any required availability dates. Proposing two engines of different levels of development allows considerable flexibility in design schedules and ensures that the power system will remain competitive for many years.

4. Reliability - The low fixed weight and stable construction materials inherent in an internal combustion positive displacement APU give it an advantage over radiation and temperature sensitive competitors such as fuel cells and batteries.

5. Maintainability - Maintenance and servicing capability will be considered as basic design goals.

4.4 System Parameter Sensitivity

The basic design parameters may be changed or modified as the problem becomes better defined and more information becomes available. The sensitivity of the basic APU to any change in requirement is a major factor to consider in the selection of an auxiliary power system. A discussion of the effects of requirement and state-of-the-art changes on the H_2-O_2 I. C. engine APU follows:

1. Weight and Volume - The weight of the system is extremely flexible, and a weight reduction can always be effected at the expense of some design complexity. Volume requirements are as low as any competitive system and the promise of future efficiency improvements can lower fuel weight and volume.

2. State-of-the-art Development - A major breakthrough in the fields of batteries, fuel cells, and other direct conversion devices will be necessary before the relative competitive position of the positive displacement internal combustion engine is changed. It is not possible to predict such a breakthrough or to anticipate its results although continuing refinements in all systems are anticipated, the Hydrox Internal Combustion engine is expected to remain a major candidate.

4.5 S/C Auxiliary Power Unit Description

The requirements for an auxiliary power unit for the 3-man Spacecraft are given in table 3-1. Enough growth potential is incorporated to permit the use of this system in a 6-man vehicle with minor modifications. Two engine design levels are discussed: (1) a simple engine that will be available in 1963 and (2) a more advanced model, benefiting from stoichiometric combustion, that will be available in 1965. Both systems are compared with a fuel cell system and with a solar cell system. All energy requirements are specified except for the time spent at rest on the surface of the moon. To allow flexibility, propellant and tankage weights are expressed as a function of the lunar rest time. The following equation gives electrical energy available from the system.

gives electrical energy available from the system.

$$E = 247.3 + 33.6t \quad \text{kilowatt-hours,}$$

where t = number of days at lunar rest

Graphically, this equation can illustrate the electrical energy and weights as a function of lunar rest time in days as shown in figures 3-3 and 3-4. The

support calculations for these weights are shown in the systems breakout chart (table 3-3).

TABLE 3-3 OLV SYSTEMS CALCULATIONS

	Advanced I.C. engine	Simple I.C. engine	Fuel Cell
Oxygen	210+28.4t	170+23.2t	216+29.3t
Oxygen tank	42+5.7t	34+4.64t	43.3+5.9t
Hydrogen	26.2+3.6t	113.5+15.5t	27.0+3.7t
Hydrogen tank	21.0+2.8t	90.8+12.4t	21.7+2.9t
Total Propellant	299.2+40.5t lbs	408.3+55.74t lbs	308.0+41.8t lbs

Adding the above weights to the engine hardware,

Power Units (3)	24	18	216
Alternators	45	45	---
Pumps	3	3	3
Plumbing	10	10	10
Total	381.2+40.5t lbs	484.3+55.74t lbs	537.0+41.8t lbs

t = number of days spent at lunar rest.

Solar cell weight is composed of:

Cells and structure	280 lb
Glass	245 lb
Storage	325 lb
	850 lb

As is shown in figure 3-4, for lunar stay times of less than one week, the simple H_2O_2 engine and fuel cells are comparable to each other with solar cells gaining the ascendancy for periods from one to two weeks. At the end of two weeks max, the solar cell becomes inoperative for the two week lunar night.

The advanced H_2O_2 engine is superior to all three above energy sources from the weight standpoint for any length of lunar stay.

The solar cell weight given in figure 3-4 has assumed a 30% decrease in weight over 1962 state-of-the-art solar cell development. Estimates for weight decrease run as high as 55% by 1965. The 30% decrease is considered a reasonable reduction factor for this report. The battery weight given is not sufficient to maintain the required KW output for earth orbital operations but is adequate for temporary shut downs required for mid-course and lunar terminal maneuvers.

Efficiency - A possible brake specific propellant consumption of 0.60 lb/hp-hr is assumed for the most efficient operating level of the advanced internal combustion hydrogen-oxygen engine. This number is based on a stoichiometric combustion ratio (O_2/H_2 = 8/1). For the simple internal com-

bustion engine, a BSPC of 0.74 lb/hp-hr at a mixture ratio of O_2/H_2 = 1.5/1 is assumed. The minimum BSPC of the fuel cell system is 0.65 lb/hp-hr at stoichiometric mixing. In figure 3-5, the off-design point efficiency of these three systems is shown. It is seen that overload operation on one engine will raise the BSPC of the advanced and simple IC engines to 0.63 lb/hp-hr and 0.77 lb/hp-hr, respectively. No correction was made for fuel cell failure since fuel cells were assumed to operate singly at the optimum load.

The overall thermal efficiency of the H_2-O_2 system is over 70 percent which is considerably higher than fuel cell efficiency.

An electrical conversion efficiency of 90 percent was assumed for all systems. This represents an alternator efficiency in the reciprocating engine systems and a DC-AC inverter efficiency for the fuel cell.

Tankage - The total propellant weight was calculated from the energy requirement given by the equation for electrical energy, using the applicable BSPC. Tankage weight was then calculated by using the following assumptions:

Oxygen tankage weight = 20% oxygen weight

Hydrogen tankage weight = 80% hydrogen weight

Tank weight estimates are based on the results of Vickers studies on advanced concepts in zero-g cryogenic tankage for space missions. See reference 6 for a discussion of the rotating fluid tank concept.

Motor Weight - Motor weight was calculated for the two reciprocating engine systems as follows:

Motor BMEP of 200 psi at maximum rpm was assumed. The greatest power output necessary is 3.6 hp (one motor, generating 2.4 kw @ 90% efficiency). If maximum engine speed is 6000 rpm for this extreme condition, an engine displacement of 1.2 $in.^3$ results.

Extrapolation from the weights of small 1-cylinder industrial engines in this displacement range yields a weight of 7.6 pounds. A conservative assumption of 8 lbs/motor was used. This weight includes engine speed controls and accessories.

OLV Reliability Discussion - Manned space flight demands a high level of reliability that can be achieved by the use of redundant systems. The following basic rules, taken from reference 2, were used to determine the necessary redundancy:

1. An orbital launch vehicle reliability of 0.85 requires an auxiliary power system reliability greater than 0.9999.

2. Separate, complete, and fully installed power units are desirable. The number of units depends on the reliability of each.

Well documented reliability figures for mechanical systems operating in a space environment will not be available for years. If a single unit reliability of 0.95 can be attained by the proposed internal combustion hydrogen-oxygen engine, the system reliability for three parallel units is .999875. This number represents the probability that all three units will not fail during the mission. In actual practice, two units will be operating under reduced load at all times, thus increasing the dependability of an individual unit. Also, the unit (or units) not operating will not deteriorate

with time for the periods considered in this mission. It is believed that these considerations will permit a three-engine APU to attain the desired reliability.

The one-engine reliability figure of 0.95 is an estimate of turbine reliability under space conditions. This reliability may be easily attained or surpassed by a small reciprocating engine since the experience of seventy years of design technology may be brought to bear on all problems which arise. The reliability of the engines used in the OLV APU depends primarily on effectively incorporating the results of the development effort and environmental testing effort into the final design.

An estimated fuel cell system reliability of 0.96 is estimated in the literature. Three parallel systems are needed to surpass the required reliability level. Due to the high fuel cell system weight, this requirement imposes a more severe penalty on the fuel cell APU than it does on the internal combustion engine APU.

Maintainability of OLV and Astro Tug - When human beings are available in a space system a great deal of complexity and system redundancy may be eliminated by effectively incorporating them into the system. Conversely, the elaborate environmental controls needed to sustain human life can be made more efficient if waste heat and by products are effectively re-cycled.

There is no reason why a standby power system for the orbital launch facility could not be designed to be repaired, serviced, and dismantled by personnel on board. Spare parts and required tools could be made available

to the facility.

It would be possible to relax the reliability requirements of the assembled unit if it could be easily serviced and overhauled by men of moderate skill working under adverse conditions. The success of non-professional automobile mechanics over the last fifty years attests to the maintainability of an internal combustion engine under these conditions (a contention that cannot be held for fuel cells or turbines).

Space tug APU's could also be made repairable and easily replaceable or exchangeable. The weight savings offered by an internal combustion engine system over a battery system for long operative periods depends on reducing the fuel weight, spare parts, extra motors, and tool weights for the internal combustion engine to less than the battery weight plus the weight of fuel used to charge the batteries.

Human decision-making with respect to instrumentation could also eliminate complexity. All machine decision-making is inherently wasteful and excessively redundant since it must be reduced to go-no-go situations, precluding judgment.

A discussion of alternate systems in which human and sealed-cabin waste heat is effectively re-cycled into the system is presented in the following section. It must be remembered that a fuel cell or battery system cannot make the effective use of this waste heat that is possible with a cryogenic expansion engine.

4.6 - Alternate Systems for OLV

The following alternate approaches to the problem of space electrical

power generation are being considered by Vickers. The feasibility of each approach cannot be determined by the information presently available on system requirements and space operating conditions. Each approach may prove valuable as a means of effectively integrating the APU into the overall vehicle.

Alternate System No. 1 - Main Tank Propellant Boiloff on OLV - An auxiliary power system which uses boiloff gases from the main propellant tanks provides weight reduction in the following areas:

1. Total tankage
2. Total hydrogen under storage
3. Main hydrogen tank insulation (since a higher boiloff rate is permissible)

A complete system design incorporating the use of boil-off gases would require detailed knowledge of the entire vehicle heat balance and the trade-offs between insulation weight and propellant weight. Even if boil-off rates are not a critical problem it may be desirable to utilize the main tanks in the interests of efficiency.

An auxiliary power system weight comparison considering fixed weight alone is shown below. This weight, taken from the OLV systems calculation, is independent of mission duration and illustrates the low fixed-weight advantage of internal combustion engine system.

Advanced IC engine	82 lbs
Simple IC engine	76 lbs
Fuel Cell	229 lbs

Alternate System No. 2 - Personnel Cabin Heat Exchange on OLV - Vehicle cooling was not included in the OLV system description because available heat from the crew compartment and equipment has not been clearly defined. For an accurate study of an integrated system, the following values should be specified.

1. Heat rejection of crew to the vehicle.
2. Heat available to the APU from the vehicle.
3. APU output power which is returnable to the APU through an APU-crew cabin feedback loop.

These values are combined into an overall heat feedback ratio which is usually represented as a percentage of APU output power. Heat feedback can be removed by radiation or by use of the APU hydrogen supply as a heat sink (or both), depending on mission duration, available feedback, and mission requirements.

For most missions that Vickers has studied in the past, the heat capacity of the hydrogen supply for an APU similar to the one described is usually less than the total available heat feedback. Therefore, a radiator may be necessary for vehicle cooling. Another method is heat feedback to the main propellant tanks (particularly if boil-off is being used). A drawing of an APU incorporating heat feedback is shown in figure 3-6.

Alternate System No. 3 - The Use of Auxiliary Power System Exhaust as a Water Source - Water in a manned spacecraft is needed for two applications:

1. Crew drinking water
2. Re-entry cooling

Considerable water will be recovered from crew waste. If more is needed, condensation of the APU exhaust gases may reduce vehicle weight. Figure 3-7 shows the internal combustion regenerative power system in schematic form. Using the present Vickers research engine in this system, approximately 75% of the vehicle exhaust water could be recovered.

Alternate System No. 4 - Standby Power - Orbital Launch Facility - Because the reliability in space is in a state of indecision, the feasibility of a chemical standby power system for the nuclear APU of the 36-man orbital launch facility cannot yet be determined. Assuming that this concept is desirable, some representative system weights have been calculated and are presented below for a system of 48-hour duration. As a comparison, a nuclear APU would weigh approximately 20,000 pounds (two units).

Oxygen fuel and tankage	2200 lbs
Hydrogen fuel and tankage	2200 lbs
Engine weight (60 in.^{3} displacement, 90 hp)	80 lbs
Alternator weight	120 lbs
Plumbing	25 lbs
Total	4625 lbs

A similar engine using storable bipropellants may be necessary if long-term availability is necessary (1-5 years storage). Some initial fuel weight would be sacrificed for storability.

4.7 Astro Tug APS - Two alternate auxiliary power systems for the space tug are compared below. A small cryogenic H_2-O_2 internal combustion system with two engines and fuel for two round trips is compared to a moderate-

discharge-rate secondary battery system using two silver-zinc battery units. It is believed that either system will satisfy the power and reliability requirements for the space tug auxiliary power system.

	Battery	Reciprocating Engine
Number of redundant systems	2	2
Conversion efficiency	90%	90%
Electrical power level	1.2kw	1.2kw
Duration	2 hrs	2 hrs
Fuel weight	-----	6 lbs
Tankage weight	-----	5.7 lbs
Fixed weight	100 lbs	32 lbs
Total Weight	100 lbs	43.7 lbs

The power requirements of the Astro Tug were shown in figure 9-5 of the Vought Design Data, Vol. II. A two-hour work duration is required with a six-hour battery charge. If a hydrogen-oxygen system were used it would offer a substantial decrease in charge time. The six hours would be reduced to a matter of minutes to refuel the six pounds of fuel weight required for vehicle operation. Slightly more than this would be required for a manned vehicle.

Quick-disconnect couplings, automatically operated on the front of the Astro Tug and mounted externally on the platform, would allow an exchange of required fuel of hydrogen and oxygen.

4.8 - Incoming Payload APS - According to recent information on the OLO program there is a distinction between retrieval and rendezvous of pay-

load. The retrieval phase of an incoming payload operation will take place within 500 feet of the platform or a designated position. However, rendezvous will be considered as the long range operation beginning at earth launch and ending at the retrieval phase.

On the basis of this we have calculated the requirements of an auxiliary power system to provide one kilowatt of power for attitude controls and generator power for electric or electronic equipment and related components. It is estimated that this one kilowatt is approximately 800 watts of electric power and 200 watts of mechanical power or hydraulic power. The trade-off studies for attitude control show that 14 nozzles would be required for complete directional and attitude control for the incoming payload. With the high reliability only 200 watts will be required for this nozzle system. The remaining 800 watts will be used for electronic equipment power.

As this is only a one-shot affair, it would be possible to propse either of two systems. The first is a hot gas system whose weight is extremely low and which would offer a required power for one hour's time. Secondly, if the payload were manned, it would be possible to incorporate a hydrogen-oxygen engine.

4.9 OLF SECONDARY POWER

Basis of Consideration

To provide a base line with which to measure the performance and weight of various secondary power systems, 30 KW and 60 KW power output levels have been assumed. These values are convenient in that the Snap 8 nuclear power plant is being specifically designed for the above two power outputs. Other power sources can readily be scaled to these values whereas modifying the Snap 8 outputs would require extensive investigation into the basic design of the Snap 8 with attendant confusion.

4.9.1 Snap 8

The Snap 8 power supply is under development by the AEC and NASA. This power plant for 30 KW and 60 KW operation weighs 1500 lbs and 2500 lbs, not including the shielding required for personnel protection, and employs 7.0 kg fuel to produce 30 KW with a probable life of one year. The shielding required varies with the distance between the OLF and the Snap 8 supply, the amount of time crew members will remain in orbit, and which sets of figures are to be believed as being the maximum radiation dose which humans can withstand and not suffer severe permanent damage. This shielding estimate varies from 10,000 lb for spot shielding to 36,000 lb for hemispheric shielding. For purposes of comparison with other APU's, the more conservative weight of 36,000 lb for the total Snap 8 weight will be assumed.

This shielding is an appreciable fraction of the total mass of the orbital launch facility. Because the crew will spend a considerable porportion of their time outside the facility, adequate protection from the nuclear reactor can be obtained only if most of the reactor shielding is located around the powerplant, which is separated from the facility, rather than around the facility itself. Unfortunately, this precludes employment of the reactor shielding in double service; as a shield against both radiation from the nuclear reactor and radiation from the solar bursts. Reactor shielding weight does not serve any additional purpose that mitigates its undesirability. Therefore, trade-off studies would give great advantage to a very small shielding weight.

Operation of the Orbital Launch Facility requires regular rendezvous with supply vehicles carrying both men and equipment. The close proximity of the nuclear powerplant to the facility (approximately 50 feet) and the dependence upon maintaining this distance from the reactor creates a hazard during the final stages of rendezvous. A larger separation of the reactor from the facility is desirable to simplify the rendezvous problem.

Future orbital space flights may show that man cannot function effectively in a weightless environment for extended periods of time greater than approximately several days. This circumstance would require the generation of artificial gravity by rotating the facility to generate a centrifugal force. Other experiments on men have shown that prolonged rotation at rates greater than 1.7 RPM produce nausea by disturbing human metabolism through the balancing mechanism in the middle ear. Therefore a space facility

designed to generate a "safe" artificial gravity rotates at approximately 1.0 RPM on a radius of approximately 6,000 feet. If the biological nature of man proves to be such that weightlessness must be avoided, then the facility design problems would be greatly aggravated. However, it is possible to take advantage of a centrifugal configuration to greatly reduce the weight of the shielding required for the nuclear powerplant and reduce the radiation hazard involved in rendezvous operations with the facility. Figure 3-8A illustrates the type of design that might be considered to solve the above problems in an advantageous manner.

The technique of Figure 3-8A is only one possible configuration which might be desirable under specific circumstances. The entire field of shielding weight, structural arrangement, rendezvous and safety offers both opportunities and requirements for extensive investigation. This is a complex of problems which may require considerable effort in future studies.

Shielding requirements have a great effect on weight in orbit, but the basis for selection is not well defined at present. Therefore a study in depth to determine merits of various facility arrangements and different shielding is recommended.

4.9.2 CHEMICAL SECONDARY POWER

A highly efficient 30 KW chemical system consuming only 1.0 pound of fuel per KW - HR. will in a year's time burn a weight of fuel which is many times greater than the weight of the nuclear reactor plus its shielding.

Wt. of Fuel = 30 KW x 365-1/4 days x 24 hrs/day x 1.0 fuel/KW-HR.

Wt. of Fuel = 263,000 pounds for one year.

This weight is enormously greater than approximately 36,000 pounds for the weight of the Snap 8 nuclear power plant plus shielding.

4.9.3 SOLAR POWER SOURCE

There is a need for an alternate type of power plant in case it becomes very desirable to eliminate the nuclear unit from the design. Possible substitutes are various solar power plants described below:

Photovoltaic Power Systems use semiconductor solar cells which absorb light quanta and convert it to electricity within a single crystalline cell.

Photoemissive Solar Converters use large photo sensitive surfaces which eject electrons to be captured on adjacent screen electrodes. They employ some of the technology of both photomultiplier and image intensifier tubes.

Thermoelectric Solar Converters employ a semiconductor thermoelectric material to generate power from temperature difference between a pair of specially coated surfaces, one of which is heated by the sun.

Solar Dynamic Systems are Rankine cycle turboelectric generators using the sun as a heat source, radiation to space as a heat sink, and using exotic materials to secure high performance under difficult conditions.

Solar Thermionic Systems generate thermionic diode power when the cathode is heated by sun's rays from a solar concentrator. Vacuum type diodes are now on the market and cesium vapor diodes with higher efficiency will soon be available.

All high power, high performance solar power plants have large reflectors called solar concentrators whose purpose is to concentrate the sun's rays on the solar energy conversion equipment. A 30 KW installation requires a large unwieldy solar collector on the order of 64.0 feet in diameter. In addition, the collector must be aimed at the sun with an accuracy ranging from several degrees down to approximately 0.1 degree depending on the choice of system. Photovoltaic, photoemissive, and thermoelectric systems require several degrees accuracy, solar dynamic systems require approximately 1.0 degree accuracy and solar thermionic converters require the maximum accuracy approaching 0.1 degree. Most of the weight of a solar static system designed for satellite service around the earth is in the storage batteries which supply power while the facility is orbiting through the shadow

of the Earth. Data on the component weights of a solar cell power plant orbiting about the Earth has been obtained from WADD Technical Report 60-699, Energy Conversion Systems Reference Handbook, Volume IX - Solar System Design by W. R. Menetrey. Figure 3-8B from the report shows the weight of the component parts in pounds/KW versus satellite altitude for different satellite altitudes with the equipment available in 1962. At the Orbital Launch Facility, altitude of 300-400 miles, system weight is a maximum. Figure 3-8C extends the above data to the future with total weight curves for both 1965 and 1970. The startling drop in total system weight from 800 pounds/KW in 1962 to 150 pounds/KW in 1970 is a projection based primarily on reduction in battery weight. This is expected to occur either as a result of the use of new types of batteries such as fuel cells or improvements in reliability in presently known batteries such as silver-zinc or nickel-cadmium batteries. Present reliability of nickel cadmium batteries limits the depth of discharge to 10.0% of maximum for the repeated cycling application characteristic of relatively low altitude Earth satellites. Therefore improvements in battery reliability without any increase in maximum battery charge could produce the future performance increases projected in Figure 3-8B. This information on battery development applies to all solar static power plants in low satellite orbits where the storage batteries are the primary component weight in the system. Note that the above weights for a solar cell system gives a total weight for a 30 KW solar cell system of 24,000 pounds by 1962, 4500 pounds by 1965, and 2100 pounds by 1970. However, it is questionable whether the above weights per KW can be

maintained at a power level as high as 30 KW for a solar cell system which are inherently suited to lower power levels. But, if the projected development of these systems takes place over the next few years then they would provide a power plant appreciably lighter than Snap 8 with its shielding.

The solar dynamic system appears to be most suitable for the power level of very approximately 4 KW to 20 KW. Proper comparisons for this type of power plant can be obtained only by employing complete designs. An excellent example of this type of solar power plant is the 15 KW Solar-Mechanical unit which WADD is now funding. This is a 15 KW solar power plant weighing 819 pounds and using a 40.5 ft. diameter solar concentrator. It employs several different chemicals for thermal storage when the satellite is in the Earth's shadow. To compare this with other techniques of power generation it is most convenient to employ weight per unit power. This 15 KW unit weights 55 pounds per KW which is the lowest of any system investigated in this report. Aside from the large solar concentrator, this unit has the disadvantages that it will not be ready for approximately four to six years if there are no major delays, and the power output is 15 KW instead of the 30 KW required by the Orbital Launch Facility. There are three alternatives to the latter problem:

1. Use two 15 KW units
2. Attempt to purchase a 30 KW unit with a considerable increase in lead time.

3. The 30 KW power requirement, which is only an estimate, could be reduced to 15 KW supplied by a single solar dynamic unit weighing 819 pounds. It has been estimated by the cognizant personnel at WADD that a 50 KW unit would weight about 45 to 50 pounds per KW. In addition, they have estimated that this specific weight is a minimum at this power level and would rise for either an appreciably higher or lower power level.

Comparative Reliability of Proposed Power Plants.

Although the Snap 8 nuclear power plant is intended to have an inherent reliability capable of insuring at least one year of operation, it does possess the disadvantage that it must be shut down for all maintenance and even then it must be handled with manipulators through a protective shield. Any reliability of power continuity during maintenance periods almost certainly requires the use of at least two Snap 8 power plants with the increased weight. Solar cells may suffer slow deterioration over a long period of time caused by bombardment by X-rays and gamma rays particularly during severe solar flares. However, this deterioration is sufficiently slow such that over a one year period the reliability should remain quite high. Because the solar cell power plant uses many solar cells, whose weight is not a major portion of total system weight, it is possible to greatly improve reliability in the solar cell bank by wiring them in a mixed series parallel arrangement having a great increase in redundancy.

The storage battery portion of the system can be easily separated into many small units with sufficient switching equipment to remove a defective battery from the circuit. The static electrical conversion and regulation equipment possesses more than adequate life. In addition, the entire system is readily repaired by the crew without any special precautions.

The solar dynamic system is a mechanical system constructed from materials that are not deteriorated by space radiation. Again, the system is readily repaired by the crew without any special precautions against radiation. Two or three 15 KW power units would permit one unit to be shut down for repairs while the other unit(s) provided continuity of electrical power.

4.9.4 SELECTED SYSTEM

The Snap 8 system is the only system which has a reasonable development schedule for dovetailing into the overall OLO schedule despite its disadvantages. These disadvantages are that presently nuclear power plants are relatively difficult to maintain; present a political problem in their operation; require heavy shielding, and have what is perhaps an unnecessary radiation hazard to the facility crew. Although other power plants such as the 15 KW solar dynamic system appear vastly more attractive from the weight and size standpoint than either solar cell or nuclear, the very severe limitation of 15 KW imposes stringent limitations on the OLF equipment. The use of two or three solar dynamic systems is feasible but requires an operational analysis of the use of the volume immediately surrounding the OLF due to the large solar concentrator areas required. Sperry Rand

proposes to study the details of Orbital Launch Facility operation to determine whether a large collector would interfere with the proper functioning of the facility. The study would determine what changes are necessary in facility operation to accommodate the operation of the solar power system. A comparison will be made to determine the net change in the usefulness and cost of the facility for the new configuration.

Solar Power plants are presently behind nuclear power plants in larger power ouputs, require large solar collectors which must be oriented to the sun, and will probably never be suitable for very large power outputs. At the present time the solar power art is developing rapidly requiring careful observation to provide the latest information on developments in this field. We therefore recommend as a specific follow-on job that Sperry Rand will visit various laboratories where the latest techniques are being developed to provide a systematic up to date comparative summary of the state of solar power plant development. For the higher power levels required by the Orbital Launch Facility, solar power plants are in the early planning or development stages. Their projected performance and reliability five or more years from now contains a measurable element of chance. However, in several years solar power development may improve to the point where it will possess a high probability of success. This information is necessary for any fundamental decisions involving the choice of secondary power sources on the Orbital Launch Facility.

5.0 - COST AND SCHEDULE

The S/C, OLF standby, payload, and astrotug power system development plan and schedules which follow indicate the magnitude of effort required before the proposed power system can be expected to achieve operational status. This program is a natural extension of the present Vickers work on an Air Force contract for development of an internal combustion hydrogen-oxygen engine for space power application. The present contract work allows the projected program schedule to be shortened correspondingly.

The R & D phase of the engineering development program can be subdivided into three areas. These are: 1.) Evaluate current Air Force contract work at Vickers to demonstrate feasibility of the internal combustion hydrox engine concept and obtain basic data; 2.) Demonstrate feasibility of the regenerative power system concept; 3.) Demonstrate feasibility and develop it to production prototype status of the advanced stoichiometric chemical power system. Beyond this R & D phase is an additional effort required in product engineering, service, reliability, qualification, logistics program support, etc., to achieve true operational status. The total program is outlined below (see figure 3-8):

I. R & D Phase

A. Current Air Force contract work.

1. Demonstrate feasibility of hydrox internal combustion engine concept.
2. Obtain basic data on combustion, cooling requirements, etc.

3. Determine performance at specific conditions.

B. Basic power system development (system R & D work leading to feasibility demonstration of complete "breadboard" power system operating at fuel-rich mixture ratio)

1. Perform power system studies, thermodynamic analysis, and controls analysis; predict system performance; identify probable problem areas.

2. Test "breadboard" system; determine actual system performance; obtain basic data on system performance, stability, and general characteristics; resolve basic problem areas.

3. Demonstrate feasibility of complete regenerative power system operating on cryogenic propellants.

C. Advanced power system development (system R & D work leading to production prototype status of complete power system to operate at stoichiometric mixture ratio)

1. Perform engine and power system studies, thermodynamic analysis, and controls analysis; predict system performance; identify probable problem areas.

2. Test "breadboard" system; determine actual system performance; obtain basic data on combustion characteristics, system stability, and general operational characteristics; verify and resolve basic problem areas.

3. Demonstrate feasibility of complete power system operating on cryogenic propellants at stoichiometric ratio.

4. Develop system to production prototype status; resolve prototype system problems.

II. Operational Qualification Phase

A. Operational system development

1. System design

2. Development testing

B. Reliability program

C. Production engineering

D. Service engineering

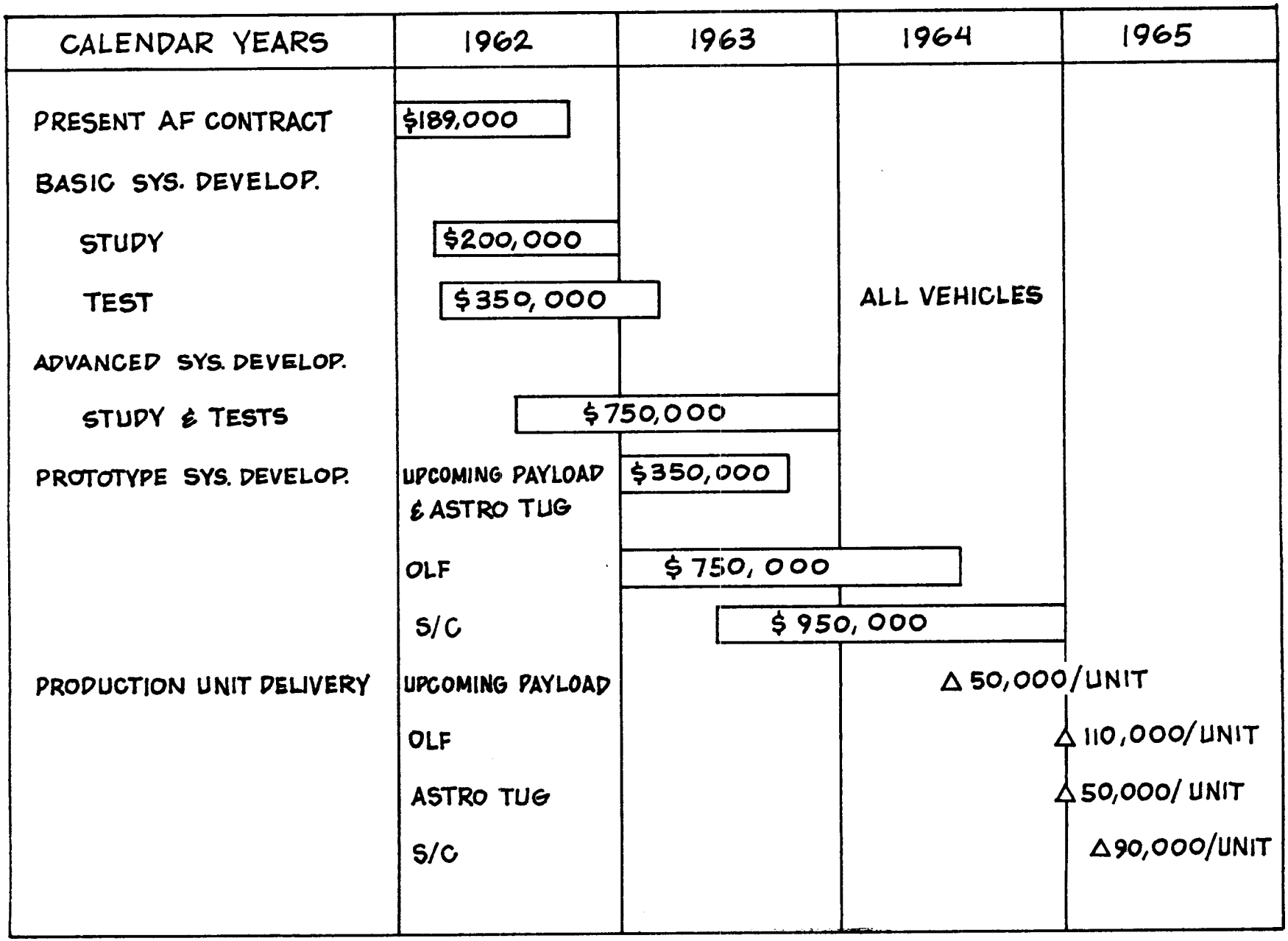
E. Manufacturing

F. Logistics planning

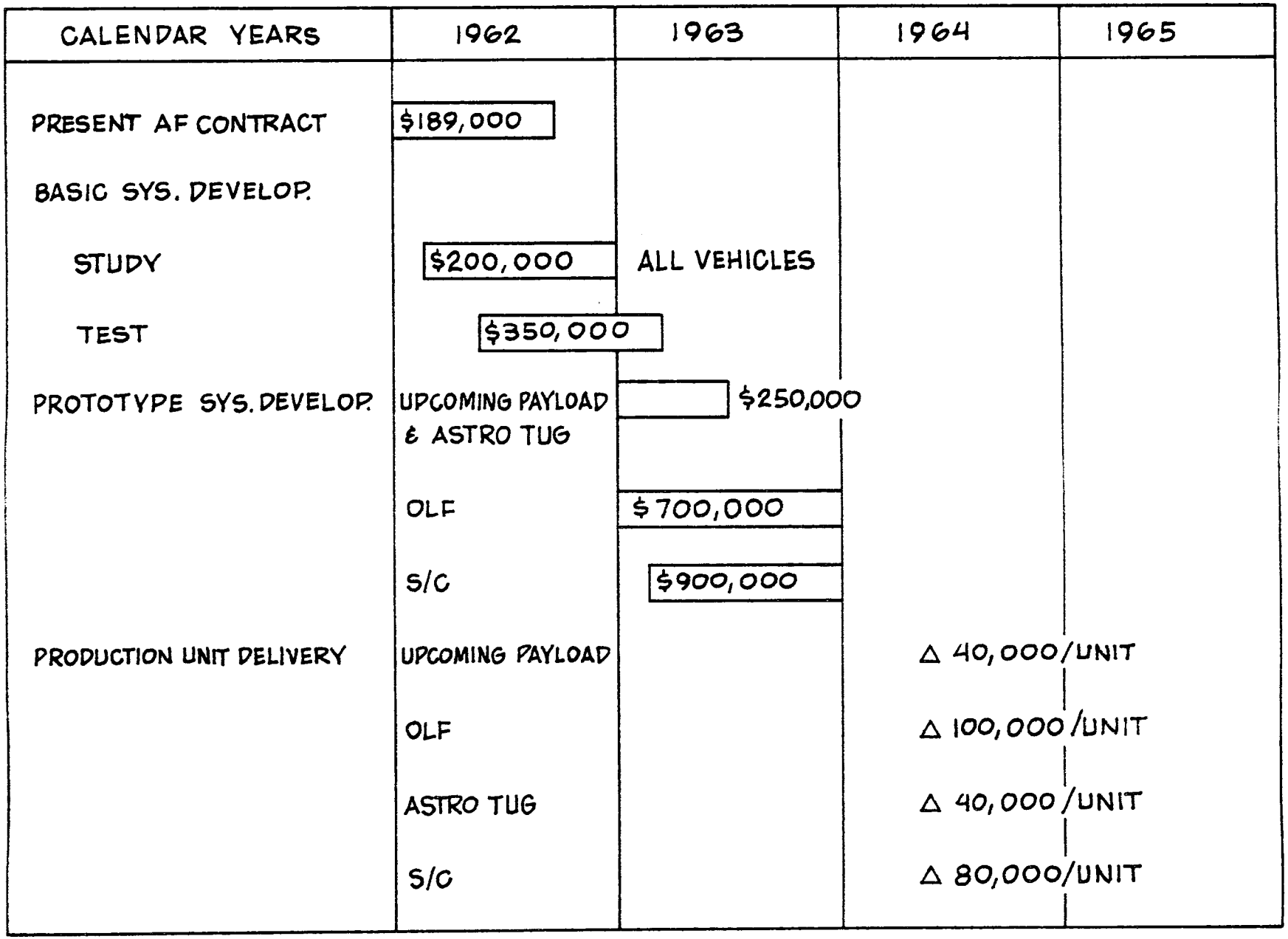
G. Program support

The R & D work on a basic power system operating fuel rich (shown as Phase I-B of the development program) is considered a necessary transitional step between the present engine program and the complete advanced power system operating stoichiometrically. The development program schedule shows that the advanced power system could become operational by 1965. If an operational system is required before that time, efforts could be concentrated on the basic fuel-rich power system which could be made operational at least one year earlier. With an early contractual authorization and sufficient support, Vickers believes that the basic Hydrox engine power system could be made operational by the end of 1963.

PROGRAM SCHEDULE AND COST, STOICHIMETRIC SYSTEM



PROGRAM SCHEDULE AND COST, FUEL RICH SYSTEM



6.0 - PROBLEM AREAS

The problem areas associated with a cryogenic hydrogen-oxygen internal combustion engine are listed below. These criteria will require the most development and will prove to be the limiting factors with respect to engine reliability, cost, complexity, and economy of operation.

1. Lubrication
2. Valve and injector mechanism design
3. Combustion

A wealth of background information, much of which is directly applicable, can be obtained from the literature on air-breathing internal combustion engines. A high degree of mechanical sophistication is permissible in a well developed system if enough redundancy is used to ensure reliability.

Some solutions to the problems mentioned above are as follows:

Lubrication - High temperature lubricants such as polyphenyl ethers are being investigated for operation in the two-stroke-cycle internal combustion hydrogen-oxygen engine currently under development at Vickers. The results of this investigation will be directly applicable. Open-cycle and closed-cycle lubrication will both be considered, and a choice will depend on lubricant consumption and relative heat transfer.

Injector and Valve Design - High inlet temperature and pressure, adverse valve and/or injector operating environments, and stringent

operating time and tolerance requirements create a difficult material and mechanical design problem for a propellant inlet mechanism. This problem is under investigation at Vickers in the program mentioned above. A high level of design sophistication in these areas is permissible, and much design experience is available.

Combustion Problems - Difficulty in the combustion of hydrogen in air has been experienced and the use of antidetonants is not desirable. However, variations in inlet pressures and mixture ratios and the possibility of premixing all or part of the combustible mixture leaves a large degree of latitude in the problem approach. This again points to the desirability of a well developed engine which can be extensively tested on land prior to incorporation into the system.

Temperature - The primary difference between the simple and the advanced engines lies in the operating and propellant inlet temperatures. (Both operate at a Carnot cycle efficiency of 0.74). An extensive investigation into the use of exotic materials, construction methods, and lubricants will be necessary to ensure reliability at the proposed stoichiometric combustion temperatures. Such an investigation would be more fruitful for a reciprocating positive displacement engine than for its turbine counterpart, for two reasons:

1. Peak cycle temperatures in a reciprocating engine can be relatively high with moderate average temperatures.
2. Operating speed and stress levels are much lower in reciprocating engines than in turbines, relaxing material strength requirements.

7.0 - CONCLUSION

Vickers has invested close to half a million company dollars over the past few years in the study and development of cryogenic auxiliary power systems. Optimization studies indicated that significant weight advantages were offered by a hydrogen-oxygen system using a positive-displacement engine. This advantage is derived from the superior characteristics of hydrogen as a fuel and the capability of the positive-displacement engine to operate efficiently at varying loads.

This same engine system is adaptable to closed-cycle operation utilizing solar or nuclear power for the long-range space vehicles. A study of the various types of expanders for power systems of this type indicates that the positive-displacement, variable-admission-ratio engine is more practical than other units contemplated for these missions.

An extension of the hydrogen-oxygen engine into a complete power and environmental system for space vehicles is foreseen in near-future development. The cryogenic hydrogen and oxygen perform several functions. Not only are they employed to drive the hydraulic pump and alternator, but also for crew compartment cooling, equipment cooling, heating, and attitude control.

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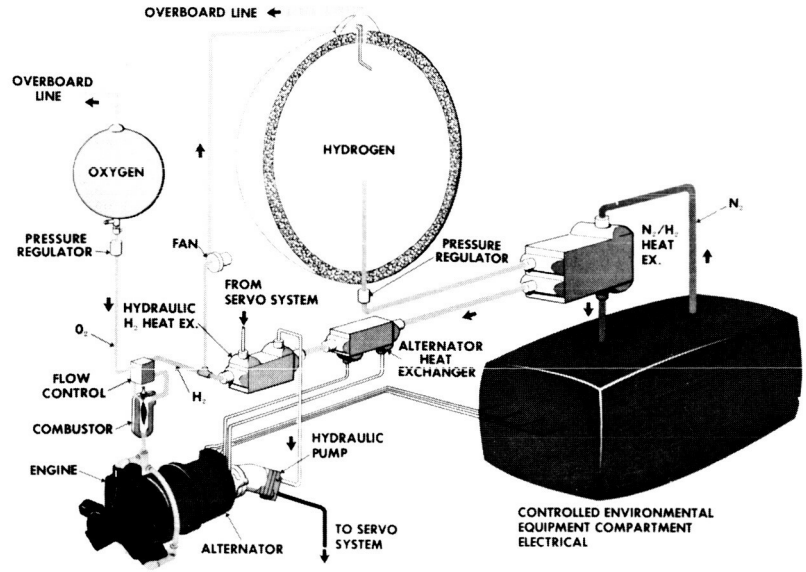


FIG. 3-1 - SYSTEM SCHEMATIC

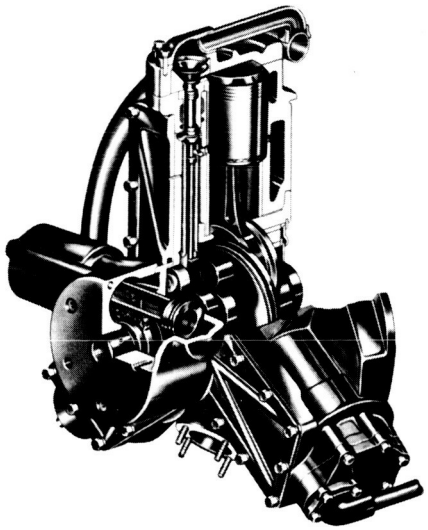


FIG. 3-2
POSITIVE-DISPLACEMENT ENGINE

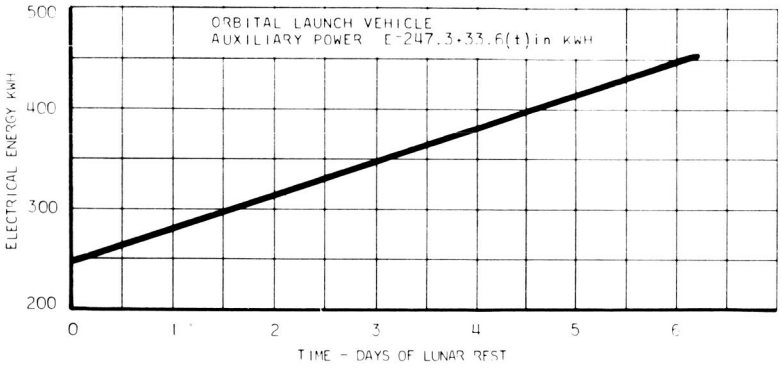


FIG. 3-3 - ELECTRICAL APS ENERGY REQUIRED IN TERMS OF LUNAR REST

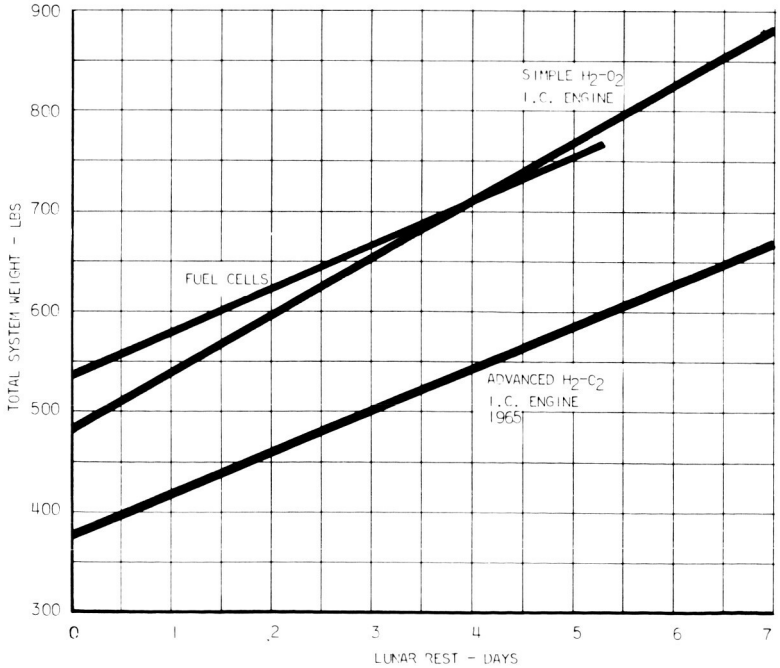


FIG. 3-4 - WEIGHT VS. DAYS AT REST

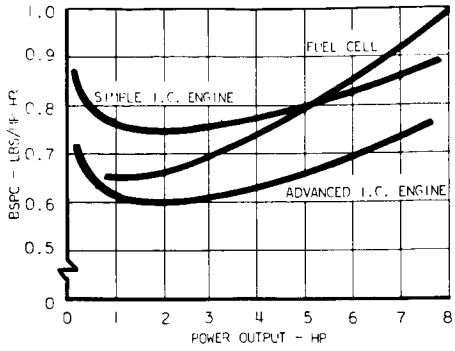


FIG. 3-5 - COMPARISON OF BSPC TO POWER OUTPUT

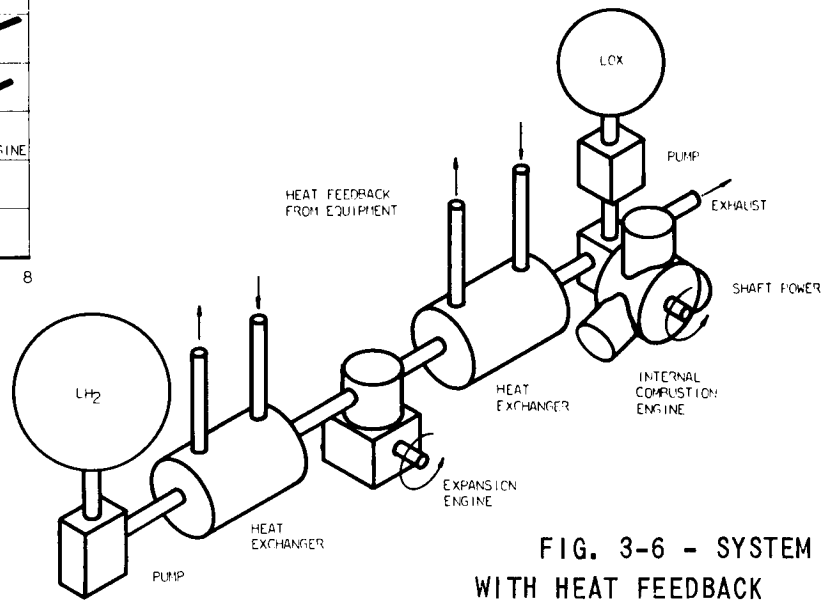


FIG. 3-6 - SYSTEM WITH HEAT FEEDBACK

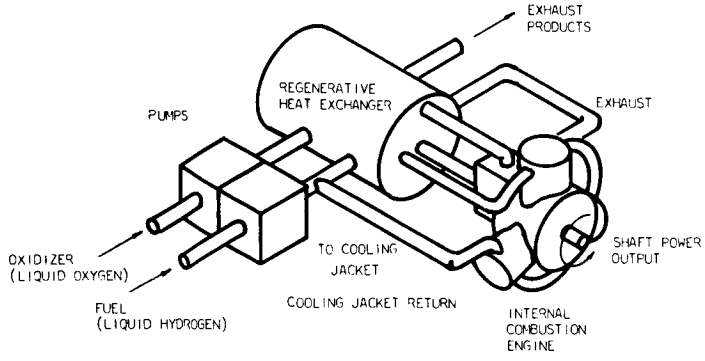


FIG. 3-7 - INTERNAL COMBUSTION REGENERATIVE POWER SYSTEM

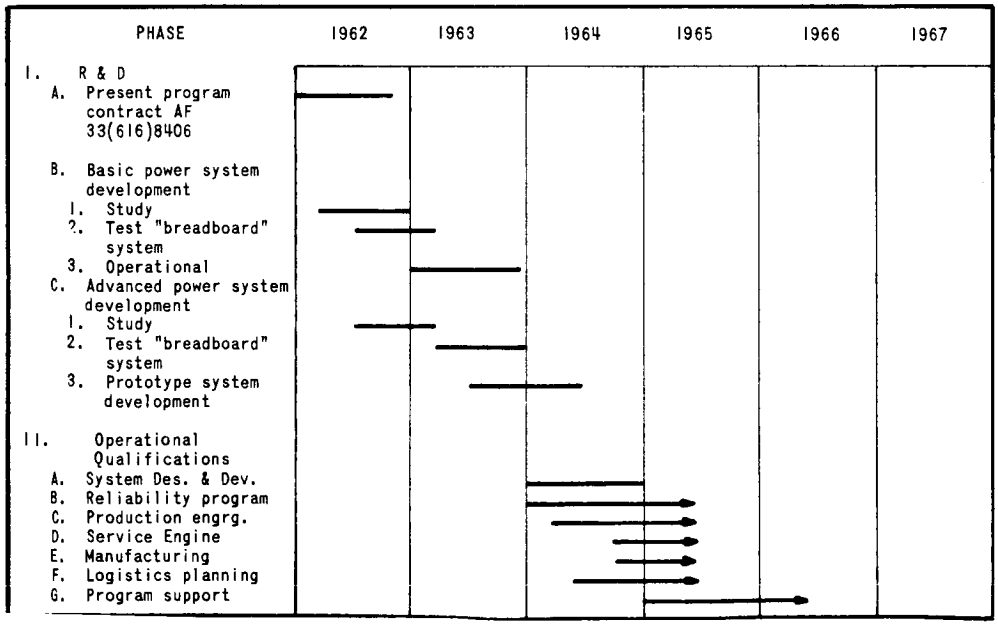


FIG. 3-8 - DEVELOPMENT PROGRAM SCHEDULE

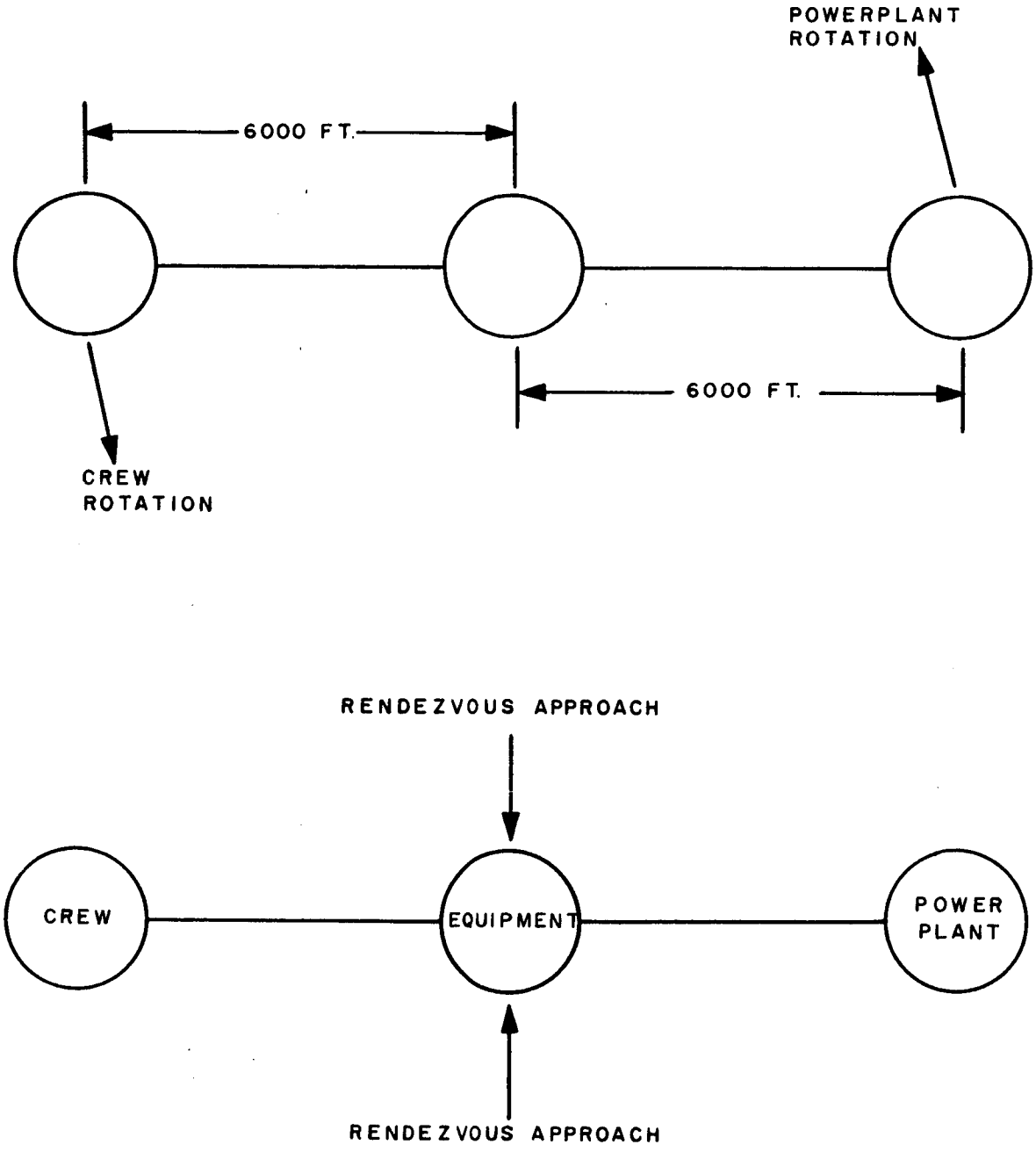


FIG. 3-8A. ORBITAL LAUNCH FACILITY WITH SAFE ARTIFICIAL GRAVITY

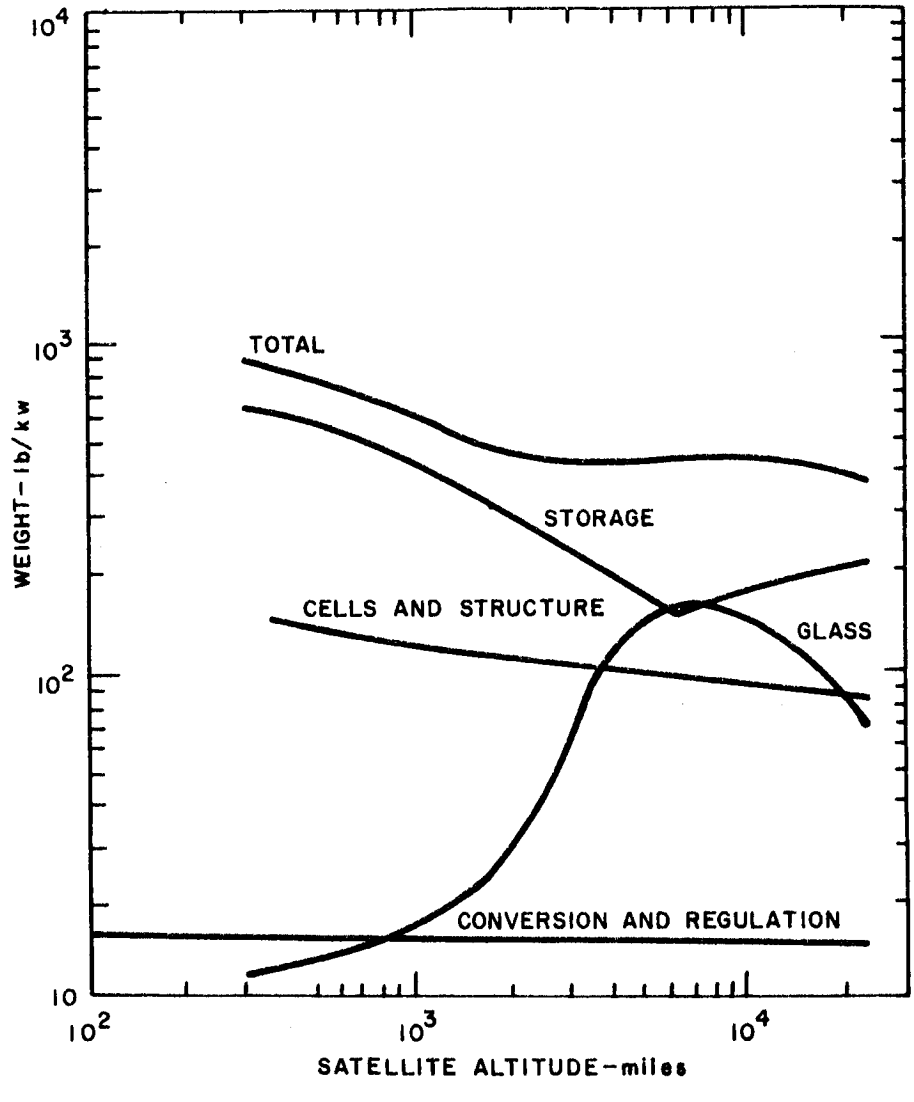


FIG. 3-8B. PHOTOVOLTAIC SYSTEM WEIGHT - 1962

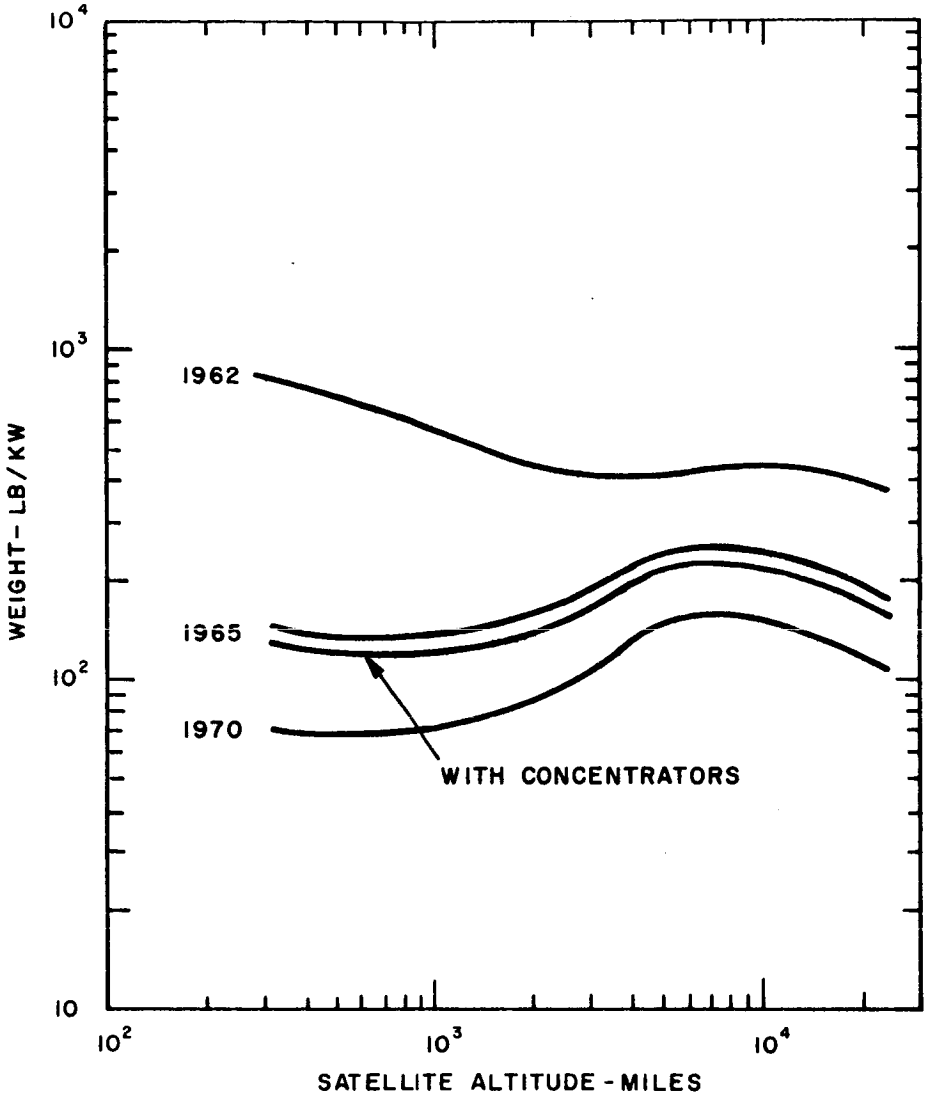


FIG. 3-8C. PHOTOVOLTAIC SYSTEM WEIGHT

APPENDIX A - SUPPLEMENTARY SYSTEMS

Hot Gas Motor APS - Although hydraulic pumps driven by electric motors or flywheels are satisfactory for many low-power, short-duration applications, Vickers has placed accelerating emphasis on the hot gas power concept because of its many attractive advantages in space.

Concept - The Vickers hot gas motor APS consists of a pressure-regulated solid-propellant gas generator powering a modified Vickers positive-displacement motor that drives a standard Vickers pump. Shown schematically in figure 3-9, this system combines the advantages of compact propellant energy with the inherently high efficiencies of positive-displacement machines driven at moderate rotating speeds. A preliminary study of power versus specific weight is presented in figure 3-10.

Design - A photograph of the hot gas APU (integrated motorpump package) is presented in figure 3-11. The hot gas pressure regulator that was developed for the system is shown in figure 3-12. The pump is a standard unit with approximately one-half the displacement of the motor so that 1500 PSI gas pressure can be converted into 3000 PSI hydraulic pressure.

Several modifications were necessary to convert the basic Vickers hydraulic motor for efficient and reliable operation with high temperature gas. New materials with high wear resistance and nearly identical thermal expansion characteristics were selected for the cylinder block and pistons. The shaft seal was omitted, and a special valve plate was provided to withstand friction and temperature conditions.

The gas generator is of simple cylindrical construction, incorporating

an igniter squib and rupture diaphragm. The generator is designed to use solid propellants, including the single-base ammonium nitrate and the double-base nitroglycerin nitrocellulose. Many tests have proven the suitability of using either type of propellant charge. Motor inlet filtration has been found unnecessary using these propellants since the only contaminant produced by combustion is a soft carbonaceous substance that does not impair motor operation.

In addition to the motorpump APU concept, Vickers has developed motors that operate exclusively on hot gas to power mechanical devices. One of these motors is discussed later under "Applications".

Operation - An electrical signal to the gas generator igniter initiates propellant combustion. The grain combustion rate is constant at the constant pressure maintained by the pressure regulating valve. The high pressure gas (e.g. 1500 psi) preloads a hydraulic accumulator and the motor inlet. The accumulator thereby maintains gas system pressure in the pump motor case. The motor outlet is open to atmosphere, and case pressure acting behind the motor pistons causes them to reciprocate by virtue of the differential pressure across the motor inlet and outlet ports (i.e., the opposing pressures on the pistons on the inlet side are balanced, whereas a 1500 PSI differential exists on the outlet pistons). The motor drives the pump through a common shaft, producing hydraulic power.

The unique feature of driving motor pistons from the back side prevents gas "blow-by" past the pistons since opposing pressures are equal. Any oil leakage through the motor outlet is negligble compared with gas

leakage under the same conditions, and volumetric efficiency is optimized. This feature also provides a sink to dissipate heat from the motor components.

Applications - Vickers and the United States Air Force invested over a half-million dollars in a development program with the conviction that the hot gas motor auxiliary power system (APS) concept represented a significant advance in the state of the non-propulsive power art. A contract was awarded to Vickers by the Thiokol Chemical Corporation, working on the control and propulsive systems for the Minuteman ICBM. An accelerated research and development program was initiated to produce a compactly integrated APS within a seven-month period.

Twelve breadboard units were built, and ninety-one tests were conducted in 30 days to establish motorpump reliability. Twelve prototype APS systems were then fabricated to substantiate system reliability through a series of 123 operational tests. The flight hardware system is shown in figure 3-13.

Aside from testing of the self-sufficient APS, a program was initiated to establish the feasibility of operating a motorpump auxiliary power unit (APU) with gas bled directly from the Thiokol rocket engine. A successful bleed-gas test was conducted utilizing a Thiokol TU-132 engine to power the hardware. A peak of 2.15 hydraulic horsepower was developed after the hydraulic flow reached steady-state of 0.54 seconds and remained at 5.2 GPM for the duration of the test. Clogging of the gas pickup terminated the test after 16 seconds of operation.

The experiment was so successful that Thiokol sponsored a second feasibility test performed several months later with the test rig shown in figure 3-14. Results of the second test definitely established the feasibility of operating the Vickers APU at high temperatures with highly contaminated gas.

The motorpump ran for the full 62-second burning duration of the Thiokol main engine propellant charge. Motor inlet temperatures were recorded up to 3600°F with gas solid-contamination estimated at 10 to 15 percent by weight. A breakdown inspection of the motorpump revealed all parts to be in excellent condition (including the motor pistons which moved freely in the cylinder bores). A peak output of 2.35 horsepower was developed during this test with a steady-state hydraulic flow of 5.2 GPM.

Additional laboratory testing was conducted to simulate maximum temperature bleed operation using a near-stoichiometric bipropellant mixture of hydrogen and oxygen. The APU operated for 39 seconds at propellant flame temperatures estimated between 5000° and 5200°F, until a stainless steel line directing gas from the combustor to the motor ruptured. The motor components were found in excellent condition as shown in figure 3-15.

The hot gas motorpump shown in figure 3-16 is currently being used on the Skybolt missile. The 100-horsepower hot gas motor presented in figure 3-17 was produced for General Electric. Delivery of prototype hardware required only ten weeks (sixty days) from receipt of the specifications. Basic design, testing and modifications, final prototype assembly, and delivery were accomplished within this period.

This unit has undergone an acceleration rate of 50,000rpm/sec. in the G.E. application at essentially constant torque over its entire speed range. The motor is designed to operate on either a double-base solid propellant or on a liquid monopropellant, although all runs to date have been powered by the double-base solid propellant.

Battery-Powered APS - Although the potential applications of battery powered auxiliary power systems are limited in the OLO concept, they may be suitable for any short duration boosters associated with the program. A brief discussion of this approach to providing auxiliary (or primary) hydraulic power is included.

Concept - A schematic diagram of an integrated battery-powered hydraulic system is presented in figure 3-18. The essential components of the system are silver-zinc batteries, a d-c motor driving a hydraulic pump, a fluid reservoir, and valves. A preliminary study of power versus specific weight is presented in figure 3-19.

Design - Many APS designs based on this concept employ the "building block" principle whereby "off-the-shelf" electric motors are mounted to "off-the-shelf" hydraulic pumps. Where space and weight parameters require a more sophisticated approach, complete integration of the APS components can be achieved as shown in the schematic diagram.

A photograph of an integrated battery APS used to provide hydraulic power for the Minuteman missile is shown in figure 3-20.

Vickers has been producing airborne electrical motorpump packages for over 15 years.

Applications - Other recent applications include d-c motorpumps used to power hydraulic systems for the Pershing, Titan, Bomarc, and Polaris missiles. The Polaris motorpump, shown in figure 3-21 is a completely integrated unit with a special uncured latex coating that absorbs ambient heat (in curing), thereby protecting the unit from its high temperature surroundings.

Flywheel APU - Concept - Flywheel-powered APU's are attractive for many specialized applications in an orbital launch operation. The system consists essentially of a flywheel, hydraulic pump/motor, reservoir, and valves. These elements are related as shown in figure 3-22.

By using the pump/motor concept, the flywheel absorbs energy whenever system pressure rises above the nominal value and delivers exact quantities of power when the pressure falls below that value.

Design - The highest specific energy storage capability for a given maximum flywheel stress is theoretically achieved with a uniformly stressed rimless disc. However, this design requires top speeds that cannot normally be attained within practical size and speed limits. Furthermore, windage losses are highly sensitive to speed with a uniform-stress disc, dissipating a large proportion of the total stored energy in most applications. These considerations have led to the heavy rim flywheel for the unit shown in figure 3-23 that was designed for the Atlas missile.

The flywheel for this design is shrouded to accommodate air evacuation, minimizing windage losses. The pump/motor employed is essentially a standard fixed-angle variable delivery unit equipped with a special valve block to enable two-way operation (pumping and motoring).

A generalized weight study comparing specific system weight (not including flywheel weight) with rated output power is presented in figure 3-24. The flywheel weight depends on operating duration. Based on a tip speed of 1500 fps and allowing for windage losses, the duration factor for a flywheel is 150 lb/hp-hr. For example, a ten-horsepower system of five-minute duration will weigh 135 pounds (125-lb flywheel). A similar system for five seconds duration will weigh only 12.1 pounds (2.1-lb flywheel).

Operation - During run-up, hydraulic power is used to accelerate the flywheel to rated speed. This is accomplished by driving the flywheel-mounted hydraulic pump as a motor, using power either from ground supply or from vehicle supply (if the flywheel system is to be used later in the mission).

The flywheel-mounted hydraulic pump comprises a unit that has the characteristics of a highly efficient accumulator. A constant-pressure hydraulic output is achieved if the pump/motor unit is pressure-compensated to provide variable delivery, thereby compensating for flywheel energy degradation.

Application - The flywheel APU is particularly attractive in staged vehicles. For example, a flywheel system can be installed in the second (or third) stage of a vehicle, and the first-stage primary hydraulic system can "charge" the flywheel APU during any period before first-stage jettison when hydraulic demands are low. The second stage is thereby endowed with an independent hydraulic power supply without paying space and weight

penalties imposed by fueled prime movers or batteries.

As mentioned previously, the flywheel APU can also be used as an accumulator in a constant pressure system where the average hydraulic power demand is low compared with intermittent peak requirements. The flywheel is "charged" during low power demand periods and is available to supplement the main hydraulic power supply when needed. The primary advantage over other accumulator devices is that compensation for energy degradation can be achieved by using a variable displacement pump/motor unit.

Availability - Since the flywheel APU is composed of relatively standard equipment, the only development necessary to achieve a production unit concerns selection of a flywheel size and configuration compatible with the mission parameters.

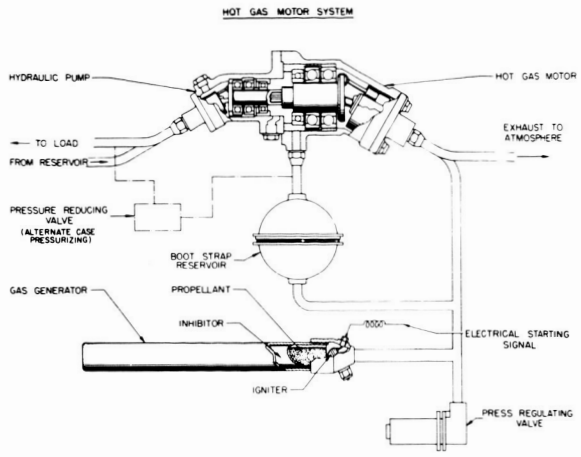


FIG. 3-9 - HOT GAS MOTOR SYSTEM

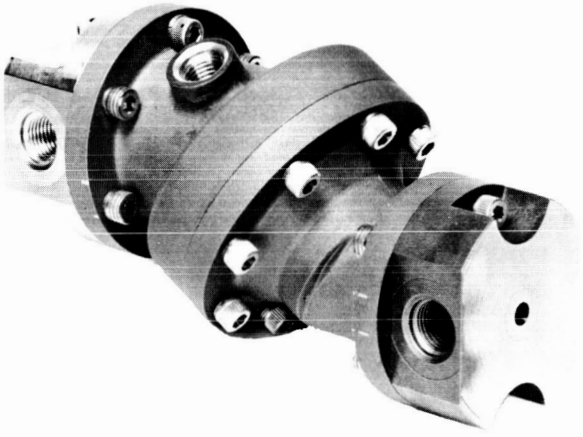


FIG. 3-11 - HOT GAS MOTORPUMP

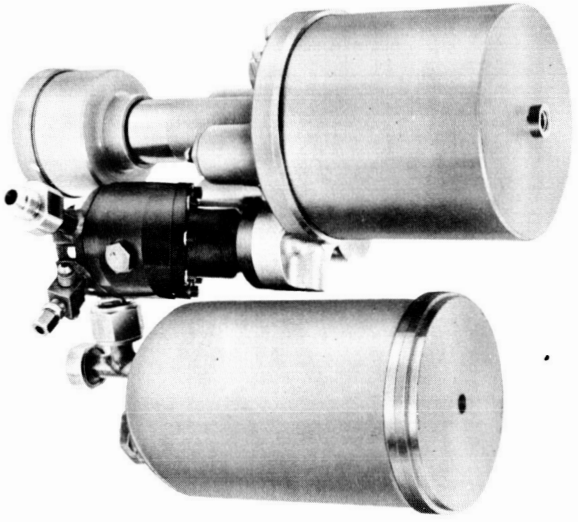
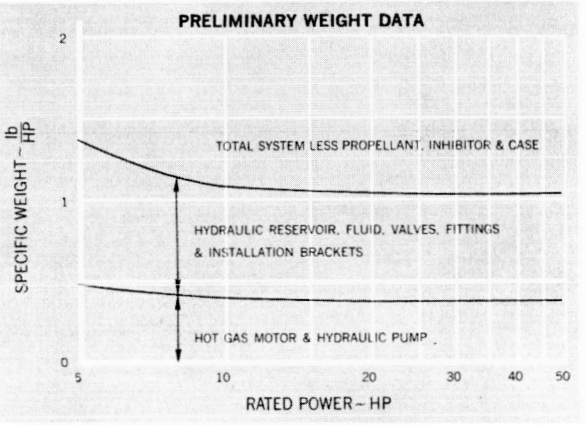


FIG. 3-13 - INTEGRATED HOT GAS APS



The duration factor for a short duration hot gas motor system utilizing a solid propellant is 27 lbs/hp-hr, which includes propellant, inhibitor and case weight. Therefore, as an example, a 10 hp system of 5 minutes duration will weigh 33.5 lbs. A 10 hp system of 5 seconds duration will weigh 11.4 lbs. For durations above 5 minutes, a partial admission motor with a duration factor of 18 lbs/hp-hr. would be used.

FIG. 3-10

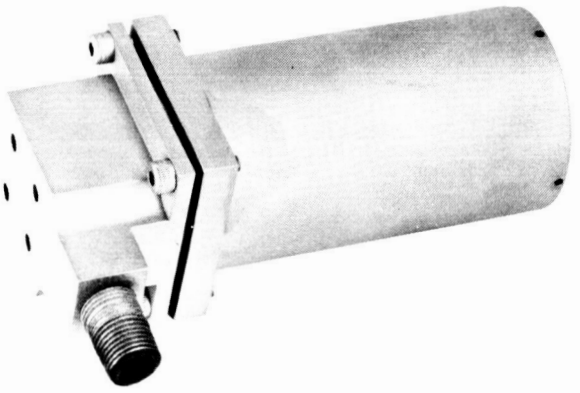


FIG. 3-12 - HOT GAS PRESSURE REGULATOR

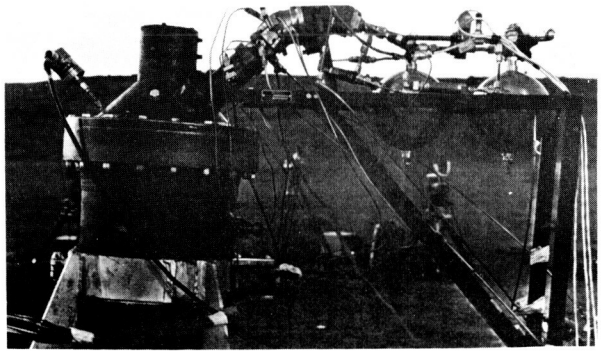


FIG. 3-14 - HOT GAS BLEED TEST

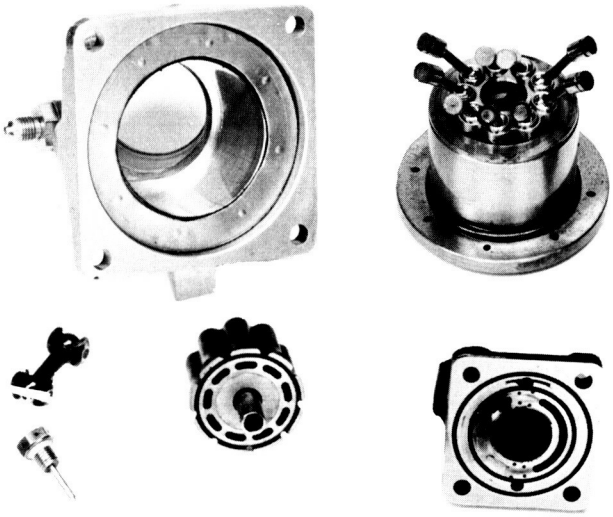


FIG. 3-15 - HOT GAS MOTOR AFTER OPERATION AT 5200°F

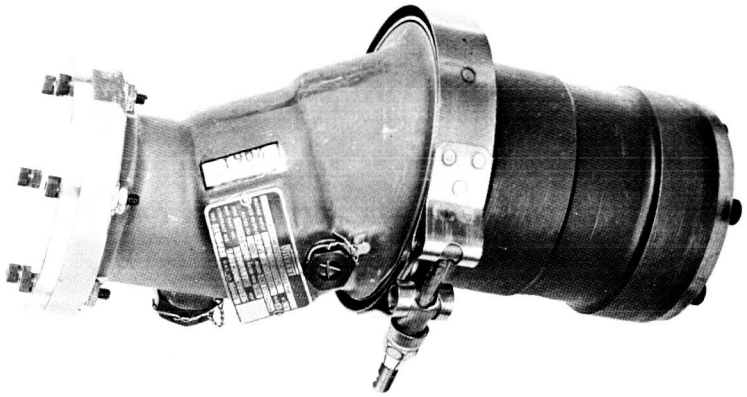


FIG. 3-16 - HOT GAS MOTORPUMP FOR SKYBOLT

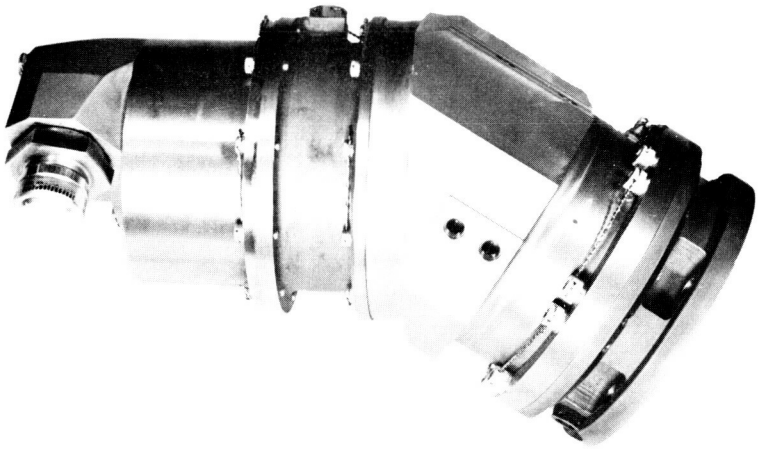


FIG. 3-17 - 100 HP HOT GAS MOTOR

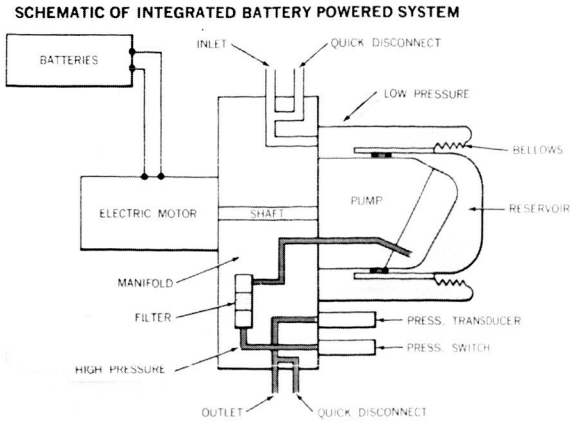
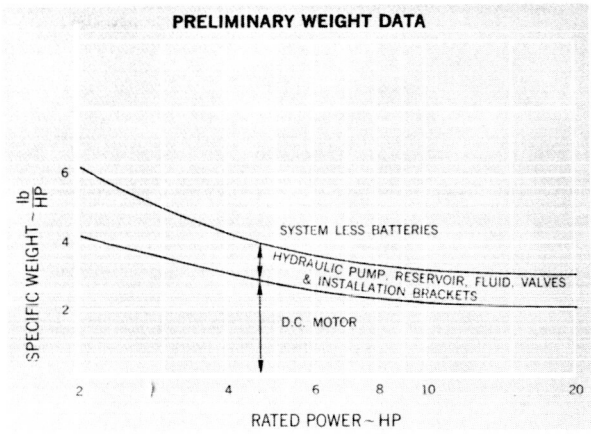


FIG. 3-18



The weight factor for a silver-zinc battery includes 1.0 lb/hp fixed weight and a duration factor of 15 lb/hp-hr. Therefore, a 1 lb/hp must be added to the d-c motor and hydraulic components specific weight shown on the curve at the left to obtain fixed system weight. As an example, a 10 hp system of 5 minutes duration will weigh 54.5 lbs. A 10 hp system of 5 seconds duration will weigh 42.2 lbs.

FIG. 3-19

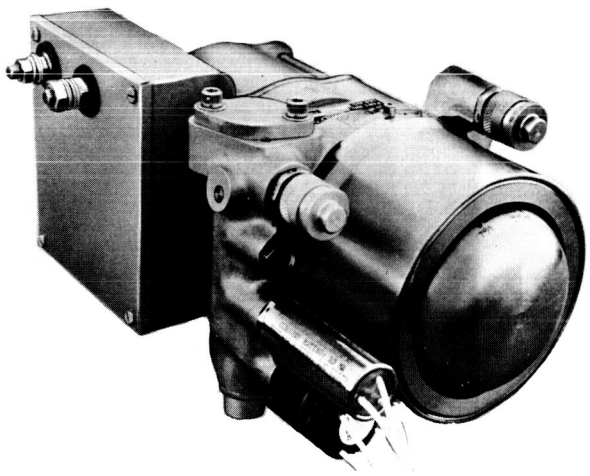


FIG. 3-20 - INTEGRATED BATTERY APS FOR MINUTEMAN

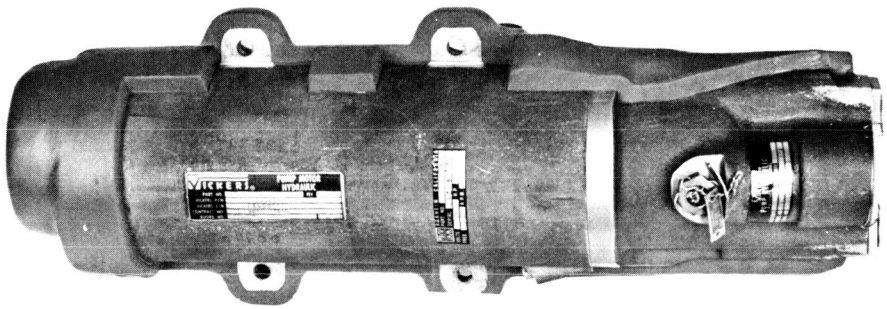


FIG. 3-21 - ELECTRIC MOTORPUMP FOR POLARIS

SYSTEM SCHEMATIC

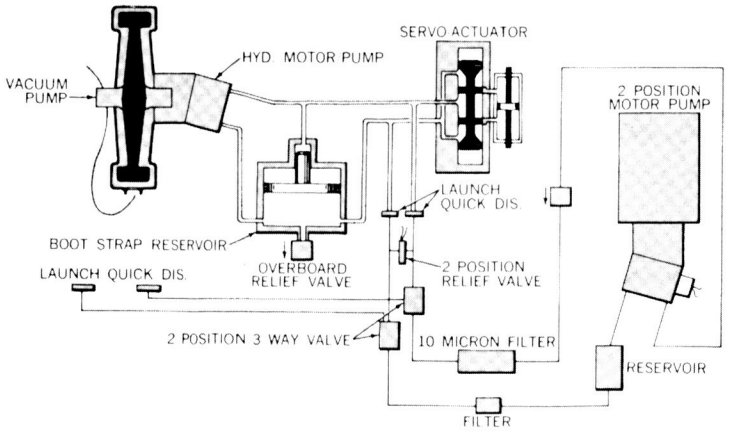


FIG. 3-22

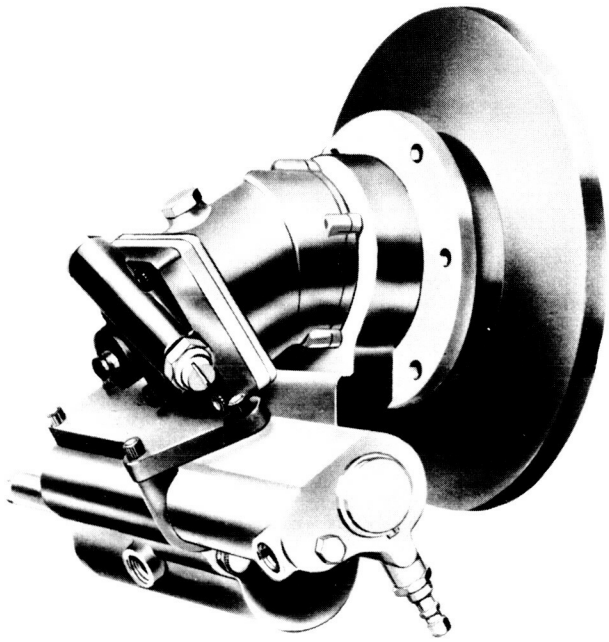
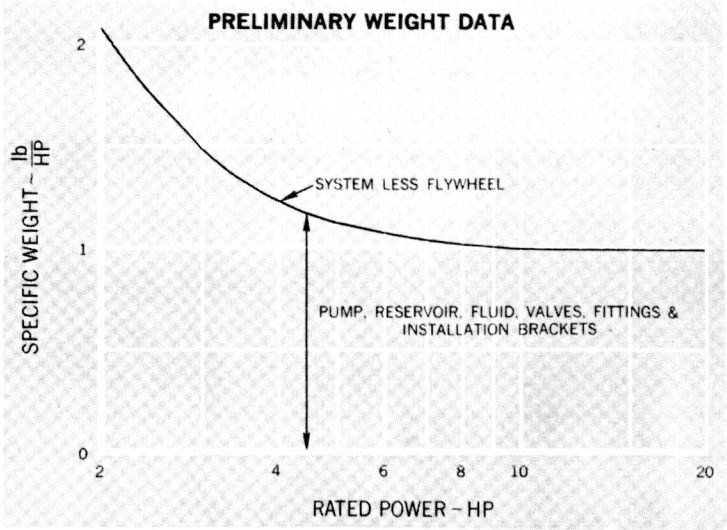


FIG. 3-23 - FLYWHEEL APU FOR ATLAS

FIG. 3-24



APPENDIX B - DEFINITIONS

1. APS: Auxiliary or Secondary Power System; a device to generate hydraulic, mechanical or electrical energy for any components or subsystems other than main propulsion.

2. BSPC: Brake Specific Propellant Consumption (Lbs/HP-hour)

3. OLO: Orbital Launch Operations

4. OLV: Orbital Launch Vehicle; vehicle used to launch spacecraft from earth orbit.

5. OLF: Orbital Launch Facility; an earth-orbiting facility used for platform assembly and launch of spacecraft and OLV.

6. LOF: Lunar Orbiting Facility; a lunar facility that serves as the orbital terminal for lunar base supplies and personnel.

7. LDV: Lunar Descent Vehicle; vehicle used to shuttle cargo and personnel between LOF and Lunar Base.

8. MLM: Manned Lunar Landing and Return Mission

9. Apollo A: 3-man Apollo spacecraft control module with specific equipment required for suborbital and orbital flights.

   Apollo B: 3-man Apollo spacecraft, control, equipment and propulsion modules for circumlunar and lunar orbital flights.

   Apollo C: 3-man Apollo spacecraft, control, equipment and propulsion modules including all equipment required for the lunar landing mission.

   Apollo D: 6-man Apollo-type spacecraft.

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117

ORBITAL LAUNCH OPERATIONS

VOLUME VII

SECONDARY POWER

PREPARED FOR
MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

PREPARED BY: VICKERS, INC., TORRANCE, CALIFORNIA

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SPERRY RAND SYSTEMS GROUP
GREAT NECK, NEW YORK

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TABLE OF CONTENTS

Paragraph	Title	Page
1.0	SUMMARY	1-1
2.0	INTRODUCTION	2-1
3.0	INITIAL CONSIDERATIONS	3-1
4.0	DESCRIPTION OF RECOMMENDED SYSTEMS	4-1
	4.1 OLF Secondary Power	4-1
	4.1.1 Nuclear Systems	4-1
	4.1.2 Chemical Systems	4-3
	4.1.3 Solar Systems	4-4
	4.1.4 Comparative Reliability	4-8
	4.1.5 Selected System	4-9
	4.2 Space Craft Secondary Power	4-10
	4.2.1 Candidate Systems	4-11
	4.2.2 Assumption	4-16
	4.2.3 Parametric Studies	4-17
	4.2.4 System Parameter Sensitivity	4-18
	4.2.5 Selected System	4-19
	4.3 Astro Tug Secondary Power	4-23
	4.4 Upcoming Payload Secondary Power	4-24
	4.5 System Options for Space Craft and Astro Tug	4-25
	4.6 Reliability and Maintainability	4-28
	4.7 Safety Aspects	4-30
5.0	SCHEDULE	5-1
6.0	PROBLEM AREAS	6-1
7.0	CONCLUSION	7-1
8.0	REFERENCES	8-1
	FIGURES (Charts, Graphs, Illustrations)	
	APPENDIX A - Supplementary Systems	
	APPENDIX B - Definitions	

List of Illustrations

Figure	Title	Page
Fig. 1-1	Electrical Power for Space Vehicles	1-4
Fig. 4-1	Orbital Launch Facility with Safe Artificial Gravity	4-32
Fig. 4-2	Photovoltaic System Weight - 1962	4-33
Fig. 4-3	Photovoltaic System Weight	4-34
Fig. 4-4	Candidate Power Systems Orbital Launch Facility 60 KW Power Level	4-35
Fig. 4-5	Internal Combustion Regenerative Power System	4-36
Fig. 4-6	Positive Displacement Engine	4-37
Fig. 4-7	Electrical APS Energy required in Terms of Lunar Rest	4-38
Fig. 4-8	Weight vs. Days at Rest	4-39
Fig. 4-9	Candidate Power Systems for Space Craft 1.7 KW Power Level	4-40
Fig. 4-10	Comparison of BSPC to Power Output	4-41
Fig. 4-11	Astro Tug and Upcoming Payload	4-42
Fig. 4-12	Power System using Propellant Boil-off	4-43
Fig. 4-13	System with Heat Feedback	4-44
Fig. 4-14	Internal Combustion Regenerative Power System	4-45
Fig. 5-1	Program Schedule, Stoichiometric System	5-4
Fig. 5-2	Program Schedule, Fuel Rich System	5-5

VOLUME VII - SECONDARY (AUXILIARY) POWER

1.0 - SUMMARY

The Sperry Rand Corporation is supporting Marshall's Orbital Launch Operations Study (Vought Astronautics Contract NAS 8-853) by investigating the major Astrionic requirements for the program. This volume, Secondary (Auxiliary) Power Systems (APS), covers the work accomplished by Vickers Incorporated, a division of Sperry Rand.

The results of studies conducted by Sperry Rand verify selection of the SNAP 8 nuclear power plant for the OLF, concurring with Vought results. The Sperry Rand studies also establish the advantage of using a chemical dynamic auxiliary power system based on a hydrogen-oxygen (hydrox) internal combustion engine for the OLF emergency standby system and for primary electrical power generation in the space craft, Astro Tug,and upcoming payload.

Hydrogen-oxygen (H_2-O_2) space power systems have been under development at the Vickers Torrance, California facility for over 3 years. As shown in Fig. 1-1, these systems are ideally suited for space vehicle applications with power requirements above 1 KW for continuous operation up to 2 and 3 weeks or intermittent duty for longer periods.

The H_2-O_2 chemical system is recommended for the space craft,upcoming payloads, Astro Tug, and as a standby system for the nuclear-powered OLF. This recommendation is based on power-time requirements, high potential reliability, light weight, and low cost. Trade-off studies on the hydrox auxiliary power system versus fuel cell and solar cell systems show considerable substantiation of this choice.

Brake specific propellant consumption (BSPC) is the most significant measure of efficiency when evaluating chemically fueled power generating systems for space applications.

The BSPC achieved during early engine tests by Vickers was 1.3 pounds per horsepower-hour. Further studies and a test program now being conducted in hydrox engine systems indicate that BSPC will approach 0.60 lb./hp.-hr. by 1964. (Equivalent to .90 lbs. per KW HR with 90% generator efficiency).

Vickers has established this hydrogen-oxygen engine provides an optimum auxiliary power unit for several applications in the orbital launch operation program.

The principal conclusions are:

1. The auxiliary power aspects of the OLO program are within the time scale for state-of-the-art development as conceived by Sperry Rand. No major technological breakthroughs are necessary to successfully establish OLO operations.

2. Certain follow-on study work is necessary to firmly establish plans for successful OLO operations. The work required in these auxiliary power areas can be accomplished by the Vickers Incorporated, Division of Sperry Rand Corporation. This work includes:

   a) Establish the design parameters necessary for OLO operations so that system design can be initiated.

   b) Specifications for APS and life support must be prepared outlining complete requirements for the overall OLO program.

   c) A more penetrating study of the S/C, Astro Tug, and upcoming payload power requirements must be made so that maximum performance and reliability can be achieved.

d) Additional support is required for follow-on work to the current Vickers hydrox engine development program to assure availability of flight rated systems within the time scale of the OLO program.

3. In addition, certain follow-on work related to the OLF also appears necessary. This work is described as follows:

   a) The shielding requirements for the Snap 8 power system appear to have a considerable influence on the configuration of the OLF. A more penetrating study of shielding requirements and its effect on the arrangement and operation of the OLF is recommended.

   b) The development of a possible alternate OLF power plant utilizing solar energy is progressing at a fairly rapid pace. This progress should be continuously monitored to take advantage of any major new developments.

4. Secondary power systems recommended by Vickers for the various OLO vehicles are summarized in Table A on the following page.

TABLE A

OLO SECONDARY POWER SYSTEMS

Vehicle	Requirements		Recommendations		
	Power, kw	Duration	Type	Weight, Lbs.	Volume, cu. ft.
OLF					
Primary	60	1 year	Nuclear Snap 8	10000 to 36000	140
Emergency	60	48 hours	Chemical Dynamic (Hydrox I.C.)	3505	96
Space Craft	1.7 avg. 2.3 peak	14 days	Chemical Dynamic (Hydrox I.C.)	780	22.0
Astro Tug	1.2	2 hours	Chemical Dynamic (Hydrox I.C.)	38	3.0
Upcoming Payload	1.0	1 hour	Chemical Dynamic (Hydrox I.C.)	35	2.5

ELECTRICAL POWER FOR SPACE VEHICLES

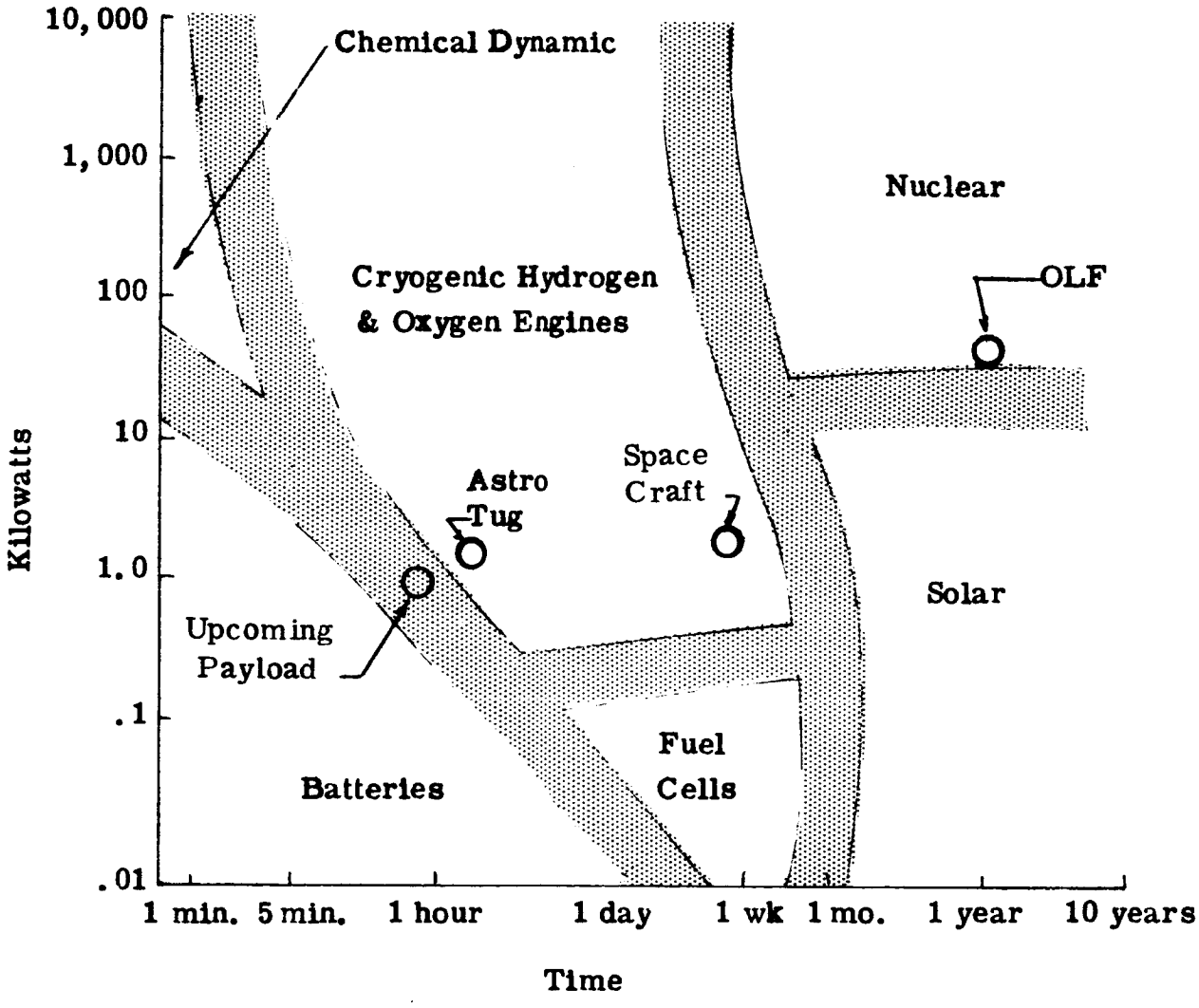


Fig. 1-1

2.0 - INTRODUCTION

The purpose of this report is to present the results of investigations conducted in the fields of auxiliary power systems. The Sperry Rand Corporation is conducting an across-the-board analysis of the various phases of Orbital Launch Astrionics to aid the Marshall Space Flight Center in developing a firm and well-documented plan for establishing the highly advantageous technique of Orbital Launch. The work is being conducted in connection with Vought Contract NAS 8-853, and is being guided by Vought work statements and coordinating documents. Every advantage is being taken of the results obtained by Vought in the earlier phases of the contract.

This phase of the Astrionic work is being funded by Sperry Rand as a contribution to establish the key engineering programs necessary to achieve success in this important endeavor.

Vickers Incorporated, a division of the Sperry Rand Corporation, has developed many "packaged power" systems. Some of these auxiliary power systems are an integrated part of the prime power system. Other systems are self-sufficient and extract primary power from an independent source. Since the forms of power (hydraulic, electrical, etc.) required by these space vehicles are similar to those used by missiles and certain conventional vehicles, the auxiliary power packages evolved by Vickers are natural extensions of Vickers basic product lines.

In this study the relative power requirements of various systems for the OLO program, the power required for each application, and the possible need for multiple systems were primary factors considered. The studies investigated auxiliary power requirements aboard the Orbital Launch Facility (OLF), Space Craft (S/C) upcoming payloads, and Astro Tug.

It became apparent early in the investigation of power plant requirements for the OLF that the results of the Sperry Rand analysis were closely following those obtained by Vought. The selection of the Snap 8 power plant system was confirmed on the basis of mission requirements, reliability, availability, size and weight. This is discussed further in Section 4.1 of this report. For standby power requirements for the OLF and for power supplies for the other OLO vehicles, however, considerable further detailed analysis appears to be required.

A significant fact relative to both research and production of chemical systems is that Astro Tug and Upcoming Payload Vehicle will be able to use the same hydrox fueled APU. The Space Craft APU is similar but has greater power and duration capabilities.

3.0 - INITIAL CONSIDERATIONS

Orbital and Space-Flight Vehicles - The requirements for the on-board OLO power systems were taken from the Vought Astro Design Data, Volume 2. Power parameters given for the four vehicles are tabulated in Table B on the following page.

The auxiliary power system is used primarily for electrical power, but will also provide a general power source. In applications where long-term nuclear power is desirable a chemically fueled APS can be utilized as a standby power system.

Table B is a breakdown of the vehicle power requirements for the guidance and control and overall electrical power. Later in the report the discussion will be illustrated to show how the hydrogen-oxygen engine system will apply directly to these three vehicles.

TABLE B

POWER SYSTEM SCHEDULE

VEHICLE	YEAR	POWER REQUIRED	
O. L. F.	1968	30	KW
	1970-1975	60	KW
SPACE CRAFT	1964-1966	1.5	KW
	1966-1969	2.5	KW
	1969-1975	3.5	KW
	3 MAN SPACE CRAFT		
	Guidance	2.3	KW/20 min.
	Misc. Power	1.7	KW/120 hr.
	Lunar Rest	1.4	KW/x days
	Final Phase	1.6	KW/1 hr.
	6 MAN SPACE CRAFT		
	Launch	3.2	KW
	Mid-Course	2.8	KW
	Lunar Land	3.2	KW
	Lunar Rest	2.7	KW
	Lunar Launch	3.2	KW
	Mid-Course	2.8	KW
	Earth-Guidance	2.5	KW
ASTRO TUG	1968-1975	1.80	KW
	G. & C.	.45	KW
	Lighting	.10	KW
	Envir. & Inst.	.60	KW
	App. Total Power (2 hour work period)	1.2	KW
UPCOMING PAYLOAD	1968 (Estimated)	1.0	KW
	G. & C. - Inst.	.8	KW
	Attitude Control	.2	KW
INTERPLANETARY TRAVEL	1971-19xx	300	KW

4.0 - DESCRIPTION OF RECOMMENDED SYSTEMS

The three types of power systems presently available for the OLO program are nuclear, solar, and chemical. Despite certain limitations, a nuclear system is recommended for primary power for the Orbital Launch Facility. Chemical systems are recommended for all of the other OLO vehicles, and also as the standby (emergency) system of the OLF. Chemical (Hydrogen - Oxygen) systems are attractive in these applications for durations up to two or three weeks. At present, solar systems are not recommended but considerable development work is being conducted relative to this type of system and it is possible that solar systems could become attractive if a significant advance is made in this field.

The system recommendations for the OLO program are based on studies conducted by the Vickers Division of Sperry Rand over the past four years and continuing at the present time. As previously outlined, these recommendations are based on the feasible systems which could, with reasonable development, be ready within the program schedule.

4.1 OLF Secondary Power

To provide a base line with which to measure the performance and weight of various secondary power systems, 30 KW and 60 KW power output levels have been assumed. These values are convenient because the Snap 8 nuclear power plant is being specifically designed for these two power outputs. Other power sources can readily be scaled to these values, but modifying the Snap 8 outputs would require extensive investigation into the basic design of the Snap 8 with attendant confusion.

4.1.1 Nuclear System - The Snap 8 power supply is under development by the AEC and NASA. This power plant for 30 KW and 60 KW operation weighs 1500 lbs and 2500 lbs, not including the shielding required for personnel protection, and employs 7.0 kg fuel to produce 30 KW with a probable life of one year. The shielding required varies with the distance between the OLF and the Snap 8 supply, the amount of time crew members will remain in orbit, and which sets of figures are to be believed as being the maximum radiation dose which humans can withstand and not suffer severe permanent damage. This shielding estimate varies from 10,000 lb for spot shielding to 36,000 lb for hemispheric shielding. For purposes of com-

parison with other APU's, the more conservative weight of 36,000 lb for the total Snap 8 weight will be assumed.

Shielding constitutes an appreciable fraction of the total mass of the orbital launch facility. Because the crew will spend a considerable proportion of their time outside the facility, adequate protection from the nuclear reactor can be obtained only if most of the reactor shielding is located around the powerplant, which is separated from the facility, rather than around the facility itself. Unfortunately, this precludes employment of the reactor shielding in double service; as a shield against both radiation from the nuclear reactor and radiation from the solar bursts. Reactor shielding weight does not serve any additional purpose that mitigates its undesirability. Therefore, trade-off studies would tend to show great advantage to a very small shielding weight.

Operation of the Orbital Launch Facility requires regular rendezvous with incoming supply vehicles carrying both men and equipment. The close proximity of the nuclear powerplant to the facility (approximately 50 feet), and the dependence upon maintaining this distance from the reactor creates a hazard during the final stages of rendezvous. A larger separation of the reactor from the facility is desirable to simplify the rendezvous problem.

Future orbital space flights may show that man cannot function effectively in a weightless environment for extended periods of time greater than approximately several days. This circumstance would require the generation of artificial centrifugal gravity force by rotating the facility. Other experiments on men have shown that prolonged rotation at rates greater than 1.7 RPM produce nausea by disturbing human metabolism through the balancing mechanism in the middle ear. Therefore a space facility designed to generate a "safe" artificial gravity rotates at approximately 1.0 RPM on a radius of approximately 6,000 feet. If the biological nature of man proves to be such that prolonged weightlessness must be avoided, then the facility design problems

would be greatly aggravated. However, it is possible to take advantage of a centrifugal configuration to greatly reduce the weight of the shielding required for the nuclear powerplant and reduce the radiation hazard involved in rendezvous operations with the facility. Fig. 4-1 illustrates the type of design that might solve the above problems in an advantageous manner.

The technique of Fig. 4-1 is only one possible configuration which might be desirable under specific circumstances. The entire field of shielding weight, structural arrangement, rendezvous and safety offers both opportunities and requirements for extensive investigation. This is a complex of problems which may require considerable effort in future studies.

Shielding requirements will have a great effect on weight in orbit, but the basis for selection is not well defined at present. Therefore a study in depth to determine merits of various facility arrangements and different shielding is recommended.

4.1.2 Chemical Secondary Power - A highly efficient 30 KW chemical system consuming only 1.0 pound of fuel per KW - HR. will in a year's time burn a weight of fuel which is many times greater than the weight of the nuclear reactor plus its shielding.
Wt. of Fuel = 30 KW x 365-1/4 days x 24 hrs/day x 1.0 fuel/KW-HR.
Wt. of Fuel = 263,000 pounds for one year.

This weight is enormously greater than approximately 36,000 pounds for the weight of the Snap 8 nuclear power plant plus shielding.

A chemical system would be more desirable as standby power for a nuclear APS. Because the reliability in space of the Snap 8 systems is not established, the need for a chemical standby power system for the nuclear APU of the 36-man orbital launch facility cannot yet be determined. Assuming that this concept is desirable, some representative system weights have been calculated and are presented below for a standby system of 48-hour duration.

System Component	Weight-lbs	Volume-cu.ft.
Oxygen fuel and tankage	2760	29 ft.^{3}
Hydrogen fuel and tankage	520	63
Engine (60 in.^{3} displacement, 90 hp)	80	2
Alternator	120	2
Plumbing	25	
System Totals	3505 lbs	96 cu.ft.

A similar engine using storable bipropellants may be necessary if long-term propellant storability is necessary (1 - 5 years storage). Some initial fuel weight would be sacrificed for storability.

4.1.3 Solar Power Source - There is a need for an alternate type of power plant in case it becomes very desirable to eliminate the nuclear unit from the design. Possible substitutes are various solar power plants described below:

Photovoltaic Power Systems - semiconductor solar cells which absorb light quanta and convert it to electricity within a single crystalline cell.

Photoemissive Solar Converters - large photo sensitive surfaces which eject electrons to be captured on an adjacent screen electrode. They employ some of the technology of both photo-multiplier and image intensifier tubes.

Thermoelectric Solar Converters - a semi-conductor thermoelectric material is employed to generate power from temperature difference between a pair of specially coated surfaces, one of which is heated by the sun.

Solar Dynamic Systems - Rankine cycle turbo-electric generators using the sun as a heat source, radiation to space as a heat sink, and usually exotic materials to secure high performance under difficult conditions.

Solar Thermionic Systems - generate thermionic diode power when the cathode is heated by sun's rays from a solar concentrator. Vacuum type diodes are now on the market and cesium vapor diodes with higher efficiency will soon be available.

All high-power, high-performance solar power plants have large reflectors called solar concentrators whose purpose is to concentrate the sun's rays on the solar energy conversion equipment. A 30 KW installation requires a large unwieldy solar collector on the order of 64.0 feet in diameter. In addition, the collector must be aimed at the sun with an accuracy ranging from

several degrees down to approximately 0.1 degree depending on the choice of system. Photovoltaic, photoemissive, and thermoelectric systems require several degrees accuracy, solar dynamic systems require approximately 1.0 degree accuracy and solar thermionic converters require the maximum accuracy approaching 0.1 degree.

Most of the weight of a solar static system designed for satellite service around the earth is in the storage batteries which supply power while the facility is orbiting through the shadow of the Earth. Data on the component weights of a solar cell power plant orbiting about the Earth has been obtained from WADD Technical Report 60-699, Energy Conversion Systems Reference Handbook, Volume IX - Solar System Design by W. R. Menetrey. Figure 4-2 from the report, shows the weight of the component parts in pounds/KW versus satellite altitude for different satellite altitudes with the equipment available in 1962. At the Orbital Launch Facility, altitude of 300-400 miles, system weight is a maximum. Fig. 4-3 extends the above data to the future with total weight curves for both 1965 and 1970. The startling drop in total system weight from 800 pounds/KW in 1962 to 150 pounds/KW in 1970 is a projection based primarily on reduction in battery weight. This is expected to occur either as a result of the use of new types of batteries such as fuel cells, or improvements in reliability in presently known batteries such as silver-zinc or nickel-cadmium batteries. Present reliability of nickel cadmium batteries limits the depth of discharge to 10.0% of maximum for the repeated cycling application characteristic of relatively low altitude Earth satellites. Therefore, improvements in battery reliability without any increase in maximum battery charge could produce the future performance increases projected in Fig. 4-3. This information on battery development applies to all solar static power plants in low satellite orbits where the storage batteries are the primary component weight in the system.

Note that the above weights for a solar cell system give a total weight for a 30 KW solar cell system of 25,000 pounds by 1962, 4500 pounds by 1965, and 2100 pounds by 1970. However, it is questionable whether the above weights per KW can be maintained at a power level as high as 30 KW for a solar cell system which is inherently suited to lower power levels. But, if the projected development of these systems takes place over the next few years, then they would provide a power plant appreciably lighter than Snap 8 with its shielding.

The solar dynamic system appears to be most suitable for the power level of very approximately 4 KW to 20 KW. Proper comparisons for this type of power plant can be obtained only by employing complete designs. An excellent example of this type of solar power plant is the 15 KW Solar Mechanical unit which WADD is now funding. This is a 15 KW solar power plant weighing 819 pounds and using a 40.5 ft. solar concentrator. It employes two different chemicals for thermal storage when the satellite is in the Earth's shadow. To compare this system with other techniques of power generation it is most convenient to evaluate on the basis of weight per unit of power. This 15 KW unit weighs 55 pounds per KW, the lowest of any non-chemical system considered. This unit, however, in addition to the large solar concentrator, actually will not be ready for approximately four to six years (if there are no major delays), and the power output is 15 KW instead of the 30 KW required by the Orbital Launch Facility. There are three alternatives to the latter problem:

1. Use two 15 KW units

2. Attempt to purchase a 30 KW unit with a considerable increase in lead time.

3. The 30 KW power requirement, which is only an estimate, could be reduced to 15 KW supplied by a single solar dynamic unit weighing 819 pounds. It has been estimated by the cognizant personnel at WADD that a 50 KW unit would weight about 45 to 50 pounds per KW. In addition, they have estimated that this specific weight is a minimum for this power level, and would rise for either an appreciably higher or lower power level.

4.1.4 Comparative Reliability -

Although the Snap 8 nuclear power plant is intended to have an inherent reliability capable of insuring at least one year of operation, it must be shut down for all maintenance, and even then, must be handled with manipulators through a protective shield. Any reliability of power continuity during maintenance periods almost certainly will require the use of at least two Snap 8 power plants with the increased weight. An alternative approach for power continuity would be the use of a H_2-O_2 or storable bi-propellant system. Solar cells suffer slow deterioration over a long period of time caused by bombardment by X-rays and gamma rays particularly during severe solar flares. However, this deterioration is sufficiently slow such that over a one year period the reliability should remain quite high. Because the solar cell power plant uses many solar cells, whose weight is not a major portion of total system weight, it is possible to greatly increase the reliability of the solar cell bank by wiring them in a mixed series - parallel arrangement to obtain a great increase in redundancy. The storage battery portion of the system can be easily separated into many small units with sufffficient switching equipment to remove defective batteries from the circuit. The static electrical conversion and regulation equipment possess more than adequate life. In addition, the entire system can be readily repaired by the crew without any special precautions by the crew.

The solar dynamic system is a mechanical system constructed from materials that are not deteriorated by space radiation. Again, the system can be readily repaired by the crew without any special precautions against radiation. Two or three 15 KW power units would permit one unit to be shut down for repairs while the other unit(s) provided continuity of electrical power.

4.1.5 Selected System - The Snap 8 system is the only system which presently has a reasonable development schedule for dovetailing into the overall OLO schedule despite its disadvantages. The disadvantages of present nuclear power plants are that they: are relatively difficult to maintain; present a political problem in their operation; require heavy shielding; have what is perhaps as unnecessary radiation hazard to the facility crew. Although other power plants such as the 15 KW solar dynamic system appear vastly more attractive from the weight and size standpoint than either solar cell or nuclear, the very severe limitation of 15 KW imposes stringent limitations on the OLF equipment. Fig. 4-4 shows the comparative weights of the candidate systems for meeting the power requirements of the OLF. As shown on this curve, assuming a 48 hour requirement for emergency power, the hydrox engine system offers a large weight advantage over all other candidate systems.

The use of two or three solar dynamic systems is feasible but requires an operational analysis of the use of the area immediately surrounding the OLF due to the space required for a large solar concentrator. Sperry Rand proposes to study the details of Orbital Launch Facility operation to determine whether a large collector would interfere with the proper functioning of the facility. The study would determine what changes are necessary in facility operation to accommodate the operation of the solar power system. A comparison will be made to determine the net change in the usefulness and cost of the facility for the new configuration.

Solar Power plants are presently behind nuclear power plants in larger power outputs, require large solar collectors which must be oriented to the sun, and probably will never be suitable for very large power outputs. For the higher power levels required by the Orbital Launch Facility, solar power plants are in the early planning or development stages. Their projected performance and reliability five or more years from now contains a measurable element of chance. However, in several years, development may improve to the point where solar power plants will possess a high probability of success. At the present time the solar power art is developing so rapidly that continuous observation is required to maintain data on a current basis. As a specific follow-on task it is recommended that Sperry Rand personnel be sent to various laboratories where the latest techniques are being developed and then prepare a systematic, up to date comparative summary on the state-of-the-art of solar power plants. This information would be necessary prior to any fundamental decisions involving the choice of secondary power sources on the Orbital Launch Facility.

4.2 Space Craft Secondary Power

Vickers studies have revealed that a dynamic hydrogen-oxygen auxiliary power system is capable of achieving performance equal to, or better than, that achieved with fuel cells and at lower system weight. In addition, the chemical dynamic system can provide two desirable functions which have also made fuel cells attractive: (1) operate on low pressure boil-off gases from the main propulsion system tanks, and (2) provide drinking water for the crew.

4.2.1 Candidate Systems - A typical hydrox APS concept using the Vickers internal combustion engine is shown schematically in Fig. 4-5. In this system, an electrical AC generator is mounted on the hydrox engine, eliminating any need for a gearbox. Hydraulic power can also be provided if required by including an engine driven hydraulic pump in the system.

The regenerative heat exchanger seen in Fig. 4-5 extracts useful energy from the internal combusion engine exhaust by cooling it with incoming cryogenic propellants. The engine cooling jacket also utilizes the heat sink properties of the liquid hydrogen propellant. Since the propellants offer inherent engine cooling capability, it may be unnecessary to equip the engine with a radiative cooling system. The system shown in Fig. 4-5 is designed to operate fuel (hydrogen) rich.

More sophisticated hydrox engine systems may be designed to operate at a stoichiometric combustion ratio. Injection of a portion of the exhaust water into the engine combustion chamber and regenerative heat exchanger at several levels will be necessary to achieve the desired BSPC of 0.60 lb./hp.hr. in this advanced system. The possibility of operating the hydrox powerplant on propellant boil-off gases from the main propulsion system has been investigated, and the concept appears not only quite feasible, but practical as well.

Advantages of a dynamic hydrogen-oxygen power plant over fuel cells include:

1. Lower installed weight per kilowatt - 32 lbs. per KW in a 2 KW system. (Compared to over 100 lb./KW for fuel cell).

2. Growth potential: lower installed weight per kilowatt - hour, (Less than 32 lbs./KW in system larger than 2 KW) lower brake specific propellant consumption (BSPC), (.9 lbs./KW-hr. compared to 1.0 lbs./KW hr. for fuel cells).

3. Fewer problems with zero-gravity conditions.

4. Capability of achieving higher thermal efficiency.

5. Early availability as a result of using proven concepts and hardware.

6. Low development cost due to parallel R & D programs now in progress.

7. More useful form of electrical output (high voltage AC instead of low voltage DC) hence lower conversion losses.

In general, the bulk of a solar cell system is very high. Solar cells are susceptible to meteoritic bombardment and cosmic radiation, and are difficult to handle. The specific advantages of the H_2 - O_2 system over conventional solar cells are:

1. The solar cell power system is presently much heavier than the combustion APU and it appears that by 1965 there will still be a weight advantage to the combustion unit.

2. Operation of a solar array requires continuous orientation to the sun during the flight.

3. Protection from shock during the lunar landing and take-off operations is required. In particular, there is a real risk that the solar array could be damaged by a hard landing on the Moon's unknown surface.

4. Operational flexibility of the combustion APU is much greater because it is capable of operating in the absence of sunlight. For scientific purposes it permits investigation of lunar conditions during the lunar night time as well as day time. Night time operation may even be required if solar flares or solar radiation levels are a problem. Sunless operational capability provides an extra safety factor in case the S/C lands in deep dust, or the shadow of a tall mountain, or some other malfunction strands a crew on the return launch from the moon. Since the latter occurence would probably take place toward the end of a daytime period the crew would face a problem in generating enough auxiliary power to maintain normal functioning of the vehicle internal environmental and to communicate with a rescue vehicle. If the return launch rocket were also an H_2-O_2 system the crew could use its own fuel supply for survival. A rescue vehicle for this operation would probably operate during lunar nighttime if the orginal exploration were a daytime expeditiin. Therefore a combustion APU is necessary.

5. The solar cell system contains other difficulties which are probably solvable but still present considerable difficulties. Some of these problems are erection of large solar arrays, destructive effects of meteorites, and deteriorating effects of solar flares on solar cells.

Studies have been made on a regenerative power system concept, indicating the capability of achieving a BSPC of 0.74 lbs./hp-hr with a fuel rich mixture ratio of 1.5 (O_2/H_2). An advanced system using partial recirculation of exhaust products is contemplated which will be capable of a BSPC of 0.60 lbs./hp-hr, at a mixture ratio of 8:1 (stoichiometric combustion, giving 100% steam exhaust). Rapid availability of such a system is assured by utilizing proven thermodynamic concepts and power cycles differing from conventional power plants primarily in the choice of fuel and oxidizer. Operation at conservative conditions of temperature, pressure, and speed further assure rapid availability of reliable components and systems.

A program is recommended which includes analysis and optimization studies, feasibility and performance testing of critical components, similar testing of a complete "breadboard" auxiliary power system and development and flight testing of prototype systems for the OLO vehicles.

Vickers is currently conducting a research and development program for Aeronautical Systems Division, Wright-Patterson Air Force Base, on a hydrogen-oxygen internal-combustion piston engine. A flight type version of this engine is proposed for the OLO program.

Figure 4-6 shows a flight design of a similar type engine. This current Air Force-sponsored program will augment the OLO program by providing basic data in the area of cylinder heat rejection, piston friction minimization, practical oxygen injection, and combustion characteristics of hydrogen and oxygen in internal combustion engines.

The properties of engine exhaust products and incoming propellants can be used in a variety of ways in an APS that is integrated with life support and electronic equipment cooling systems in the space vehicle. Such an integrated system would save many pounds of overall weight.

The superheated exhaust and/or the cryogenic propellants can provide temperature regulation through heat exchangers. Water condensed from the exhaust will be available for drinking, a source of moisture for cabin humidity control, or may be stored for use as a coolant during re-entry. For these reasons, Vickers believes it worthwhile to investigate this integrated system for the OLO program.

A regenerative power system has been analyzed for basic feasibility and is presented in the following paragraphs. It was assumed in this analysis that no heat sink would be available in space other than the incoming cryogenic propellants. To make a fair comparison between fuel cells and combustion engines, it is necessary to study a complete spectrum of systems operating with oxygen/hydrogen ratios between 1.0 and 8.0 (the latter ratio representing the oxygen/hydrogen ratio of a fuel cell). There may well be an optimum combination of heat sinks and oxygen/hydrogen ratios which result in the most suitable power plant system. It is important to note that reciprocating power plant systems can be designed to operate at oxygen/hydrogen ratios of 8:1 with thermal

efficiencies of 60% or better; the resulting fuel consumption should be directly competitive with practical fuel cells. The system would be far lighter than silver zinc batteries for duration over one hour and lighter than solar cells for durations up to two to three weeks.

4.2.2 Assumptions - Because only a limited amount of pertinent information is available to formulate the bases for preliminary studies of possible auxiliary power systems, the following assumptions have been made to clarify the problem statement:

1. The auxiliary power system is designed as a self-contained unit.
2. Reliability is achieved by the use of redundant systems with weight and efficiency calculations taken into consideration.
3. A high degree of mechanical design sophistication is considered permissible.
4. A progression of the state-of-the-art will result from any further study.
5. The influence of solar heating (which has been omitted from this preliminary study) can be made available.
6. Propellant tanks and engine lubrication systems will be designed to operate in a zero-g environment.

7. APU operation may be monitored by personnel and/or remote servo systems, and the power units may be completely regulated during flight.

8. No special "overload" components are required. Power over the full range can easily be supplied by a single hydrox engine.

4.2.3 - Parametric Studies

During the research and development phase of the internal combustion engine program, a continuing study will be made of the fundamental parameters in an effort to minimize development time.

1. Weight and Volume - Materials, packaging techniques and hardware design will be investigated for minimum weight and space requirements.

2. Power Requirements - Sufficient growth potential will be incorporated into all candidate systems to allow for future growth. The most effective means of creating the desired growth potential will be determined.

3. Availability and Development - An internal combustion engine system can be operable within required availability dates. Proposing two engines of different levels of development allows considerable flexibility in design schedules and enables the power system to remain competitive for many years.

4. Reliability - The low fixed weight and stable construction materials inherent in an internal combustion positive displacement APU give it an advantage over radiation and temperature sensitive competitors such as fuel cells and batteries.

5. Maintainability - The elimination of maintenance and servicing problems will be considered as a basic design goal.

4.2.4 System Parameter Sensitivity

The basic design parameters may be changed or modified as the problem becomes better defined and more information becomes available. The sensitivity of the basic APU to any change in requirement is a major factor to consider in the selection of an auxiliary power system. A discussion of the effects of requirement and state-of-the-art changes on the H_2-O_2 internal combustion engine APU follows:

1. Weight and Volume - The weight of the system is extremely flexible, and a weight reduction may be effected at the expense of some design complexity. Volume requirements are as low as any competitive system and the promise of future efficiency improvements can further reduce fuel weight and volume.

2. State-of-the-Art Development - Battery development is proceeding strongly at present but a major breakthrough in the fields of batteries, fuel cells, and other direct conversion devices will be necessary before the relative competitive position of the positive displacement internal combustion engine is changed. Although continuing refinements in all systems are anticipated, the hydrox internal combustion engine is expected to remain a major candidate.

4.2.5 Space Craft Auxiliary Power System

The requirements for an auxiliary power unit for the 3-man Space Craft were given in Table B. Enough growth potential is incorporated to permit the use of this system in a 6-man vehicle with minor modifications. Two system design levels are discussed: (1) incorporating a simple engine that will be available in 1963, and (2) a more advanced model, with an engine benefiting from stoichiometric combustion, that will be available in 1965. Both systems are compared with a fuel cell system and with a solar cell system.

The lunar rest phase of the Space Craft mission is of unspecified duration in Table B. Therefore propellant and tankage weights for the Space Craft power system are expressed in terms of lunar rest time to provide flexibility in applying the results to a variety of missions.

The following equation gives electrical energy available from the system:

$$E = 247.3 + 33.6t \quad \text{kilowatt hours,}$$

where t = number of days at lunar rest (time spent at rest on surface of moon)

Graphically, this equation can illustrate the electrical energy and weights as a function of lunar rest time in days as shown in Figs. 4-7 and 4-8. The support calculations for these weights are shown in the systems breakout chart (Table C).

TABLE C

SPACE CRAFT POWER SYSTEMS CALCULATIONS

	Advanced I. C. engine	Simple I. C. engine	Fuel Cell
Oxygen	210+28.4t	170+23.2t	216+29.3t
Oxygen tank	42+5.7t	34+4.64t	43.3+5.9t
Hydrogen	26.2+3.6t	113.5+15.5t	27.0+3.7t
Hydrogen tank	21.0+2.8t	90.8+12.4t	21.7+2.9t
Total Propellant	299.2+40.5t lbs	408.3+55.74t lbs	308.0+41.8t lbs

Adding the above weights to the engine hardware,

Power Units(3)	24	18	216
Alternators	45	45	---
Pumps	3	3	3
Plumbing	10	10	10
Total	381.2+40.5t lbs	484.3+55.74t lbs	537.0+41.8t lbs
Volume (Est.)	9.1+ 1.2t ft^3	27.5+ 3.7t ft^3	10.4+ 1.2t ft^3

t = number of days spent at lunar rest.

Solar cell system is composed of:

Cells and structure	280 lb
Glass	245 lb
Storage	325 lb
	850 lb

As is shown in Fig. 4-8 for lunar stay times of less than one week, the simple hydrox engine and fuel cells are comparable to each other with solar cells gaining the ascendancy for periods from one to two weeks. However, at the end of two weeks maximum, the solar cell becomes inoperative for the two week lunar night.

The advanced H_2-O_2 engine is superior to all three above energy sources from the weight standpoint for lunar rest times up to almost 2 weeks.

The solar cell weight given in Table C has assumed a 30% decrease in weight over 1962 state-of-the-art solar cell development. Estimates for weight decrease run as high as 55% by 1965. The 30% decrease is considered a reasonable reduction factor for this report. The battery weight given is not sufficient to maintain the required KW output for earth orbital operations but is adequate for temporary shut downs required for mid-course and lunar terminal maneuvers.

For reference purposes, Fig. 4-9 is presented, showing the comparative weights of Space Craft candidate power systems as a function to total mission duration. It is seen that the hydrox engine systems offer considerable weight saving of about 14 days duration compared to other candidates. In addition, it provides greater flexibility than fuel cells and solar cell systems.

Efficiency - A possible brake specific propellant consumption of 0.60 lb/hp-hr is assumed for the most efficient operating level of the advanced internal combustion hydrogen-oxygen engine. This number is based on a stoichiometric combustion ratio (O_2/H_2 = 8/1). For the simple internal combustion engine, a BSPC of 0.74 lb/hp-hr at a mixture ratio of O_2/H_2 = 1.5/1 is assumed. The minimum BSPC of the fuel cell system is 0.65 lb/hp-hr at stoichiometric mixture. In Fig. 4-10 the off-design point efficiency of these three systems is shown. It is seen that overload operations on one engine will raise the BSPC

of the advanced and simple IC engines to 0.63 lb/hp-hr and 0.77 lb/hp-hr, respectively. No correction was made for fuel cell failure since fuel cells were assumed to operate singly at the optimum load.

The overall thermal efficiency of the H_2-O_2 system is over 70 percent which is considerably higher than presently anticipated fuel cell efficiency.

An electrical conversion efficiency of 90 percent was assumed for all systems. This represents an alternator efficiency in the H_2-O_2 engine systems and a DC-AC inverter efficiency for the fuel cell.

Tankage - The total propellant weight was calculated from the energy requirement given by the equation for electrical energy, using the applicable BSPC. Tankage weight was then calculated by using the following assumptions:

Oxygen tankage weight = 20% oxygen weight

Hydrogen tankage weight = 80% hydrogen weight

Tank weight estimates are based on the results of Vickers studies on advanced concepts in zero-g cryogenic tankage for space missions. For a discussion of the rotating fluid tank concept, see paragraph 6.4.

Engine Weight - Engine weight was calculated for the two hydrox engine systems as follows:

Continuous duty BMEP of 200 psi at rated speed was assumed. The greatest power output necessary is 3.6 hp (one engine, generating ?.4 k @ 90% efficiency). If rated engine speed is 6000 rpm, an engine displacement of 1.2 $in.^3$ results.

Detailed design studies and prototype engine development work conducted by Vickers indicate that a flight design engine of this performance level would weigh 7.6 pounds. A conservative assumption of 8 lbs/motor was used. This weight includes engine speed controls and accessories but not propellant pumps, plumbing, etc. which are listed separately in Table C.

4.3 Astro Tug APS

Two alte rnate auxiliary power systems for the space tug are compared below. A small cryogenic H_2-O_2 internal combustion system with two engines and fuel for two round trips is compared to a moderate-discharge-rate secondary battery system using two silver-zinc battery units. It is believed that either system will satisfy the power and reliability requirements for the space tug auxiliary power system.

TABLE D

	Power Source	
	Battery	H_2-O_2 Engine
Number of redundant systems	2	2
Conversion efficiency	90%	90%
Electrical power level	1.2kw	1.2kw
Duration	2 hrs	2 hrs
Fuel weight	------	6 lbs
Tankage weight	------	5.7 lbs
Fixed weight	100 lbs	32 lbs
Total Weight	100 lbs	43.7 lbs

The power requirements of the Astro Tug shown in Fig. 9-5 of the Vought design data, Volume 2, required a two-hour work duration and allowed a six hour battery re-charge. If a hydrogen-oxygen system were used it would offer a substantial decrease in turn-around time. The six hours would be reduced to a matter of minutes to refuel the six pounds of fuel weight required for vehicle operation. Fig. 4-11 shows a comparison between batteries and a hydrox engine system for both the Astro Tug and the Upcoming Payload.

Additional studies are required to establish the best methods of fluid transfer for orbital re-fueling of the Astro Tug. Vickers has conducted preliminary studies of advanced concepts in two

phase fluid handling techniques under zero G conditions which may be applicable.

4.4 Upcoming Payload Vehicle APS

According to recent information on the OLO program there is a distinction between retrieval and rendezvous of payload. The retrieval phase will take place within 500 feet of the platform or a designated position. However, rendezvous will be considered as the long range operation beginning at earth launch and ending at the retrieval phase.

On the basis of this data the requirements of an auxiliary power system have been calculated to provide one kilowatt of power for attitude controls, guidance, and other vehicle systems. It is estimated that this one kilowatt is approximately 800 watts of electric power and 200 watts of mechanical power or hydraulic power. The trade-off studies for attitude control show that 14 nozzles would be required for complete directional and attitude control for the upcoming payload. With the high reliability only 200 watts will be required for this nozzle system. The remaining 800 watts will be used for electronic equipment power. Although this is a one-shot affair, duration requirements are indeterminate. Fig. 4-11 shows a comparison between a silver-zinc secondary battery system and the H_2-O_2 system for both the upcoming payload and Astro Tug power levels.

Since the hydrox system growth potential is so much greater in regard to duration, it may be desirable to utilize an H_2-O_2 system in the upcoming payload if duration exceeds the tentative one hour requirement. The Astro Tug system could be used with minor modifications, eliminating costly development. A manned payload vehicle would render the hydrox system even more desirable.

4.5 System Options for Space Craft and Manned Astro Tug

Space electrical power generation systems being considered by Vickers include several optional approaches. The feasibility of each of the following approaches cannot be determined on the basis of the limited information presently available on system requirements and space operating conditions. Each approach, however, may prove valuable as a means of effectively integrating the APU into the overall vehicle.

4.5.1 Option I - Main Tank Propellant Boiloff - An auxiliary power system which uses boiloff gases from the main propellant tanks provides weight reduction in the following areas:

1. Total tankage
2. Total amount of hydrogen under storage
3. Main hydrogen tank insulation (since a higher boiloff rate is permissible)

A complete system design incorporating the use of boiloff gases would require detailed knowledge of the entire vehicle heat balance and the trade-offs between insulation weight and propellant weight. Even if boiloff rates are not critical, it may be desirable to utilize the main tanks in the interests of efficiency.

An auxiliary power system weight comparison considering fixed weight alone is shown below. This weight, taken from the S/C systems calculation, is independent of mission duration and illustrates the low fixed-weight advantage of H_2-O_2 internal combustion engine system.

Advanced H_2-O_2 engine(stoichiometric)... 82 lbs

Simple H_2-O_2 engine (fuel righ) .....76 lbs

Fuel cell .....229 lbs

A schematic arrangement of a power system capable of operation on propellant boiloff is shown in Fig. 4-12.

4.5.2 Option II - Integration of APS with Environmental Control System - Vehicle cooling was not included in the system description because available heat from the crew compartment and equipment has not yet been clearly defined. For an accurate study of an integrated Auxiliary Power-Environmental Control system, the following values must be specified.

1. Heat rejection from crew to the ECS.
2. Heat rejection from vehicle equipment to ECS.
3. Heat available to the APU from the ECS.
4. Required temperature levels in ECS for crew compartment and for various equipment.
5. APU output power which is returnable to the APU in the form of heat through an APU-ECS feedback loop.

These values are combined into an overall heat feedback ratio, usually represented as a percentage of APU output power. Heat can be removed by radiation, by using the APU hydrogen supply as a heat sink, or both, depending on mission duration, thermal efficiency or BSPC of the APU, heat feedback ratio and other factors.

For most missions that Vickers has studied, the heat capacity of the hydrogen supply (for an APU similar to the one described) is usually less than the total available heat feedback. Therefore, a radiator may be necessary for vehicle cooling. This is particularly true of an APS with high thermal efficiency since lower BSPCs tend to reduce the heat sink capacity of the stored propellants. A drawing of an APU incorporating heat feedback is shown in Fig. 4-13.

4.5.3 Option III - Exhaust Water Utilization - Water in a manned spacecraft is needed for:

1. Crew drinking water
2. Re-entry cooling
3. Crew cabin humidity control

Considerable water may be recovered from crew waste. If more is needed, condensation of the APU exhaust gases will provide an additional source at little or no weight penalty. A portion of the exhaust water can be condensed in the regenerative heat exchanger of a system such as shown in Fig. 4-14 by using the heat sink capacity of the incoming propellants. Additional water can be condensed by use of an additional heat sink in the form of a space radiator. A preliminary analysis has been conducted to indicate approximately how much exhaust water can be condensed for drinking purposes and/or other uses. Also, estimates of required radiator weight for complete condensation have been made. These are tabulated below:

System	$\%H_2O$ In Exhaust By Wgt.	$\%$ H_2O Condensed in Regenerator (Propellant Heat Sink)	Radiator Weight * For Complete Condensation Lbs/KW	Total Weight of H_2O Condensible Lbs/KW-HR
Fuel Rich (O_2/H_2=2.0:1)	75%	25%	6.4	1.1
Stoichiometric (O_2/H_2=8.0:1)	100%	5%	9.7	.9

* Radiator weights are based on a temperature of 492°R., heat rejection of 100 BTU/Hr. Ft.^{2} and a unit weight of 1 lb. per Sq. Ft. This analysis was based on conservative assumptions but does not allow for additional weight which may be necessary for successful Zero G operation of the condensing regenerator and/or the condensing radiator.

4.6 Reliability and Maintainability

4.6.1 Reliability - Manned space flight demands a high level of reliability that can be achieved by the use of redundant systems. The following gasic rules, taken from Reference 8.2 were used to determine the necessary redundancy:

1. An overall vehicle reliability of 0.85 requires an auxiliary power system reliability greater than 0.9999.

2. Separate, complete, and fully installed power units are desirable. The number of units depends on the reliability of each.

Well documented reliability figures for mechanical systems operating in a space environment will not be available for several years. However, if a single unit reliability of 0.95 can be attained by the proposed internal combustion hydrogen-oxygen engine, the system reliability for three parallel units is .999875. This number represents the probability that all three units will not fail during the mission. In actual practice, two units will be operating under reduced load at all times, thus increasing the reliability of an individual unit. Also, the unit (or units) not operating will not deteriorate with time within the time periods considered in this mission. These considerations lead to confidence that a three-engine APU will attain the desired reliability.

The one-engine reliability Figure of 0.95 has been estimated as turbine engine reliability under space conditions. This reliability will be easily attained or surpassed by a small reciprocating engine because seventy years of design technology experience may be brought to bear on all problems which may arise. The reliability of the engines used in the OLO vehicle APUs depends primarily on effectively incorporating the results of the development

effort and environmental testing effort into the final design.

A fuel cell system reliability of 0.96 is estimated in published reports. Three parallel systems are needed to obtain the required reliability level. Due to the high fuel cell system weight, this requirement imposes a more severe penalty on the fuel cell APU than it does on the internal combustion engine APU.

4.6.2 Maintainability - When human beings are available in a space vehicle, a great deal of complexity and system redundancy may be eliminated by effectively incorporating them into the system. Human decision-making with respect to instrumentation can eliminate complexity. All machine decision-making is inherently and excessively redundant since it must be reduced to a complex series of go-no-go situations, precluding judgment. The elaborate environmental controls needed to sustain human life can be made more efficient if waste heat and by-products are effectively re-cycled.

A standby power system for the orbital launch facility can be designed to be repaired, serviced, and dismantled by personnel on board. Spare parts and required tools could be made available to the facility.

It would be possible to relax the basic reliability requirements of the assembled power supply unit if it could be easily serviced and overhauled by men of moderate skill working under adverse conditions. The success of non-professional automobile mechanics over the last fifty years attests to the maintainability of an internal combusion engine under these conditions (a contention that cannot be held for fuel cells or turbines).

Space Tug APUs could also be made repairable and easily replaceable or exchangeable. The weight savings

offered by an internal combustion engine system over a battery system for long operative periods depends on reducing the fuel weight, spare parts, extra motors, and tool weights for the internal combustion engine to less than the battery weight plus the weight of fuel used to charge the batteries.

4.7 Safety Aspects of Hydrogen-Oxygen Systems

Although there was considerable concern about the safety of H_2-O_2 Systems three or four years ago when they were first considered, many of these fears have proven to be exaggerated as actual experience has been accumulated. Experience gained in the Centaur Program indicates that liquid hydrogen is no more difficult or hazardous to handle than LOX as long as proper precautions are taken, such as adequate venting. Hydrogen-oxygen propellants have been selected for both power generation and reaction control on the manned Dyna Soar vehicle. In addition the liquid hydrogen also forms the heat sink for vehicle equipment and crew compartment cooling. The Air Force is funding at least three additional hydrogen-oxygen power system development programs which are applicable to future manned space vehicles.

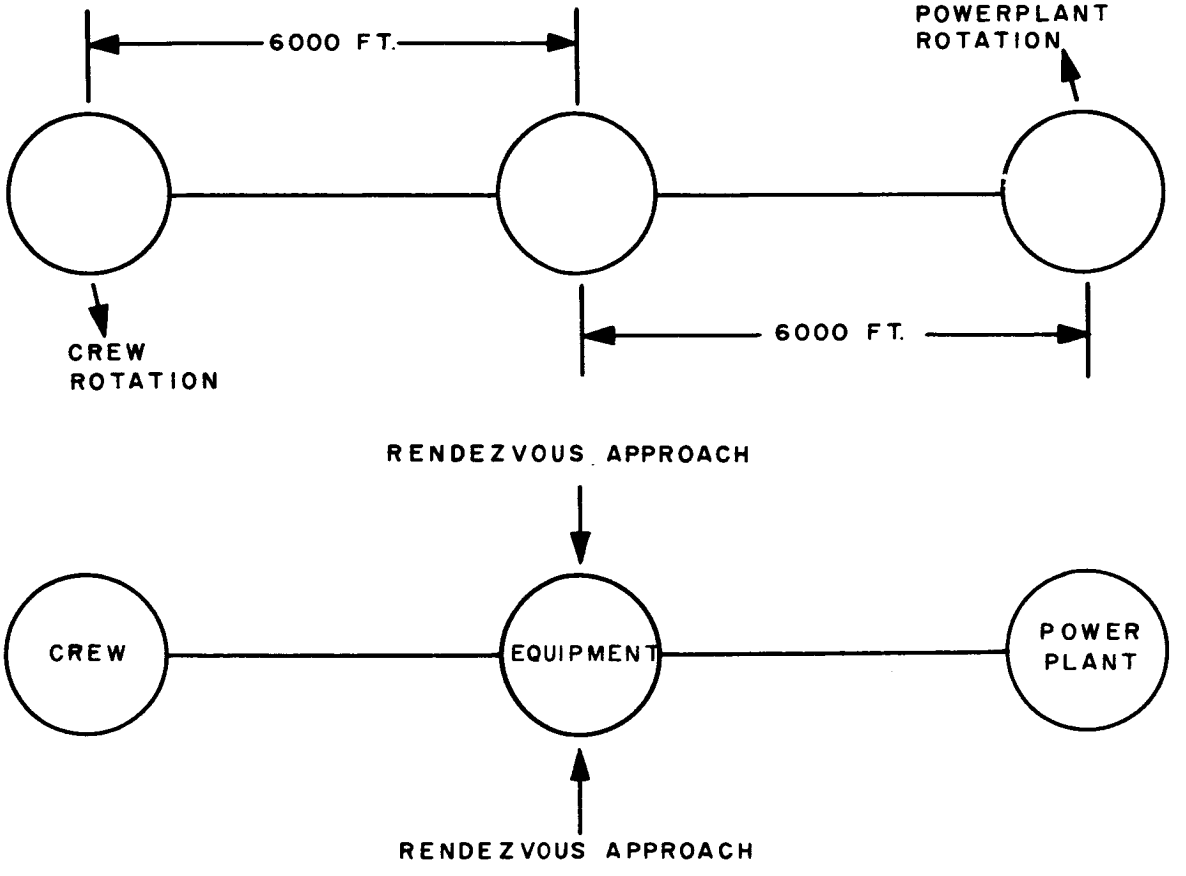
Consideration by N.A.S.A. of hydrogen-oxygen fueled boosters and fuel cells for the manned Apollo vehicles also indicates confidence by that agency that any safety problems can be readily resolved.

Leakage of hydrogen into a space environment(vacuum) would not represent a hazardous condition unless there were also an oxygen leakage in the immediate proximity. This condition would represent a double failure of a propellant storage or feed system. Combustion of the leaking gas could occur only if a

source of ignition were present, e.g., a spark, or hot surfaces. Unless combustion occurred in a confined space the result would probably be a fire rather than an explosion.

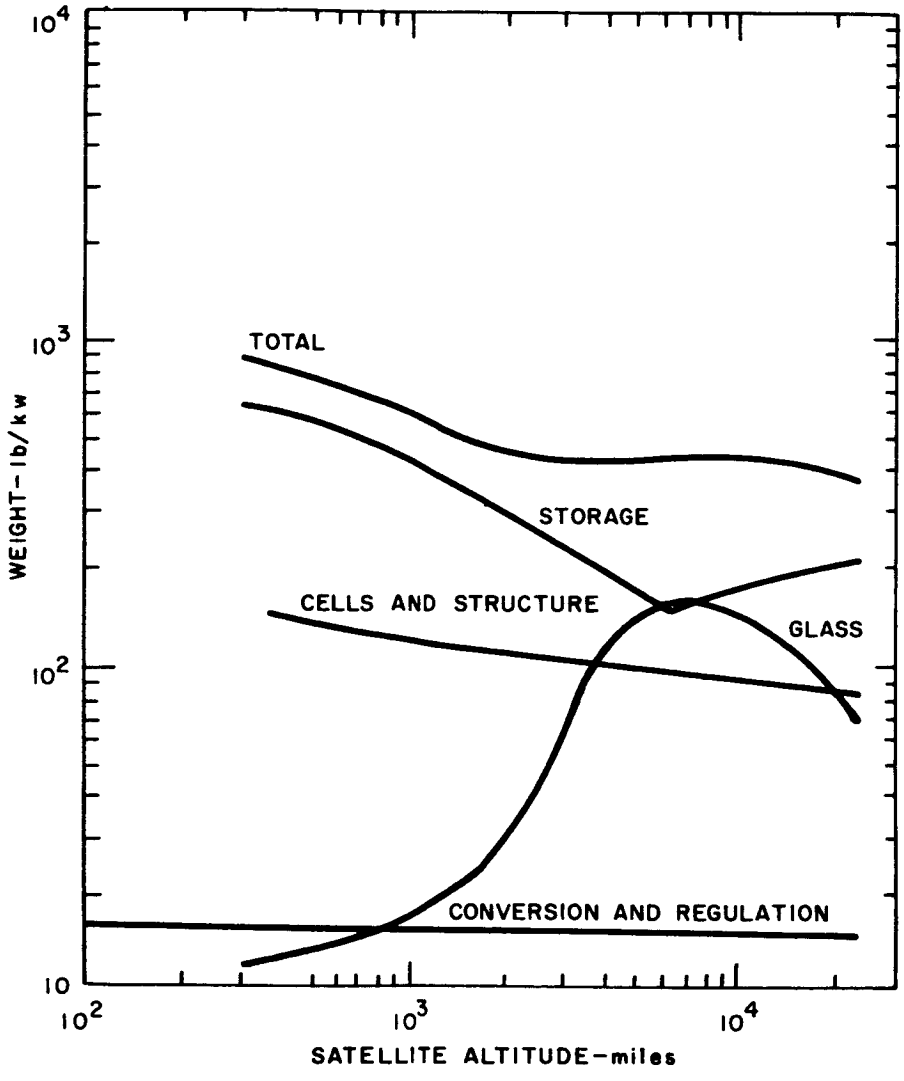
Of course precautions must be taken to prevent accumulation of combustible mixtures in lines, gas generators, valves, etc. Mixing of the propellants should occur only in locations where combustion is desired and controllable such as in the system combustion chamber. A continuous source of ignition would burn any combustible mixtures before dangerous amounts could accumulate.

From a safety aspect, the internal combustion hydrox engine has the advantage that the propellants are not mixed before entering the combustion chamber. Hydrogen and oxygen are injected separately into the cylinder where mixing, combustion and expansion then occur.



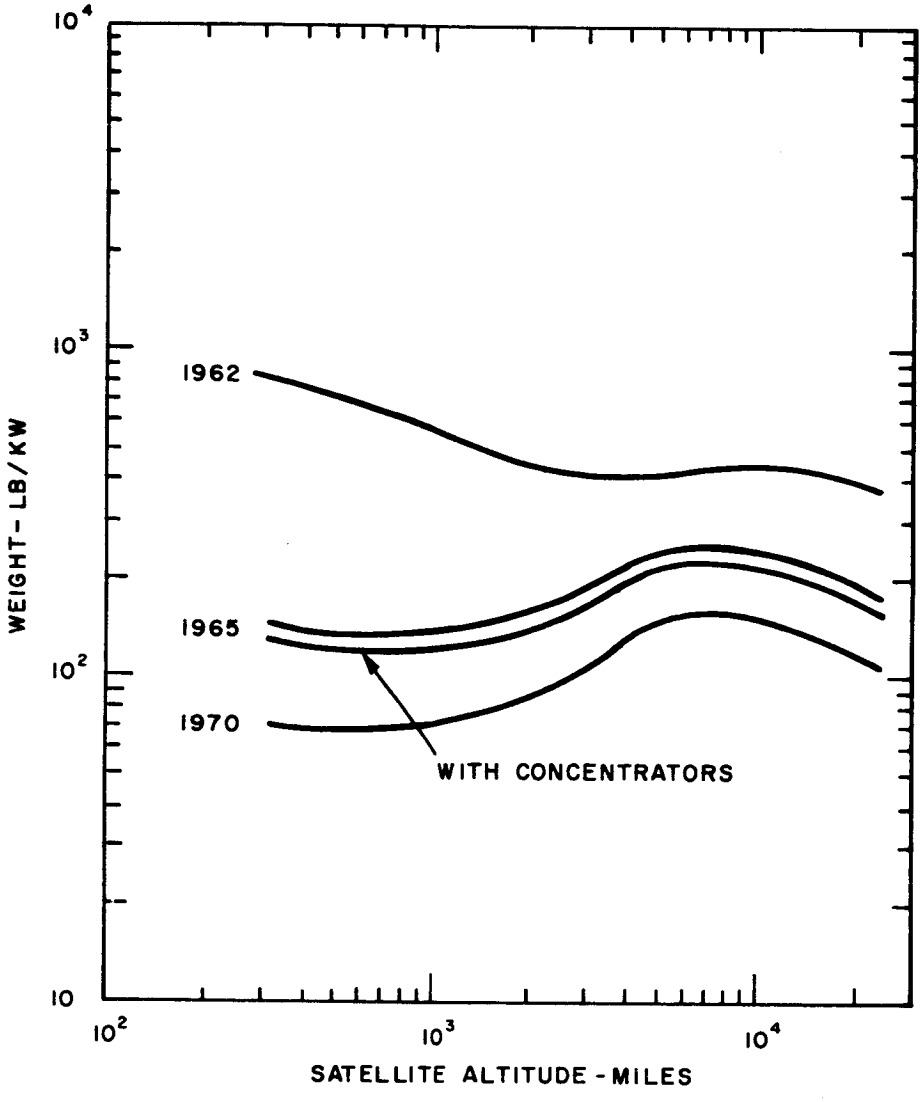
ORBITAL LAUNCH FACILITY WITH SAFE ARTIFICIAL GRAVITY

Fig. 4-1



PHOTOVOLTAIC SYSTEM WEIGHT - 1962

Fig. 4-2



PHOTOVOLTAIC SYSTEM WEIGHT

Fig. 4-3

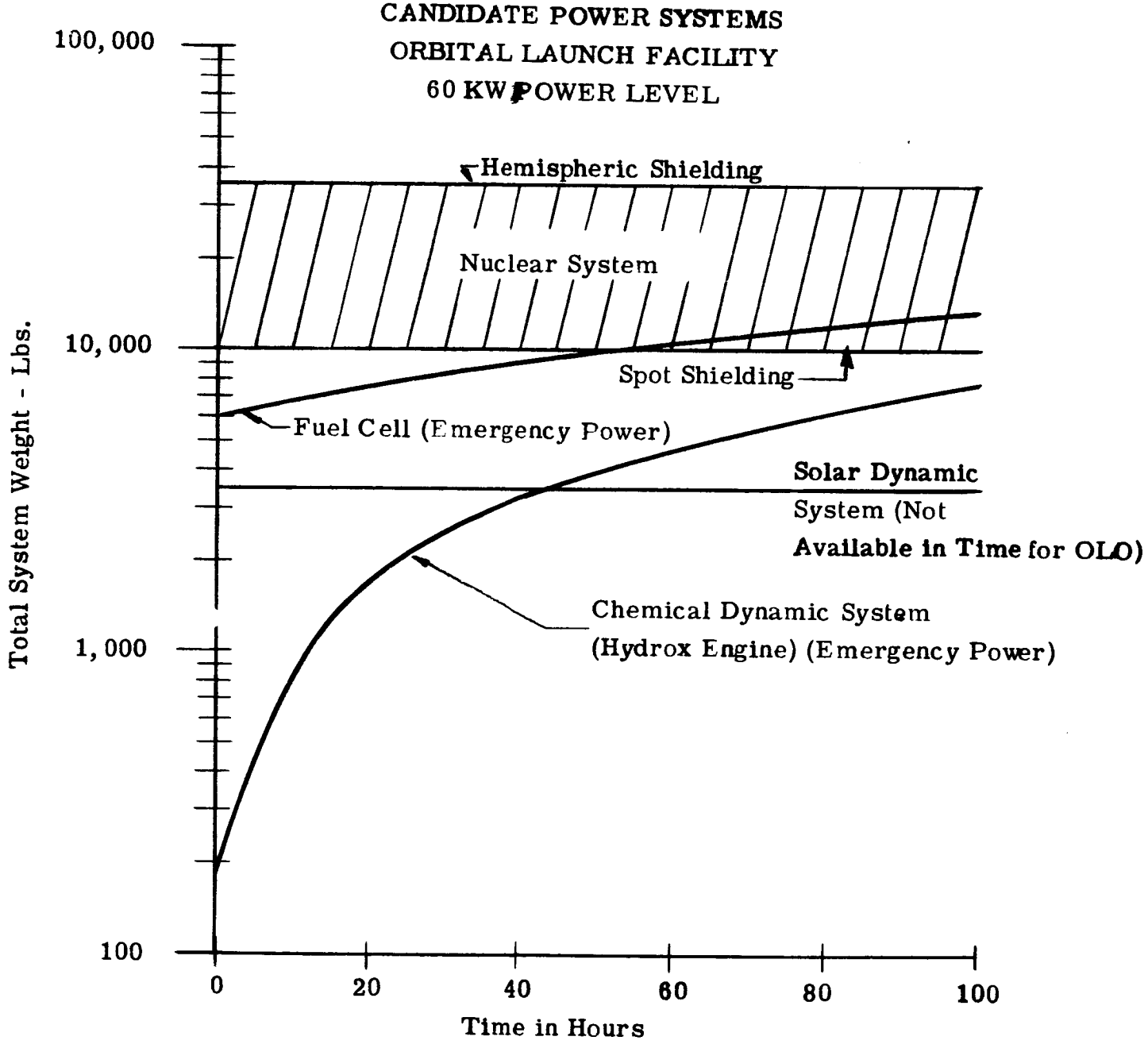
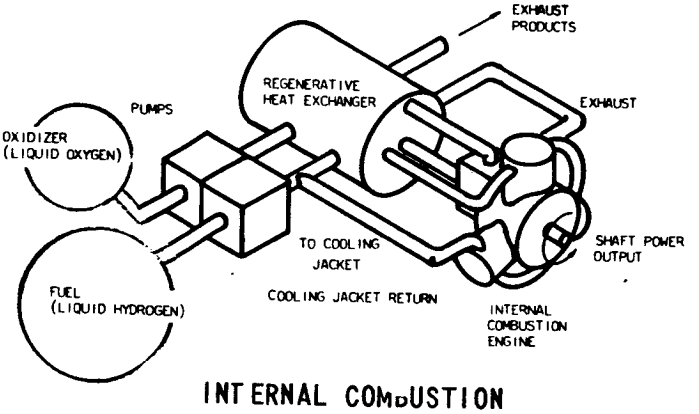

CANDIDATE POWER SYSTEMS
ORBITAL LAUNCH FACILITY
60 KW POWER LEVEL
100,000
10,000
1,000
100
Total System Weight - Lbs.
Hemispheric Shielding
Nuclear System
Spot Shielding
Fuel Cell (Emergency Power)
Solar Dynamic
System (Not
Available in Time for OLO)
Chemical Dynamic System
(Hydrox Engine) (Emergency Power)
0
20
40
60
80
100
Time in Hours

Fig. 4-4



INTERNAL COMBUSTION
REGENERATIVE POWER SYSTEM

Fig. 4 - 5

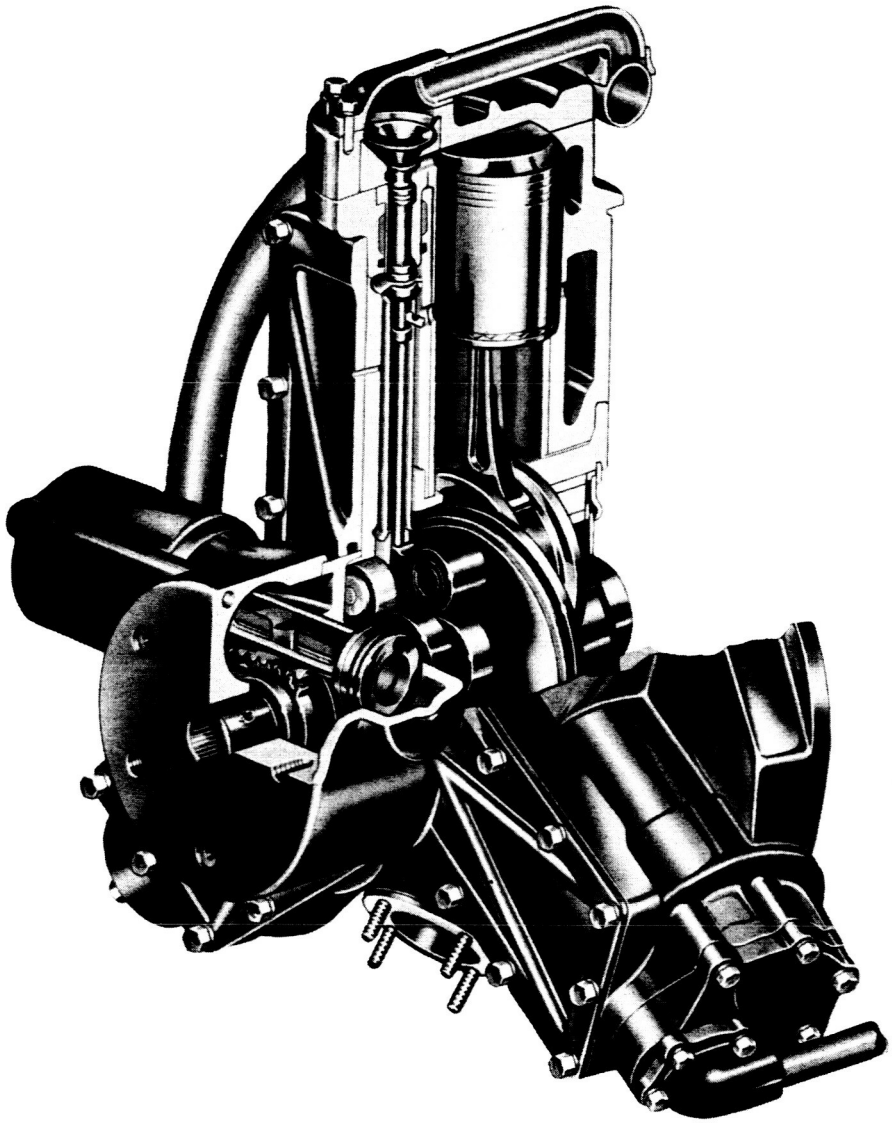
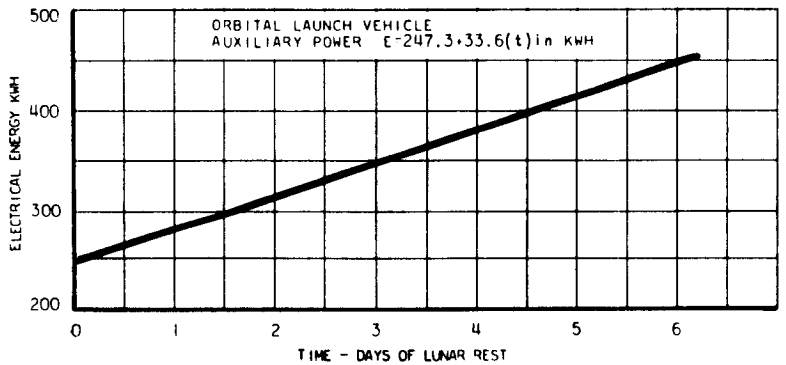
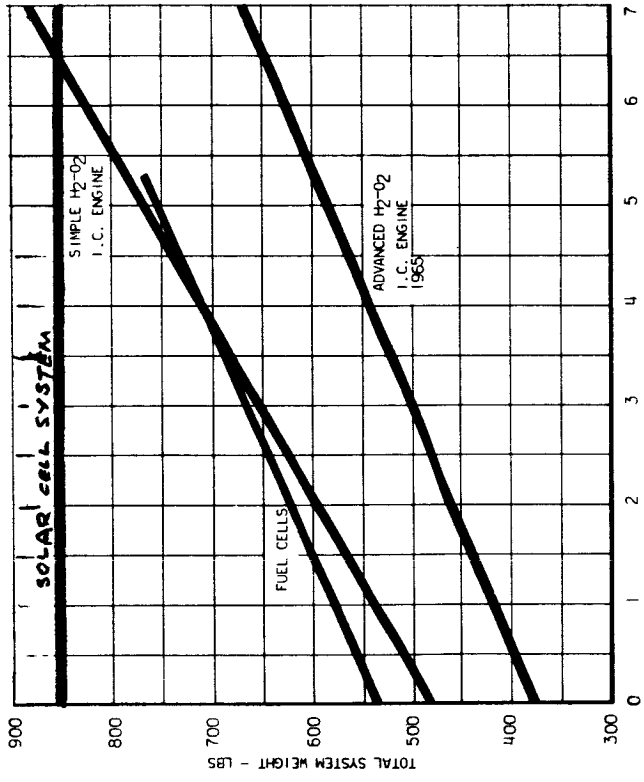


Fig. 4 - 6
Positive Displacement Engine



ELECTRICAL APS ENERGY REQUIRED
IN TERMS OF LUNAR REST

Fig. 4-7



WEIGHT VS. DAYS AT REST

Fig. 4 - 8

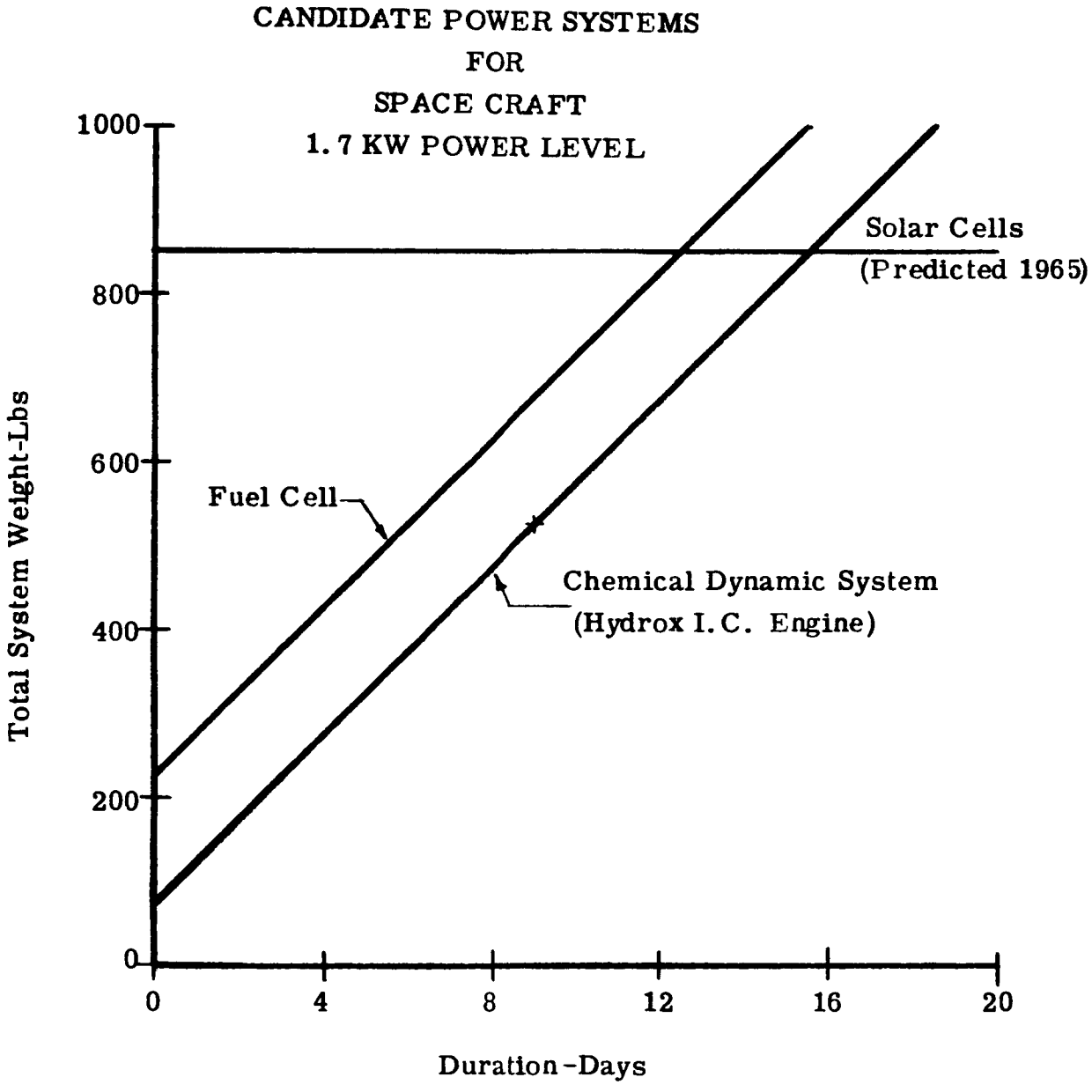
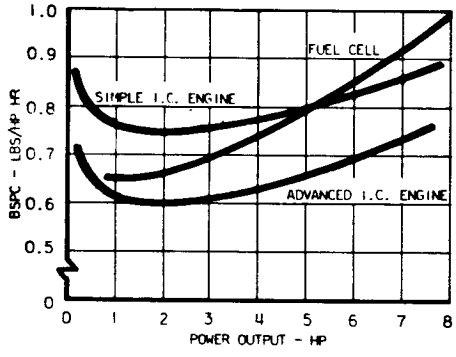

CANDIDATE POWER SYSTEMS
FOR
SPACE CRAFT
1.7 KW POWER LEVEL
Solar Cells
(Predicted 1965)
Fuel Cell
Chemical Dynamic System
(Hydrox I.C. Engine)
Total System Weight-Lbs
1000
800
600
400
200
0
0
4
8
12
16
20
Duration-Days

Fig. 4 - 9

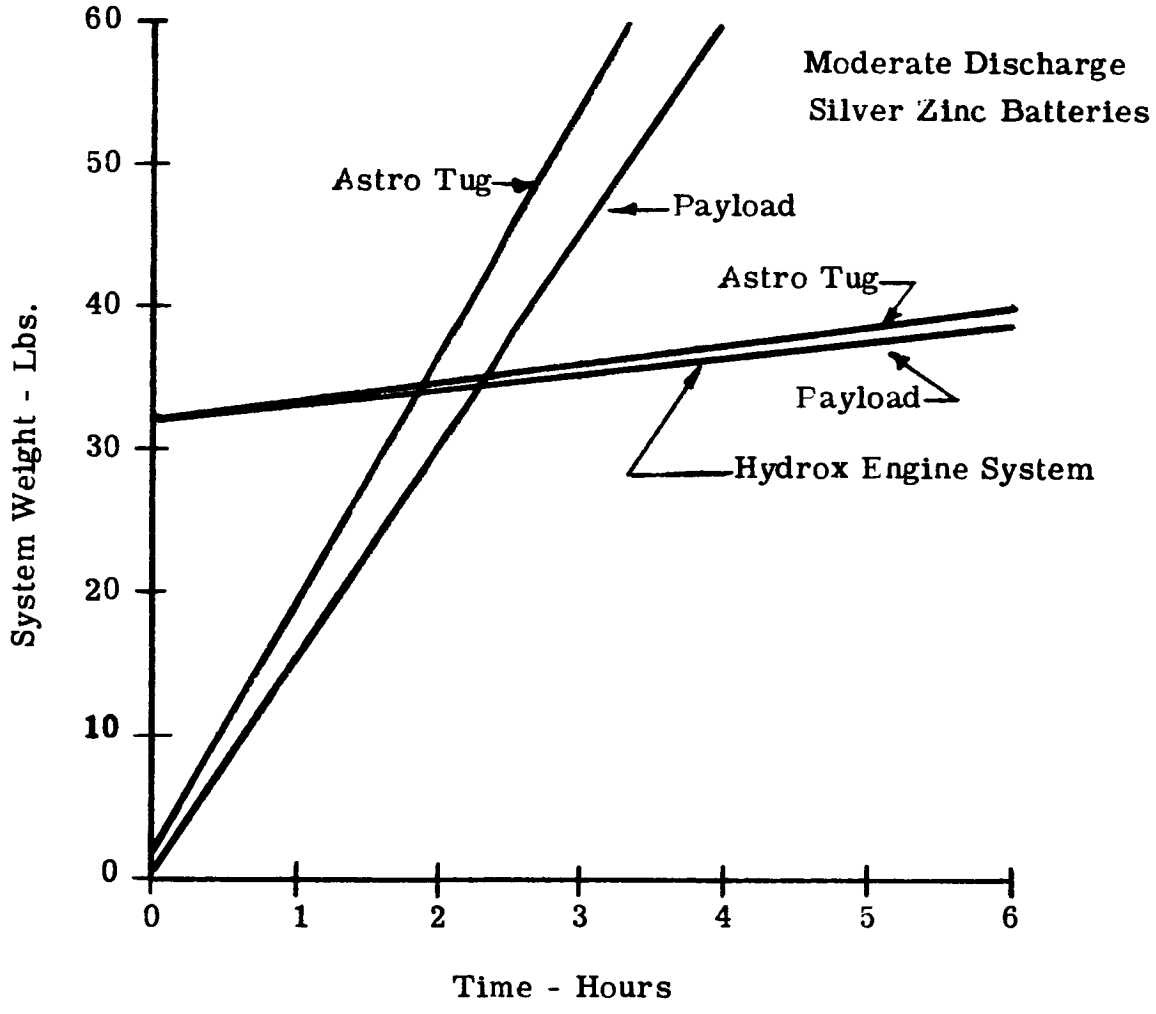


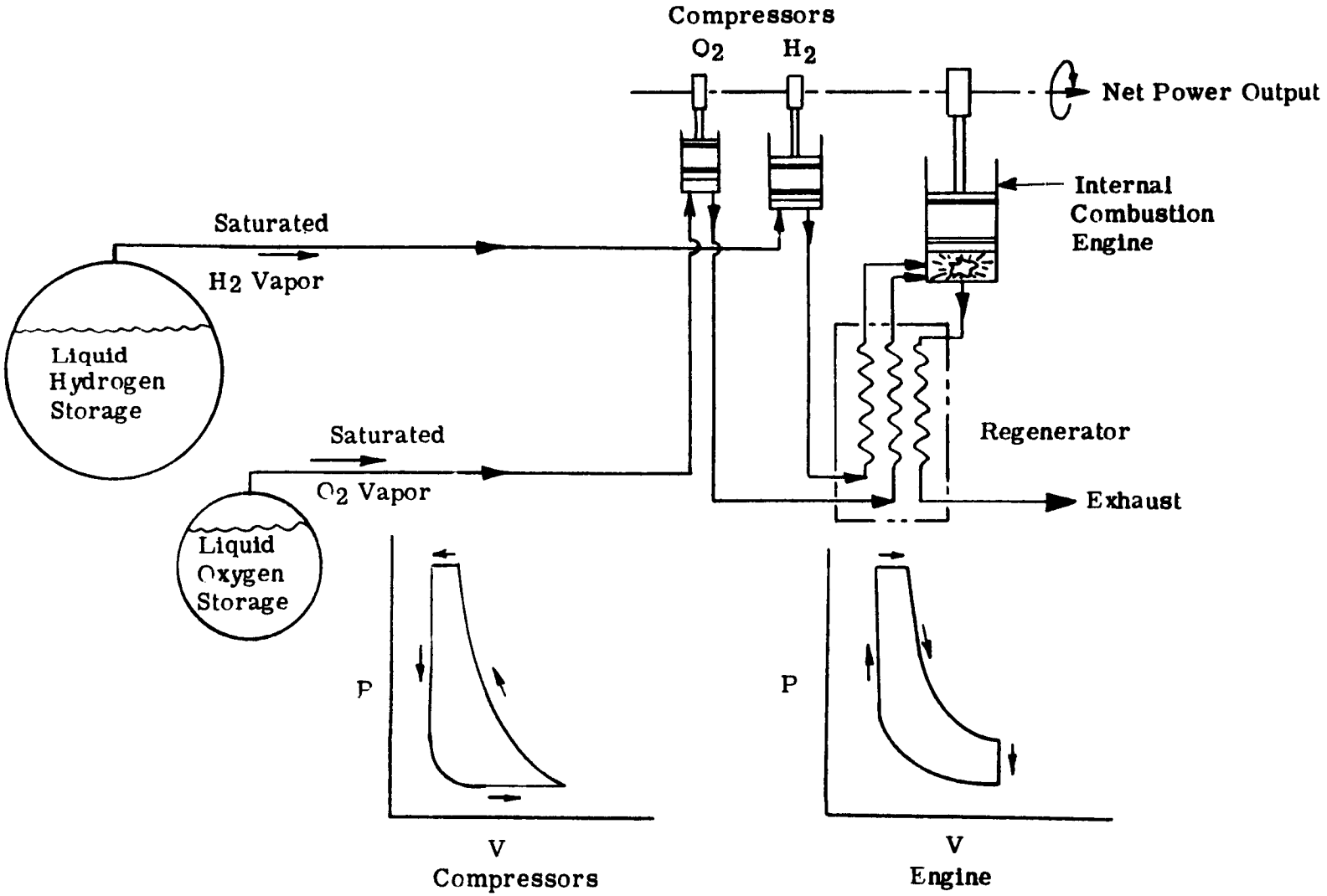
COMPARISON
OF BSPC TO POWER OUTPUT

Fig. 4 - 10

Fig. 4 - 11

ASTRO TUG & UPCOMING PAYLOAD





POWER SYSTEM USING PROPELLANT BOIL-OFF

Fig. 4 - 12

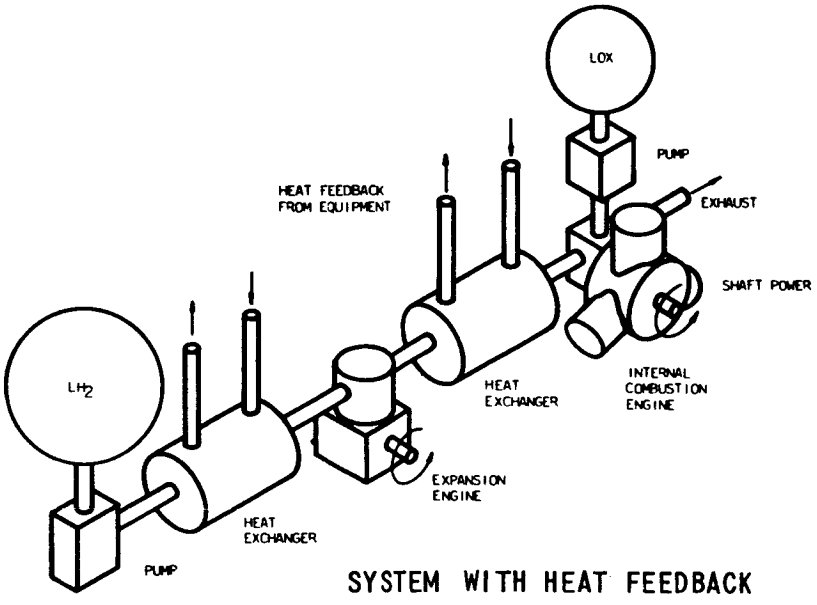

LOX
PUMP
EXHAUST
HEAT FEEDBACK
FROM EQUIPMENT
SHAFT POWER
INTERNAL
COMBUSTION
ENGINE
HEAT
EXCHANGER
LH_2
EXPANSION
ENGINE
HEAT
EXCHANGER
PUMP
SYSTEM WITH HEAT FEEDBACK

Fig. 4 - 13

INTERNAL COMBUSTION REGENERATIVE
POWER SYSTEM

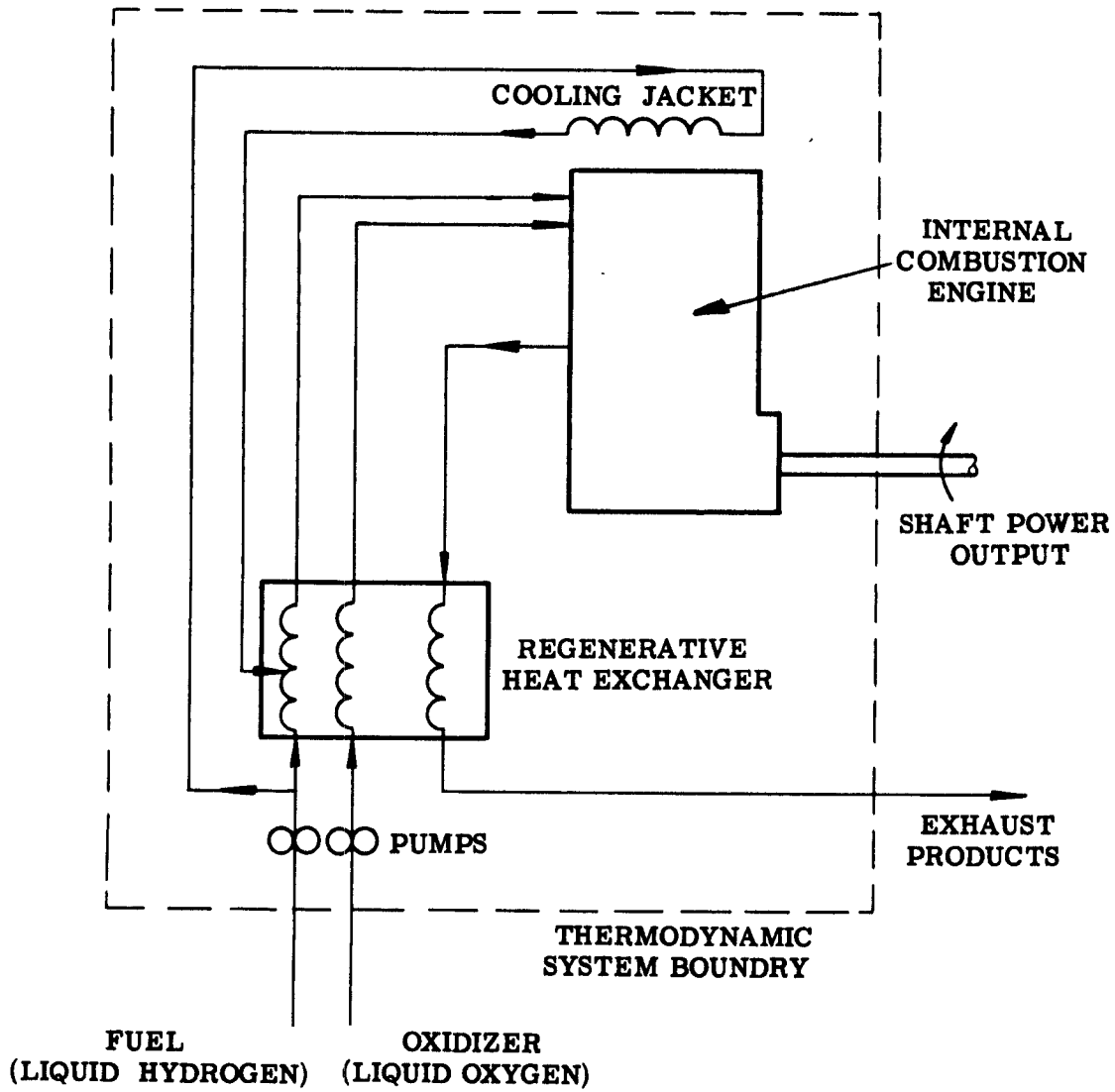


Fig. 4 - 14

5.0 - SCHEDULE

The S/C, OLF standby, payload, and Astro Tug power system development plan and schedules which follow indicate the type of program effort required before recommended power system could achieve operational status. This program would be a natural extension of Vickers present program, under an Air Force Contract, for development of an internal combustion hydrogen-oxygen engine for space power application. The work accomplished under the present contract allows the projected program schedule to be shortened correspondingly.

The R & D phase of the engineering development program can be subdivided into three areas. These are: 1) Evaluate current Air Force Contract work at Vickers to demonstrate feasibility of the internal combustion hydrox engine concept and obtain basic data; 2) Demonstrate feasibility of the regenerative power system concept; 3) Demonstrate feasibility, and develop to production prototype status, the advanced stoichiometric chemical power system. Beyond the R & D phase, additional effort is required in product engineering, service, reliability, qualification, logistics program support, and others to achieve true operational status. The total program is outlined below (and shown in Fig. 5-1).

Phase I R & D

- A. Current Air Force Contract work. (Engine only)
  1. Demonstrate feasibility of hydrox internal combustion engine concept.
  2. Obrain basic data on combustion, cooling requirements, etc.
  3. Determine performance at specific conditions.
- B. Basic power system development (system R & D work leading to feasibility demonstration of complete "breadboard" power system operating at fuel-rich mixture ratio).

1. Perform power system studies, thermodynamic analysis, and controls analysis; predict system performance; identify probable problem areas.

2. Test "breadboard" system; determine actual system performance; obtain basic data on system performance, stability, and general characteristics; resolve basic problem areas.

3. Demonstrate feasibility of complete regenerative power system operating on cryogenic propellants.

C. Advanced power system development (system R & D work leading to production prototype status of a complete power system designed to operate at stoichiometric mixture ratio).

1. Perform engine and power system studies, thermodynamic analysis, and controls analysis; predict system performance; identify probable problem areas.

2. Test "breadboard" system; determine actual system performance; obtain basic data on combustion characteristics, system stability, and general operational characteristics; verify and resolve basic problem areas.

3. Demonstrate feasibility of complete power system operating on cryogenic propellants at stoichiometric ratio.

4. Develop system to production prototype status; resolve prototype system problems.

D. Flight Testing

Conduct flight testing of prototype systems to demonstrate performance and reliability under actual space flight conditions. This is necessary to prove system capabilities when operating under zero G conditions in a hard vacuum environment for extended periods of time.

Phase II Operational Qualification

A. Operational system development
   1. System design
   2. Development Testing
B. Reliability program
C. Production engineering
D. Service engineering
E. Manufacturing
F. Logistics planning
G. Program support

The R & D work on a basic power system operating fuel rich, (shown above as Phase I-B of the development program) is considered a necessary transitional step between the present engine program and the complete advanced power system operating stoichiometrically. The development program schedule shows that the advanced power system could become operational by 1965. If an operational system is required before that time, efforts could be concentrated on the basic fuel-rich power system which could be made operational at least one year earlier. With an early contractual authorization and sufficient support, Vickers believes that the basic hydrox engine power system could be made operational by the end of 1963, as shown in Fig. 5-2.

PROGRAM SCHEDULE, STOICHIOMETRIC SYSTEM

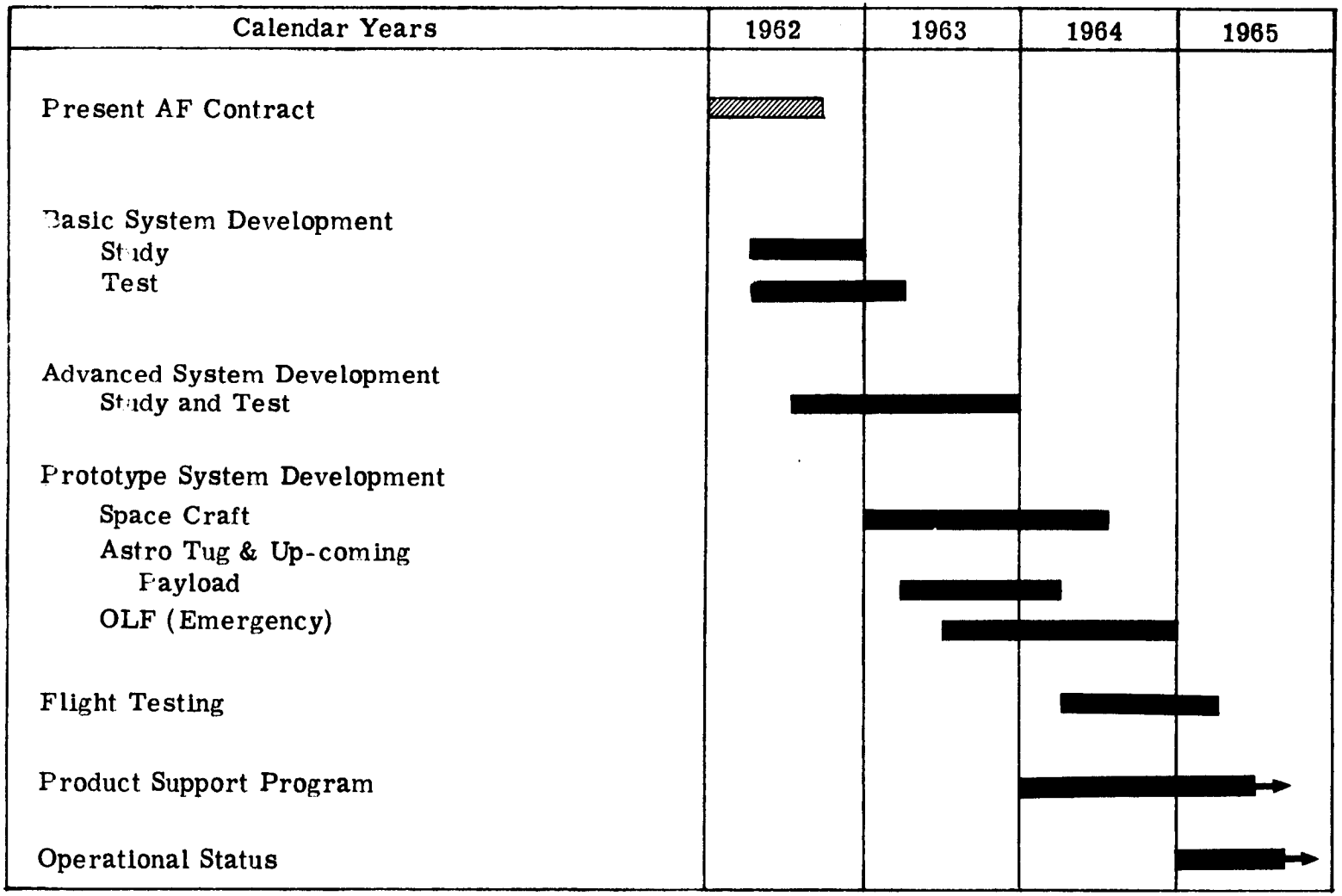


Fig. 5 - 1

PROGRAM SCHEDULE, FUEL RICH SYSTEM

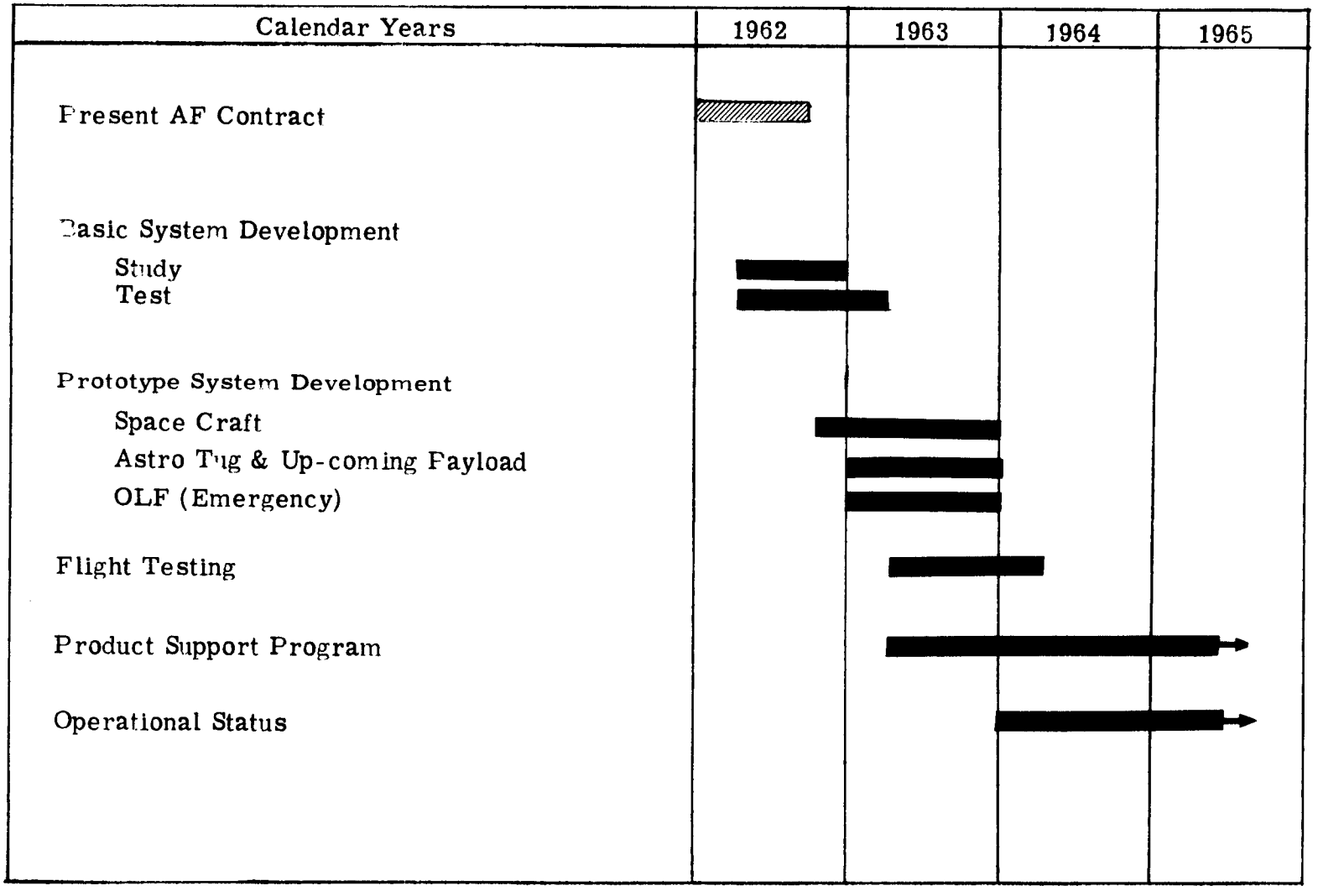


Fig. 5 - 2

6.0 - PROBLEM AREAS

The problem areas associated with a cryogenic hydrogen-oxygen internal combustion engine are listed below. These will require the most development and will prove to be the limiting factors with respect to engine reliability, cost, complexity, and economy of operation.

1. Lubrication

2. Injector and valve design

3. Combustion

4. Zero Gravity Tankage

A wealth of background information, much of which is directly applicable, can be obtained from the literature on air-breathing internal combustion engines. A high degree of mechanical sophistication is permissible in a well developed system if enough redundancy is used to ensure reliability.

Some solutions to the problems mentioned above are as follows:

6.1 Lubrication - High temperature lubricants such as polyphenyl ethers are being investigated for operation in the two-stroke-cycle internal combustion hydrogen-oxygen engine currently under development at Vickers. The results of this investigation will be directly applicable. Open-cycle and closed-cycle lubrication will both be considered, and a choice will depend on lubricant consumption, and relative complexity and weight.

Although successful operation of a lubrication system under actual Zero "G" conditions for an extended duration has yet to be demonstrated, there are lubrication techniques which appear very feasible for such applications. Vickers has devoted considerable study to the problem of Zero "G" lubrication methods for both engines and gear boxes. Among the types studied have been sealed systems, open cycle mist systems and recirculating systems. The sealed lubricant type system appears particularly adaptable to gear boxes. A quantity of lubricant and gas is sealed in the gear box. The gas forms the transport medium for circulation of oil mist. Windage forces resulting from the rotating gears insures circulation of oil mist throughout the gear box. Open cycle and recirculating lube systems have usually been considered for piston engines in space power system designs. A metered-splash system similar to techniques used for many years in engines and other machinery, appears very attractive for zero G applications.

6.2 Injector and Valve Design - High inlet temperature and pressure, adverse valve and/or injector operating environments, and stringent operating time and tolerance requirements create a difficult material and mechanical design problem for a propellant inlet mechanism. This problem is under investigation at Vickers in the program mentioned above. A high level of design sophistication in these areas is permissible, and much design experience is available.

Combustion - Detonation and pre-ignition problems have been experienced in internal combustion engines operating on hydrogen and air due to high combustion rates. However, variations in inlet pressure, mixture ratio, injection rate, and the possibility of premixing all or part of the combustible mixture leaves a large degree of latitude in the problem approach. This again points to the desirability of a well developed engine which can be extensively tested prior to incorporation into the system.

6.3 Temperature - The primary difference between the fuel rich and the stoichiometric engines lies in the peak cycle temperature and system complexity. An extensive investigation into the use of exotic materials, construction methods, and lubricants will be necessary to ensure reliability at the proposed stoichiometric combustion temperatures.

The internal combustion piston engine has a definite advantage over turbomachinery in system where high thermal efficiency is important because peak cycle temperatures in a reciprocating engine can be relatively high with moderate average temperatures.

6.4 Zero Gravity Tankage - Propellant storage at very high pressures would involve a prohibitive weight penalty in a space vehicle. Therefore, the propellant must be stored at low pressure, then boosted to operating pressure by means of a pump or a compressor. Two basic concepts in storing cryogenic fluids have been considered to date:

1. Storage at supercritical pressure
2. Storage at low pressure with positive expulsion

Storage at Supercritical Pressure - Storage at supercritical pressure (225 psia for hydrogen and 800 psia for oxygen) is not likely to be satisfactory for long duration space missions - up to 300 hours for example. The supercritical storage system suffers a weight penalty as compared to low pressure storage because of the increased tank shell thickness necessitated by elevated fluid pressures. Furthermore, the density of the supercritical fluid decreases with time

throughout the mission due to the heat input to the tank required to maintain the desired pressure level as fluid is withdrawn from the tank. Therefore, to meet the high inlet pressure requirement of the hydrox engine, a compressor capable of operating efficiently over a wide range of inlet densities would be required. The development of a variable density compressor for cryogenic fluids would be considerably more difficult than development of a small liquid pump to achieve the same pressure level. Furthermore the average compressor power requirement would be several times that required for pumping a liquid.

Storage at Low Pressure with Positive Expulsion - The most common means of positive expulsion considered to date for low pressure cryogenic storage tanks has been either a bladder or diaphragm within the tank. Although this type of tank appears to have a lower weight than the supercritical tank for mission durations of a few hours, the weight penalty resulting from increased insulation requirements for longer missions must be considered. This insulation thickness must be sufficient to keep the heat leak into the tank low enough to avoid formation of a two phase condition. Therefore, for long missions, the insulation weight may become prohibitive.

Because of the limitations of the bladder or diaphragm concept in positive expulsion, there is need for a low pressure cryogenic tank capable of expelling a single phase liquid at all times. - even in the presence of considerable vapor within the tank. Such a tank would avoid the structural weight penalty of the supercritical tank and the insulation weight penalty of the bladder type tank.

Vickers Incorporated has completed a preliminary analysis of a concept where an artificial "g" field is created by rotating the fluid within the tank to maintain liquid - vapor phase separation under random or zero "g" conditions. In this concept, a rotary motion of the fluid is induced to set up the artificial "g" field. Preliminary study results indicate that evaporative losses due to liquid motion can be held to comparatively low values for extended periods of operation, and the power requirements for maintaining fluid motion are extremely small.

As the mission duration increases it becomes increasingly important to reduce the fluid drag losses and hence fluid boil off. It is quite conceivable that fluid rotational velocities corresponding to 0.1 g may prove to be adequate for liquid-vapor separation during the major portion of the mission where a zero "g" external environment exists. Prediction of tank performance at low "g" rotational speeds will require further studies and correlation of experimental data on fluid drag.

7.0 - CONCLUSION

The conclusion reached by Vickers Division of Sperry Rand as a result of their study of secondary power requirements for the OLO program are as follows:

Orbital Launch Facility

1. The nuclear Snap 8 power system is recommended as being the only acceptable system expected to be available within the projected time scale of the OLO program. This system has the disadvantage of requiring heavy shielding to provide radiation protection for the OLF crew.

2. Recent developments in solar dynamic systems indicate large potential weight savings compared to nuclear power systems. However, it does not appear that such systems could be operational soon enough for the first OLF's. Technical progress in this area should be continuously monitored so that solar dynamic power can be provided for later versions of OLF if successful development is achieved.

3. For emergency standby power in the OLF, a chemical dynamic power system incorporating an internal combustion engine and using either cryogenic hydrogen and oxygen or storable bi-propellants offers a significant weight advantage over other candidate emergency systems for operating durations of several days.

Space Craft, Astro Tug, and Upcoming Payload

1. Chemical dynamic power systems using a cryogenically fueled internal combustion engine (hydrox engine) offer substantial weight saving over other candidate systems for these OLO vehicles.

2. The power requirements of the Astro Tug and the Upcoming Payload are similar enough to allow the same APS to be used for both vehicles, thus substantially reducing development cost. The Space Craft APS would be similar to that of the other two vehicles but capable of higher power level and longer operating duration.

3. Additional vehicle weight reduction may be possible by integrating the manned vehicle APS with the environmental system. Also consideration should be given to operation of the APS from rocket engine propellant tank boil-off.

Recommended Action

1. Study shielding requirements of OLF relative to facility configuration to provide maximum radiation protection with minimum weight.

2. Monitor solar dynamic power system developments for possible future application to OLF.

3. Conduct more penetrating studies of secondary power requirements of all OLO vehicles. Investigate advantages of integration of APS with vehicle Environmental Control System and life support requirements.

4. Initiate follow-on programs to Vickers current hydrox engine development contract with Air Force ASD. The support of NASA - MSFC is solicited in support of proposals for follow-on programs currently being evaluated by NASA Headquarters, NASA - Lewis Research Center, and Air Force ASD. Early initiating of these follow-on programs is necessary to provide operational APS ' within the time scale of the OLO program.

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APPENDIX A - SUPPLEMENTARY INFORMATION ON SHORT-DURATION SECONDARY POWER SYSTEMS.

Hot Gas Motor APS - Although hydraulic pumps driven by electric motors or flywheels are satisfactory for many low-power, short-duration applications, Vickers has placed accelerating emphasis on the hot gas power concept because of its many attractive advantages in space.

Concept - The Vickers hot gas motor APS consists of a pressure-regulated solid-propellant gas generator powering a modified Vickers positive-displacement motor that drives a standard Vickers pump. Shown schematically in figure 3-A1, this system combines the advantages of compact propellant energy with the inherently high efficiencies of positive-displacement machines driven at moderate rotating speeds. A preliminary study of power versus specific weight is presented in figure 3-A2.

Design - A photograph of the hot gas APU (integrated motorpump package) is presented in figure 3-A3. The hot gas pressure regulator that was developed for the system is shown in figure 3-A4. The pump is a standard unit with approximately one-half the displacement of the motor so that 1500 PSI gas pressure can be converted into 3000 PSI hydraulic pressure.

Several modifications were necessary to convert the basic Vickers hydraulic motor for efficient and reliable operation with high temperature gas. New materials with high wear resistance and nearly identical thermal expansion characteristics were selected for the cylinder block and pistons. The shaft seal was omitted, and a special valve plate was provided to withstand friction and temperature conditions.

The gas generator is of simple cylindrical construction, incorporating

an igniter squib and rupture diaphragm. The generator is designed to use solid propellants, including the single-base ammonium nitrate and the double-base nitroglycerin nitrocellulose. Many tests have proven the suitability of using either type of propellant charge. Motor inlet filtration has been found unnecessary using these propellants since the only contaminant produced by combustion is a soft carbonaceous substance that does not impair motor operation.

In addition to the motorpump APU concept, Vickers has developed motors that operate exclusively on hot gas to power mechanical devices. One of these motors is discussed later under "Applications".

Operation - An electrical signal to the gas generator igniter initiates propellant combustion. The grain combustion rate is constant at the constant pressure maintained by the pressure regulating valve. The high pressure gas (e.g. 1500 psi) preloads a hydraulic accumulator and the motor inlet. The accumulator thereby maintains gas system pressure in the pump motor case. The motor outlet is open to atmosphere, and case pressure acting behind the motor pistons causes them to reciprocate by virtue of the differential pressure across the motor inlet and outlet ports (i.e., the opposing pressures on the pistons on the inlet side are balanced, whereas a 1500 PSI differential exists on the outlet pistons). The motor drives the pump through a common shaft, producing hydraulic power.

The unique feature of driving motor pistons from the back side prevents gas "blow-by" past the pistons since opposing pressures are equal. Any oil leakage through the motor outlet is negligble compared with gas

leakage under the same conditions, and volumetric efficiency is optimized. This feature also provides a sink to dissipate heat from the motor components.

Applications - Vickers and the United States Air Force invested over a half-million dollars in a development program with the conviction that the hot gas motor auxiliary power system (APS) concept represented a significant advance in the state of the non-propulsive power art. A contract was awarded to Vickers by the Thiokol Chemical Corporation, working on the control and propulsive systems for the Minuteman ICBM. An accelerated research and development program was initiated to produce a compactly integrated APS within a seven-month period.

Twelve breadboard units were built, and ninety-one tests were conducted in 30 days to establish motorpump reliability. Twelve prototype APS systems were then fabricated to substantiate system reliability through a series of 123 operational tests. The flight hardware system is shown in figure 3-A5.

Aside from testing of the self-sufficient APS, a program was initiated to establish the feasibility of operating a motorpump auxiliary power unit (APU) with gas bled directly from the Thiokol rocket engine. A successful bleed-gas test was conducted utilizing a Thiokol TU-132 engine to power the hardware. A peak of 2.15 hydraulic horsepower was developed after the hydraulic flow reached steady-state of 0.54 seconds and remained at 5.2 GPM for the duration of the test. Clogging of the gas pickup terminated the test after 16 seconds of operation.

The experiment was so successful that Thiokol sponsored a second feasibility test performed several months later with the test rig shown in figure 3-A6. Results of the second test definitely established the feasibility of operating the Vickers APU at high temperatures with highly contaminated gas.

The motorpump ran for the full 62-second burning duration of the Thiokol main engine propellant charge. Motor inlet temperatures were recorded up to 3600°F with gas solid-contamination estimated at 10 to 15 percent by weight. A breakdown inspection of the motorpump revealed all parts to be in excellent condition (including the motor pistons which moved freely in the cylinder bores). A peak output of 2.35 horsepower was developed during this test with a steady-state hydraulic flow of 5.2 GPM.

Additional laboratory testing was conducted to simulate maximum temperature bleed operation using a near-stoichiometric bipropellant mixture of hydrogen and oxygen. The APU operated for 39 seconds at propellant flame temperatures estimated between 5000° and 5200°F, until a stainless steel line directing gas from the combustor to the motor ruptured. The motor components were found in excellent condition as shown in figure 3-A7.

The hot gas motorpump shown in figure 3-A8 is currently being used on the Skybolt missile. The 100-horsepower hot gas motor presented in figure 3-A9 was produced for General Electric. Delivery of prototype hardware required only ten weeks (sixty days) from receipt of the specifications. Basic design, testing and modifications, final prototype assembly, and delivery were accomplished within this period.

This unit has undergone an acceleration rate of 50,000rpm/sec. in the G.E. application at essentially constant torque over its entire speed range. The motor is designed to operate on either a double-base solid propellant or on a liquid monopropellant, although all runs to date have been powered by the double-base solid propellant.

Battery-Powered APS - Although the potential applications of battery powered auxiliary power systems are limited in the OLO concept, they may be suitable for any short duration boosters associated with the program. A brief discussion of this approach to providing auxiliary (or primary) hydraulic power is included.

Concept - A schematic diagram of an integrated battery-powered hydraulic system is presented in figure 3-A10. The essential components of the system are silver-zinc batteries, a d-c motor driving a hydraulic pump, a fluid reservoir, and valves. A preliminary study of power versus specific weight is presented in figure 3-A11.

Design - Many APS designs based on this concept employ the "building block" principle whereby "off-the-shelf" electric motors are mounted to "off-the-shelf" hydraulic pumps. Where space and weight parameters require a more sophisticated approach, complete integration of the APS components can be achieved as shown in the schematic diagram.

A photograph of an integrated battery APS used to provide hydraulic power for the Minuteman missile is shown in figure 3-A12.

Vickers has been producing airborne electrical motorpump packages for over 15 years.

Applications - Other recent applications include d-c motorpumps used to power hydraulic systems for the Pershing, Titan, Bomarc, and Polaris missiles. The Polaris motorpump, shown in figure 3-A13 is a completely integrated unit with a special uncured latex coating that absorbs ambient heat (in curing), thereby protecting the unit from its high temperature surroundings.

Flywheel APU - Concept - Flywheel-powered APU's are attractive for many specialized applications in an orbital launch operation. The system consists essentially of a flywheel, hydraulic pump/motor, reservoir, and valves. These elements are related as shown in figure 3-A14.

By using the pump/motor concept, the flywheel absorbs energy whenever system pressure rises above the nominal value and delivers exact quantities of power when the pressure falls below that value.

Design - The highest specific energy storage capability for a given maximum flywheel stress is theoretically achieved with a uniformly stressed rimless disc. However, this design requires top speeds that cannot normally be attained within practical size and speed limits. Furthermore, windage losses are highly sensitive to speed with a uniform-stress disc, dissipating a large proportion of the total stored energy in most applications. These considerations have led to the heavy rim flywheel for the unit shown in figure 3-A15 that was designed for the Atlas missile.

The flywheel for this design is shrouded to accommodate air evacuation, minimizing windage losses. The pump/motor employed is essentially a standard fixed-angle variable delivery unit equipped with a special valve block to enable two-way operation (pumping and motoring).

A generalized weight study comparing specific system weight (not including flywheel weight) with rated output power is presented in figure 3-A16. The flywheel weight depends on operating duration. Based on a tip speed of 1500 fps and allowing for windage losses, the duration factor for a flywheel is 150 lb/hp-hr. For example, a ten-horsepower system of five-minute duration will weight 135 pounds (125-lb flywheel). A similar system for five seconds duration will weigh only 12.1 pounds (2.1-lb flywheel).

Operation - During run-up, hydraulic power is used to accelerate the flywheel to rated speed. This is accomplished by driving the flywheel-mounted hydraulic pump as a motor, using power either from ground supply or from vehicle supply (if the flywheel system is to be used later in the mission).

The flywheel-mounted hydraulic pump comprises a unit that has the characteristics of a highly efficient accumulator. A constant-pressure hydraulic output is achieved if the pump/motor unit is pressure-compensated to provide variable delivery, thereby compensating for flywheel energy degradation.

Application - The flywheel APU is particularly attractive in staged vehicles. For example, a flywheel system can be installed in the second (or third) stage of a vehicle, and the first-stage primary hydraulic system can "charge" the flywheel APU during any period before first-stage jettison when hydraulic demands are low. The second stage is thereby endowed with an independent hydraulic power supply without paying space and weight

penalties imposed by fueled prime movers or batteries.

As mentioned previously, the flywheel APU can also be used as an accumulator in a constant pressure system where the average hydraulic power demand is low compared with intermittent peak requirements. The flywheel is "charged" during low power demand periods and is available to supplement the main hydraulic power supply when needed. The primary advantage over other accumulator devices is that compensation for energy degradation can be achieved by using a variable displacement pump/motor unit.

Availability - Since the flywheel APU is composed of relatively standard equipment, the only development necessary to achieve a production unit concerns selection of a flywheel size and configuration compatible with the mission parameters.

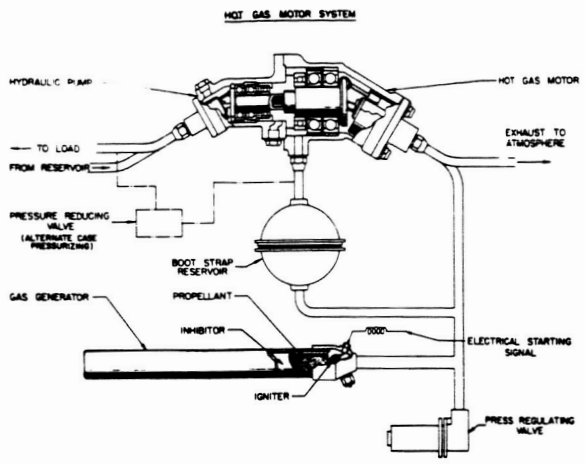


FIG. 3-A1- HOT GAS MOTOR SYSTEM

FIG. 3-A3- HOT GAS MOTORPUMP

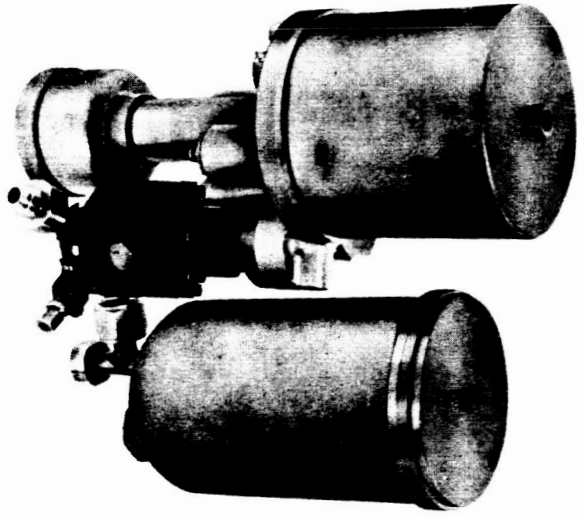
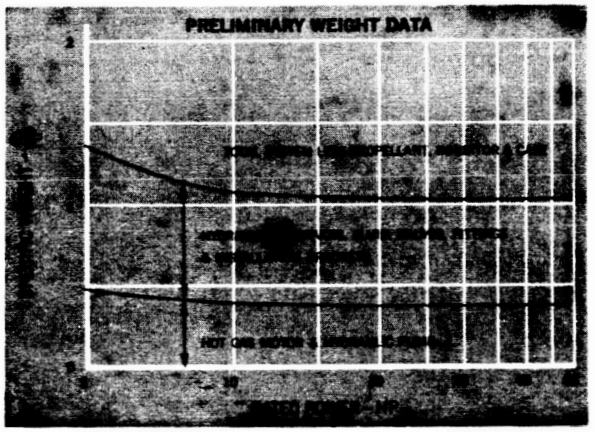


FIG. 3-A5- INTEGRATED HOT GAS APS



The duration factor for a short duration hot gas motor system utilizing a solid propellant is 27 lbs/hp-hr, which includes propellant, inhibitor and case weight. Therefore, as an example, a 10 hp system of 5 minutes duration will weigh 33.5 lbs. A 10 hp system of 5 seconds duration will weigh 11.4 lbs. For durations above 5 minutes, a partial admission motor with a duration factor of 18 lbs/hp-hr. would be used.

FIG. 3-A2

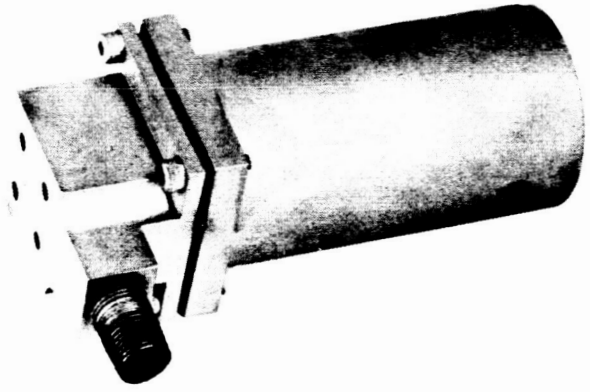


FIG. 3-A4- HOT GAS PRESSURE REGULATOR

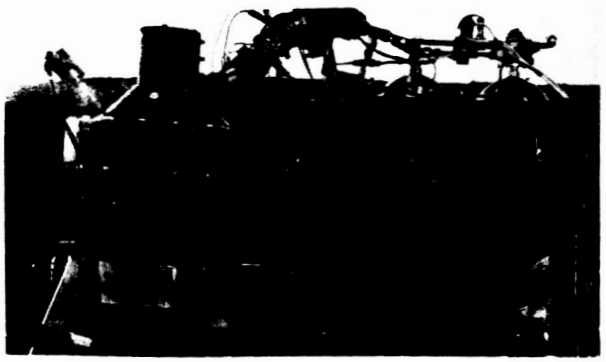


FIG. 3-A6 - HOT GAS BLEED TEST

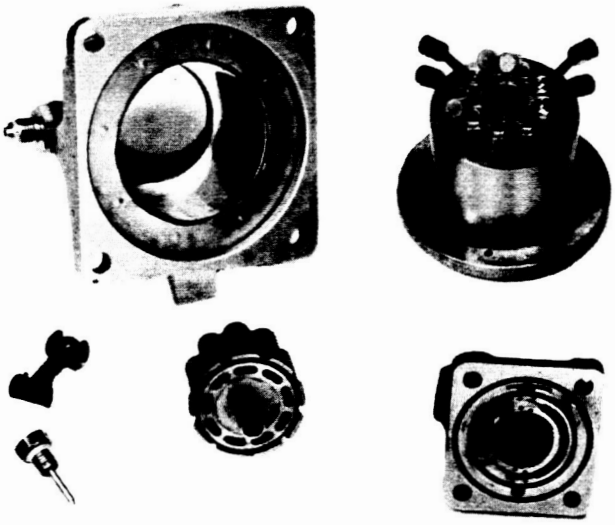


FIG. 3-A7 - HOT GAS MOTOR AFTER OPERATION AT 5200°F

FIG. 3-A8 - HOT GAS MOTORPUMP FOR SKYBOLT

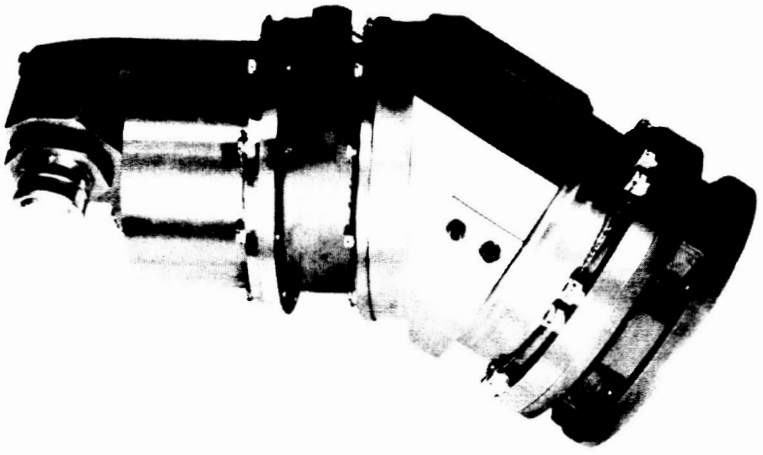


FIG. 3-A9 - 100 HP HOT GAS MOTOR

SCHEMATIC OF INTEGRATED BATTERY POWERED SYSTEM

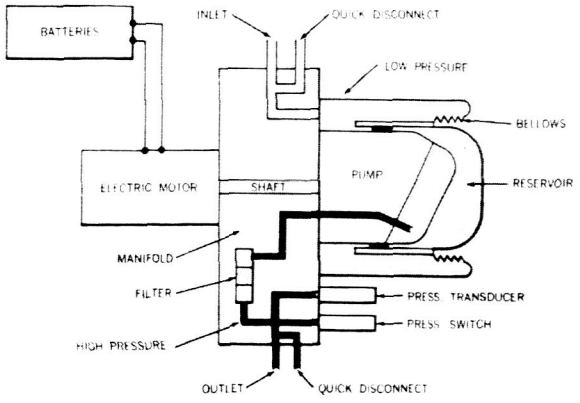
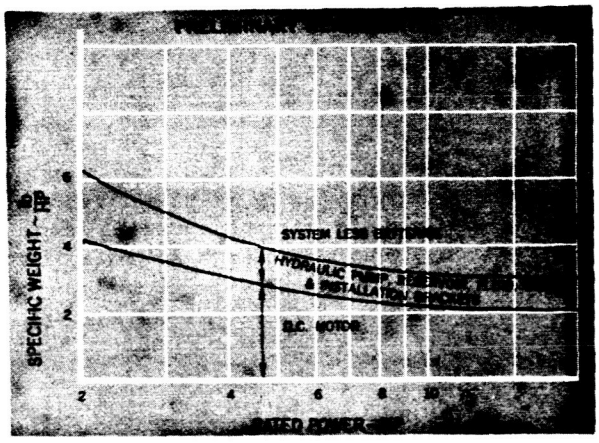


FIG. 3-A10



The weight factor for a silver-zinc battery includes 1.0 lb/hp fixed weight and a duration factor of 15 lb/hp-hr. Therefore, a 1 lb/hp must be added to the d-c motor and hydraulic components specific weight shown on the curve at the left to obtain fixed system weight. As an example, a 10 hp system of 5 minutes duration will weigh 54.5 lbs. A 10 hp system of 5 seconds duration will weigh 42.2 lbs.

FIG. 3-A11

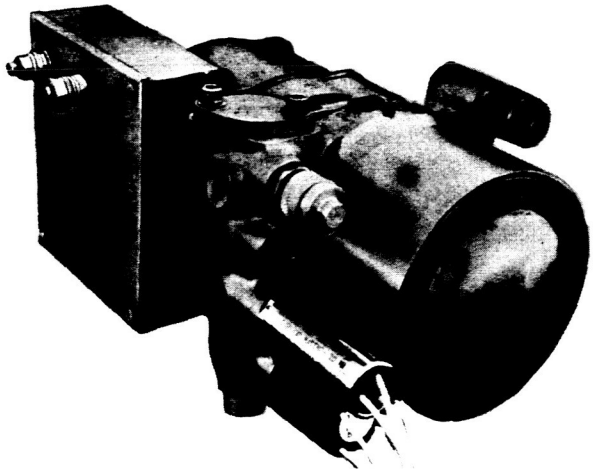


FIG. 3-A12 - INTEGRATED BATTERY APS FOR MINUTEMAN

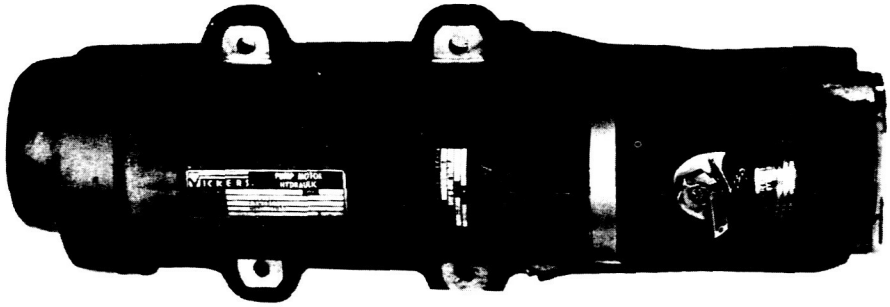


FIG. 3-A13 - ELECTRIC MOTORPUMP FOR POLARIS

SYSTEM SCHEMATIC

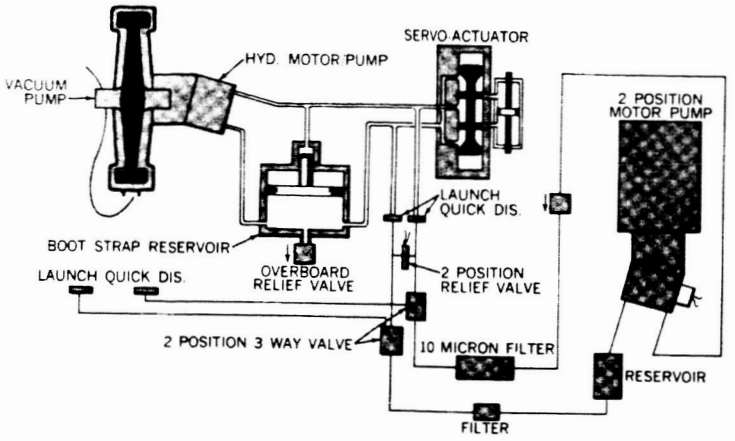


FIG. 3-A14

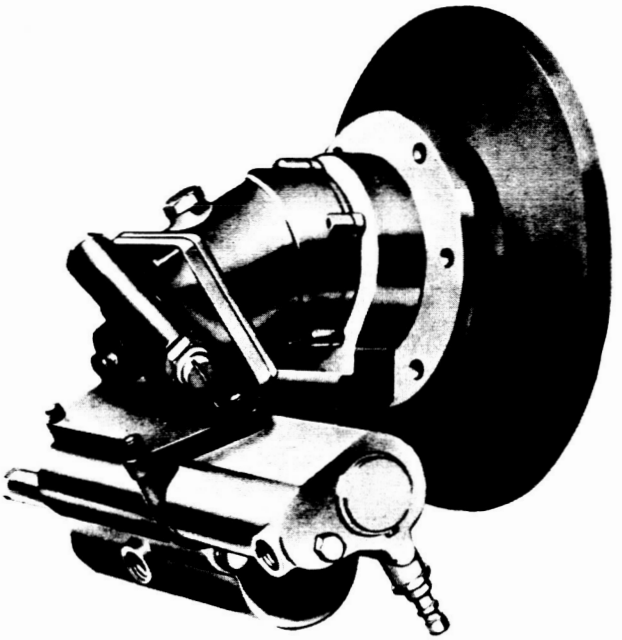
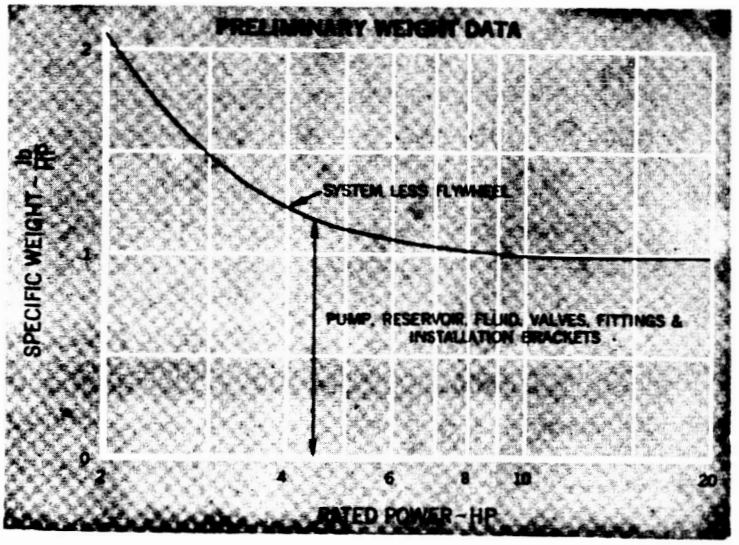


FIG. 3-A15 - FLYWHEEL APU FOR ATLAS

FIG. 3-A16



APPENDIX B - DEFINITIONS

1. APS: Auxiliary or Secondary Power System; a device to generate hydraulic, mechanical or electrical energy for any components or subsystems other than main propulsion.

2. BSPC: Brake Specific Propellant Consumption (Lbs/HP-hour)

3. OLO: Orbital Launch Operations

4. OLV: Orbital Launch Vehicle; vehicle used to launch spacecraft from earth orbit.

5. OLF: Orbital Launch Facility; an earth-orbiting facility used for platform assembly and launch of spacecraft and OLV.

6. LOF: Lunar Orbiting Facility; a lunar facility that serves as the orbital terminal for lunar base supplies and personnel.

7. LDV: Lunar Descent Vehicle; vehicle used to shuttle cargo and personnel between LOF and Lunar Base.

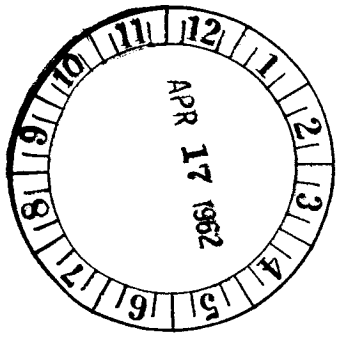
8. MLM: Manned Lunar Landing and Return Mission

9. Apollo A: 3-man Apollo spacecraft control module with specific equipment required for suborbital and orbital flights.

   Apollo B: 3-man Apollo spacecraft, control, equipment and propulsion modules for circumlunar and lunar orbital flights.

   Apollo C: 3-man Apollo spacecraft, control, equipment and propulsion modules including all equipment required for the lunar landing mission.

   Apollo D: 6-man Apollo-type spacecraft.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